AIRPLANE MAINTENANCE MANUAL

(RB050002J)

Cessna 350

Model LC42-550FG

THIS HANDBOOK INCLUDES THE MAINTENANCE INFORMATION REQUIRED TO BE AVAILABLE BY FEDERAL AVIATION REGULATIONS PART 23

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MAINTENANCE MANUAL
LOG OF REVISIONS

The Log of Revisions pages are updated by Cessna each time revisions are issued. The Log of Revisions contains a list of all revisions made to the maintenance manual since its original issue. By comparing the Log of Revisions to the Record of Incorporated Revisions, the user of this manual can verify that all applicable revisions are included.

Revision Number	Date Issued	Affected Chapters	Affected pages	Approved By
Original Issue	3/12/2003	N/A	N/A	N/A
A	3/28/2003	Ch. 4, 5, 20, 21, 28, 32, 73, 74, 80	See Narrative Discussion of Revisions	N/A
B	05/20/03	All Chapters	See Narrative Discussion of Revisions	N/A
C	05/23/03	Ch. 4, 24, 32 and 34	See Narrative Discussion of Revisions	Cynthia Cole 05/30/03
D	09/24/03	Ch. 1, 2, 3, 4, 5, 11, 12, 20, 21, 22, 23, 24, 25, 27, 28, 32, 33, 34, 53, 56, 57, 61, 71, and 76.	See Narrative Discussion of Revisions	Luann Schweitzer 09/29/03
E	02/10/04	Pgs. i, and ix through xxviii, Ch. 4, 5, 20, 21, 22, 24, 25, 27, 28, 31, 32, 33, 34, 51, 56, 61, 71, 76, 77, and 78.	See Narrative Discussion of Revisions	Luann Schweitzer 02/11/04

F	02/02/05	Pgs. i, ii, ix through xxxiv, Ch. 4, 5, 12, 20, 21, 23, 24, 25, 27, 28, 32, 33, 34, 35, 51, 53, 57, 71 and 77.	See Narrative Discussion of Revisions	Luann Schweitzer 02/03/05
G	06/27/06	All	See Narrative Discussion of Revisions	Luann Abrams 06/27/06
H	10/10/06	Pgs. ii, xlii to xliv, Ch. 4, 5, 20, 21, 23, 24, 25, 34, 52	See Narrative Discussion of Revisions	Luann Abrams 10/10/06
I	12/07/07	All	See Narrative Discussion of Revisions	Tom Bowen 12/07/07
J	01/08/08	Pgs. ii, xxxiii to lii, Ch. 2, 4, 5, 6, 11, 12, 20, 21, 22, 23, 24, 25, 27, 28, 30, 32, 33, 34, 51, 52, 53, 56, 57, 71, 76, 77	See Narrative Discussion of Revisions	Tom Bowen 01/08/08

MAINTENANCE MANUAL
RECORD OF INCORPORATED REVISIONS

All revisions must be incorporated into the maintenance manual as soon as possible. The pages that are removed should be destroyed. The Record of Incorporated Revisions is maintained manually by inserting applicable information in the table below.

Revision No.	Date of issue	Date of Insertion	Inserted By	Revision No.	Date of issue	Date of Insertion	Inserted By

Revision No.	Date of issue	Date of Insertion	Inserted By	Revision No.	Date of issue	Date of Insertion	Inserted By

MAINTENANCE MANUAL
RECORD OF TEMPORARY REVISIONS

Temporary revisions are issued to facilitate timely changes of important information and are printed on yellow paper. The revisions are numbered sequentially and can affect more than one chapter. Temporary revisions are normally incorporated in the next regularly scheduled revision cycle. When a temporary revision is received, the first four columns below are used to document its incorporation into the maintenance manual. Later, when the temporary revision is superseded by incorporation as a scheduled revision, the temporary revision is removed from the manual and appropriate notations are made in the last two columns.

Temporary Revision No.	Date of Issue	Date of Insertion	Inserted By	Date of Removal	Removed By

Temporary Revision No.	Date of Issue	Date of Insertion	Inserted By	Date of Removal	Removed By

SERVICE BULLETINS

At the time of delivery, applicable service bulletins are normally incorporated. When a subsequent service bulletin is issued after the delivery date, Cessna will notify the airplane owner concerning details for compliance. Since service bulletins are often technical and lengthy, the notice of an applicable bulletin will contain excerpted information that is pertinent for proper compliance.

Service Bulletin No.	Date of Bulletin	Title of Service Bulletin	Incorporated By

Service Bulletin No.	Date of Bulletin	Title of Service Bulletin	Incorporated By

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
A		**Chapter 4**
	04-00-00 pg. 2	04-2 Wing Attachment Step 5 deleted sentence “Bolts may be removed and replaced one at a time in order to avoid complete removal and realignment of the wing.” The use of the word “may” could cause confusion. Corrected the number of bolts from 28 to 30.
	04-00-00 pg. 3	Corrected reference to wing flap actuator. Was 27-24 and is 27-23.
A		**Chapter 5**
	05-10-00 pg. 2	Added step 16 for elevator trim motor inspection (renumbered following inspections).
	05-10-00 pg. 2	Added step 25 for exhaust system pressure test.
	05-10-00 pg. 2	Added step 28 for cleaning of door seal system pump filter
	05-10-00 pg. 2	Added step 29 for a hard landing inspection.
	05-10-00 pg. 2	Added step 30 for a lightning strike inspection.
	05-10-00 pg. 3	Added step 34 for replacement of clock battery (renumbered following replacements).
A		**Chapter 20**
	20-00-00 pg. 1	Updated reference of AC43.13-1B. Added 20-3 (d) section on Torque Adapters
	20-00-00 pg. 4	Updated reference of AC43.13-1B
	20-00-00 pg. 6	Updated reference of AC43.13-1B
	20-70-00 pg. 6	Corrected typo in paragraph 20-23 (c)(2)(c) to say “pot life”.
A		**Chapter 21**
	21-00-00 pg. 1-2	Added additional information on the description and operation of the ECS system. Added figure.
	21-60-00 pg. 1-4	Added additional figures and instructions to 21-12 Temperature and Air Control Unit to clarify the removal and installation of the ECS control panel. Added additional instructions for the check of the ECS control system.
A		**Chapter 28**
	28-20-00 pg. 19	Added location of the K-7 Latching Relay
	28-40-00 pg. 1	Updated reference to fuel level sensors to 28-5 in sections 28-21 and 28-23.
A		**Chapter 32**
	32-10-00 pg. 7	Added description for inspection of gear box bolts and included torque value for bolts.
A		**Chapter 73**
	73-00-00 pg. 1	Corrected Engine Maintenance Manual number to X30634A.
A		**Chapter 74**
	Chapter 74	Updated entire chapter to include 74-2 Ignition Switch. This section contains a description, removal and installation instructions, and troubleshooting procedures.
	74-30-00 pg. 1	Corrected Engine Maintenance Manual number to X30634A.
A		**Chapter 80**
	80 Title Page, pg. 1	Corrected Engine Maintenance Manual number to X30634A.
B		**Part 0 Administrative**
	TOC Page pgs. xi &xii	Revised Table of Maintenance Manual Chapters to match Chapter titles.
B		**All Chapters**
	All Chapters LOEP pg. 1	Revised LOEP.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
B		**Chapter 1**
	01-00-00 pgs. 2 & 3	Revised Maintenance Manual Table of Contents to match Chapter titles.
B		**Chapter 3**
	Ch. 3 Title Page pg. 1,	Added "(LC42-550FG)" to Chapter Title.
B		**Chapter 4**
	Ch. 4 Revisions pg. 1, 40-00-00 pg. 5	Revised Maintenance Manual Log of Revisions page. Added Requirements for Alternators
B		**Chapter 5**
	Ch. 5 TOC pg. 1	Revised Table of Contents.
	05-10-00 pg. 1	Item 10. Changed "X420002-1" to "X42002-1"
	05-10-00, pgs. 2 & 3	Item 17. Changed "Ensure the clutch resistance is still 140 in.-lbs of torque or greater." to "For alternator No. 1 (right) ensure the clutch resistance is still 140 in.-lbs of torque or greater." Item 18. Changed "Service alternator bearings." to "For alternator No. 1 (right) service alternator bearings." Deleted Item 24 and renumbered following Items
	05-20-00 pg. 4	Item 31. Deleted "Replace filter if the wire screens or the cotton gauze mesh is damaged."
	05-20-00, pg. 5	Item 42. Revised "Inspect for security, general condition and signs of oil leakage" to "Inspect for security and general condition.". Removed sentence "Any evidence of alternator malfunction requires removal to conduct drive gear hub slippage inspection." Removed sentence "Check air blast tube for obstructions." Added sentences "Inspect belt for tension, fraying, and dryrot. Inspect pulleys for nicks, scratches, warpage, and alignment."
B		**Chapter 11**
	11-00-00, pg. 3	Added placard "SEAT BOTTOM CAVITY MUST REMAIN CLEAR."
B		**Chapter 12**
	TOC pg. 1	Revised Table of Contents
	12-30-00 pgs. 1 through 5	12-15. Revised paragraph to describe Induction Filter Servicing for new air filter assembly.
B		**Chapter 24**
	Ch. 24 TOC Pg. 1	Revised Table of Contents
	24-00-00 pg. 4 & 5	Moved title of Paragraph 24-2 from page 4 to page 5.
	24-30-00 pgs. 1 through 8	24.3.a Revised description of the No. 2 alternator. Revised and moved Figure 24-4. 24-3.d. Revised to describe removal of belt driven alternator. 24-3.e. Revised to describe installation of belt driven alternator and renumbered following subparagraphs. Added subparagraph 24-3.g. Added pages 7 and 8.
	24-50-00 pg. 1	24-7.b.2. Deleted "with six washers." Added "The standoffs must protrude through the nutplate flush or better." Removed washers from Figure 24-9.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
B		**Chapter 25**
	25-12-00 pgs. 1 & 2	25-20. Added description of drink holder attachment and renumbered following subparagraphs. Added Figure 25-17 Drink Holder and renumbered following figures.
	25-31-00 pg. 1	Revised Figure 25-24 to remove left side forward wing saddle access panel.
	25-60-00 pg. 2	25-45.5. Changed “shown in see” to “as shown in”.
	25-12-00 through 25-90-00,	Revised all references to figures to correspond to new figure numbers.
B		**Chapter 27**
	27-01-00 pg. 1	27-4.d. Changed “smart level” to “digital inclinometer”.
	27-10-00 pg. 1	27-10.b.8. Added “While the aileron is in neutral position”.
	27-10-00 pg. 3	27-11.b.3. Changed “smart level” to “digital inclinometer”.
B		**Chapter 28**
	28-20-00 pg. 3 & 4	Added border and lines to Figure 28-18 and moved figure title from page 4 to page 3.
B		**Chapter 52**
	52-00-00 pgs. 2 through 4,	Replaced Figure 52-1 with a more detailed representation of door hinge location. 52-1.e. Moved Figures 52-2 and 52-3 so that text in subparagraph 52-1.e. is consolidated.
	52-10-00 pgs. 3 through 7	52-4.a. Added subparagraph 52-4.a.6. for reconnection of gas strut and pneumatic air supply. 52-7. Added subparagraph 52-7.8. for reconnection of gas strut and pneumatic air supply.
B		**Chapter 53**
	53-10-00 pg. 3 and 4	Replaced Figure 53-2 with a new figure. Revised subparagraph 53-5.m. to indicate three mounting brackets and two threaded inserts; and deleted brackets for knee bolsters.
	53-30-00 pg. 1 and 2	Revised Figure 53-3 to remove one forward cabin floor panel and indicate A/C evaporator panel. Added Figure 53-4 for A/C evaporator panel. Retitled subparagraph 53-6.e. to “Air conditioning (A/C) System Bay Access Panel.” Added subparagraph 53-6.f. Air conditioning (A/C) Evaporator Access Panel and renumbered following paragraphs. Revised subparagraph 53-6.j. to indicate only one panel.
B		**Chapter 61**
	61-10-00 pg. 1	Added subparagraph 61-4.a.3 for loosening the left alternator adjustment rod and unmounting the drive belt. Renumbered following subparagraphs. Added subparagraphs 61-4.b.4 through 61-4.b.7., inclusive, for adjusting the left alternator belt and renumbered following subparagraph. Revised Figure 61-1.
	61-20-00 pgs. 1 through 5	61-5.a. Revised paragraph to include procedure for removal of left alternator adjustment rod attachment bracket. Deleted subparagraph describing baffling removal and renumbered following subparagraphs. 61-5.b. Revised paragraph to include procedure for installation of left alternator adjustment rod attachment bracket and tensioning of drive belt. Revised Figure 61-2.

NARRATIVE DISCUSSION OF REVISIONS

Revision Level	Page No.	Comment
B		**Chapter 71**
	71-00-00 pgs. 2 through 8	Added subparagraph 71-4.a.6. for loosening the left alternator adjustment rod and unmounting the drive belt and renumbered following subparagraphs. Changed "and hub assembly" to "hub assembly, and alternator drive pulley" in subparagraph 71-4.a.7. 71-4.b.19. Changed "and hub assembly" to "hub assembly, and alternator drive pulley". Replaced Figures 71-1 through 71-7, inclusive. Deleted Figure 71-8 and renumbered following figures.
	71-10-00 pgs. 1 through 3	Revised figure titles. Revised all figure references to indicate correct figure.
	71-20-00 pgs. 1 and 2	Revised figure titles. Revised all figure references to indicate correct figure.
	71-60-00 pgs. 1 and 2	71-7 Revised entire paragraph to describe new air filter assembly. Revised Figure 71-15 Engine Air Filter. Revised all figure references to indicate correct figure.
	71-70-00 pgs. 1 and 2	Revised figure titles. Revised all figure references to indicate correct figure.
	71-80-00 pg. 1	Revised figure title. Revised all figure references to indicate correct figure.
B		**Chapter 76**
	TOC pg. 1	Deleted Paragraph 76-5 and revised page numbers on the Table of Contents.
	76-10-00 pgs. 2 through 6	76-3.a.5. Changed "four panel screws." to "two or four panel screws depending on cover plate installed. " Changed "(see Figure 76-9)" to "(See Figure 76-9, Figure 76-10, or Figure 76-11). 76-3.b.2. Changed "four panel screws." to "two or four panel screws depending on cover plate installed. " Changed "(see Figure 76-9)"to "(see Figure 76-9, Figure 76-10, or Figure 76-11). 76-3.b.4. Revised reference from Figure 76-10 to Figure 76-12. Revised all figure references to indicate correct figure.
	76-20-00 pgs. 3 through 12	Replaced Figure 76-9 Fuel Mixture Control Panel and added Figure 76-10 and Figure 76-11. Renumbered following figures. 76-4.b.2. Changed "install four panel screws" to "install two or four panel screws depending upon type of cover plate installed." and added reference to Figure 76-10 and Figure 76-11. 76-5. Revised paragraph to describe one butterfly valve inside the engine induction tube. Replaced Figure 76-11, Figure 76-12, and Figure 76-13 with new figures. Deleted Figure 76-16. 76-5.a. Revised paragraph and subparagraphs to describe engine induction system with only one butterfly valve. 76-5.b. Revised paragraph and subparagraphs to describe engine induction system with only one butterfly valve. 76-6 Deleted paragraph and subparagraphs in their entirety. Revised all figure references to indicate correct figure.
B		**Chapter 79**
	79-00-00 pg. 1	Changed font on general note and revised page number.
C		**Chapter 4**
	Ch. 4 Revisions pg 1, LOEP pg. 1	Revised Maintenance Manual Log of Revisions Page. Revised LOEP.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
C		**Chapter 24**
	LOEP pg. 1	Revised LOEP
	24-30-00 pgs. 2 through 5.	Revised Figure 24-4 Added subparagraphs 24-3.f., 24-3.g, and 24-3.h describing removal and installation of drive and alternator pulleys. Renumbered following subparagraphs. Moved Figure 24-5.
C		**Chapter 32**
	LOEP pg. 1	Revised LOEP.
	32-20-00 pg. 6	32-9.a.5. Changed “420002” to “42002” 32-9.a.6. Changed “40003” to “42003”.
C		**Chapter 34**
	LOEP pgs. 1 and 2. TOC pg. 1	Revised LOEP. Revised Table of Contents.
	34-50-00 pgs. 8 through 13.	Revised Figure 34-33 Added New Figure 34-34 and renumbered following figures. Revised Figure 34-35. Revised figure references to indicate correct figure.
D		**Chapter 1**
	LOEP pg. 1	Revised LOEP
	01-00-00 pg. 1	01-1.b.2. Changed “contain” to “contained.” 01-1.b.3. Changed “40004” to “42001.” 01-2.a.1. Changed “RA050001” to “RB050000.” 01-2.a.3. Changed “Wiring” to “Electrical.” Changed “RA240000” to “RB24000X.”
D		**Chapter 2**
	LOEP pg. 1	Revised LOEP
	02-00-00 pgs. 2 and 3	02-7. Changed all occurrences of “UPS” to “Garmin.” Deleted “S-Tec 30” from Auto Flight. Changed “S-Tec 55” to “S-Tec 55X.” Added Garmin GTX 330 Transponder. Added Avidyne PFD. Revised Avidyne EX5000 publication number and title. Revised Garmin AT SL70 publication number.
D		**Chapter 3**
	LOEP pg. 1	Updated LOEP
	03-00-00 pg. 3	03-9.a. Changed “gear driven alternator” to “gear driven and belt driven alternator.” Changed ”magnetos and second alternator” to “magnetos.”
	03-00-00 pg. 5	03-14. Changed wing area from 32.2 sq. ft. to 141.2 sq. ft.
D		**Chapter 4**
	Ch. 4 Revisions pg 1, LOEP pg. 1	Revised Maintenance Manual Log of Revisions Page. Revised LOEP.
	04-00-00 pg. 3	For Rudder, changed “Check condition of hardware for cracks, corrosion, and wear” to “Check condition of hardware and cables for cracks, corrosion, and wear.”
	04-00-00 pg. 5	04-3. Added replacement of 3-volt lithium battery in Avidyne MFD. Added requirements for SpeedBrakes.
D		**Chapter 5**
	LOEP pg. 1	Revised LOEP

NARRATIVE DISCUSSION OF REVISIONS

Revision Level	Page No.	Comment
	05-10-00 pg. 1	Added Item 5 “Perform compass swing” and renumbered following items.
	05-10-00 pg. 2	Item 18. Changed “For alternator No. 1 (right)” to “Service alternator bearings.” Item 20. Changed “12 megohms” to “1-20 megohms.” Item 29. Referenced Lightning Strike inspection and repair to Chapter 20.
	05-20-00 pg. 7	Item No. 78. Added inspection of rudder crossover cable. Added removal and inspection of rudder crossover cable at every 1000 hrs.
	05-20-00 pg. 9	Item Nos. 95 and 96. Added requirements for SpeedBrakes and renumbered following items
	05-20-00 pgs. 7 through 15	Text on all pages shifted down due to addition of rudder crossover cable inspection.
	05-20-00 pg. 16	Added page.
	05-22-00 pg. 2	Deleted “, altitude hold computer for system 30 autopilot.”
D		**Chapter 11**
	LOEP pg. 1	Revised LOEP
	TOC pg. 1	Revised Table of Contents
	11-00-00 pgs. 2 through 12	11-4. Interior Placards. Moved entire paragraph after 11-5. Exterior Placards to conform to ATA Specification 100 format. Renumbered pages.
	11-20-00 pg. 2	Added Ground Power Supply placard.
	11-30-00 pg. 1	Revised Near Pilot and Copilot Interior Door Handles placards to indicate “(S/N 42001 to 42005).”
	11-30-00 pg. 2	Added new Near Pilot and Copilot Interior Door Handles placards for S/N 42006 and on.
	11-30-00 pg. 5	Added placard for Autopilot limitations. Added placards for Autopilot Master Switch. Indicated placard embroidered on the Back Lower Portion of the Front Seat Headrests for S/N 42001 to 42005. Replaced placard “On Forward Portion of Front Seat Center Armrest Near Fuel Selector” with placard “Engraved on fuel Selector Knob.”
	11-30-00 pg. 6	Added placard for oxygen distribution manifold Added placard for oxygen fill port. Added placard on oxygen lines located behind cockpit floor and panels.
	11-30-00 pgs. 7 and 8	Added pages due to addition of placards
D		**Chapter 12**
	LOEP pg. 1	Revised LOEP
	12-10-00 pgs. 3 through 6	12-14. Revised to indicate differing conditions due to Garmin G1000 system. Added pages 5 and 6
	12-30-00 pg. 1	12-15.a. Changed “an oil disposable” to “a disposable.” 12-17.a. Changed “visible on the nose strut” to “visible on the nose strut rod.” 12-17.a.1. Revised to delete access panel.
	12-30-00 pg. 6	12-19. Deleted “left” from the second sentence in the paragraph.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
D		**Chapter 20**
	LOEP pgs. 1 and 2	Revised LOEP
	20-00-00 pg. 5	Removed and replaced Figure 20-4.
	20-70-00 pg. 2	Revised Figure 20-33 to show forward wing access panel only on right side of fuselage floor.
	20-70-00 pgs. 5 through 14	Added 20-22.g. Inspection and Repair After Lightning Strike and renumbered following pages.
	20-100-00 pg. 2	Revised footer.
D		**Chapter 21**
	LOEP pg. 1	Revised LOEP
	TOC pg. 1	Revised Table of Contents
	21-20-00 pgs. 3, 4, and 5	21-6. Consolidated removal and installation procedures of left and right eyeball vents. 21-7. Deleted removal and installation procedures of right eyeball vents. Figure 21-3. Revised. Figure 21-4. Deleted. Renumbered following figures. Revised all figure references.
	21-40-00 pgs. 1 and 2	Renumbered Figures 21-6 and 21-7 to Figures 21-5 and 21-6, respectively. Revised all figure references.
	21-60-00 pgs. 1, 2, and 3	Renumbered Figures 21-8, 21-9, and 21-10 to Figures 21-7, 21-8, and 21-9, respectively. Revised all figure references.
D		**Chapter 22**
	LOEP pg. 1	Revised LOEP
	22-00-00 pg. 1	22-1.a. Changed “System 55” to “System 55X.” 22-2. Change “SYSTEM 55” to “SYSTEM 55X” in the title. 22-2.c. Deleted “(S-Tec 55X).”
	22-00-00 pgs. 2 through 10.	Revised S-Tec 55X autopilot information and procedures. Added S-Tec 360 Autopilot Altitude Preselect (Optional Equipment) information and procedures.
	22-10-00 pgs. 1 and 2	Revised Sections 22-12 Instructions for Continued Airworthiness, 22-12, Troubleshooting Information, and 22-14 Special Issues.
	22-11-00 pgs. 1 through 4	Revised Paragraph numbers. Renumbered figures and revised all figure references. 22-15.b. Changed nut and washer designations. 22-16.b. Changed screw, nut and washer designations.
D		**Chapter 23**
	LOEP pg. 1	Revised LOEP
	23-00-00 pg. 1	23-1.a. Changed “UPS” to “Garmin AT” and “UPSAT” to “Garmin AT.”
D		**Chapter 24**
	LOEP pg. 1	Revised LOEP
	24-00-00 pg. 2	24-1.j. Changed “reaker” to “breaker.”
	24-00-00 pg. 4	Figure 24-2. Added 4 diodes and inverted the ground symbol.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
	24-30-00 pg. 3	24-3.e.2. Added sentence "Ensure that a minimum of 2 threads are exposed through the self-locking nut after final installation." 24-3.e.5. Changed "20 to 30 lb.-ft." to "15 to 25 lb.-ft." Changed "30 to 35 lb.-ft." to "15 to 20 lb.-ft."
	24-30-00 pgs. 5 through 8	24-4.b. Added "and Figure 24-8." 24-4.b.1. Changed "Loosen" to "for S/N42001 to 42022 loosen." Added "For S/N 42023 and on remove the four bolts securing the battery box cover to the battery box and remove the cover." 24-4.c.1. Deleted "Battery terminals face forward when properly installed." 24-4.c.2. Added sentence "Torque bolts 36 to 40 in.-lb." 24-4.c.3. Changed : Close" to "For S/N 42001 to 42022 close." Added "For S/N 42023 and on replace the battery box cover and install the four bolts. Hand tighten the bolts until snug. Added "(S/N 42001 to 42022)" to the title of Figure 24-6. Also indicate right side of firewall. Added Figure 24-8 and renumbered following figures
	24-40-00 pgs. 1 and 2.	24-6.b.4 Revised figure reference Revised Figure 24-8 to indicate negative cable may be attached to either bolt of the ground power plug. Renumbered Figure 24-8 to figure 24-9. 24-6.c. Renumbered subparagraphs
	24-50-00 pg. 1	24-7.a.4. Revised figure reference. Renumbered Figure 24-9 to Figure 24-10 and indicated "Adjustment hole cover. Adjust voltage to 14.2 VDC."
D		**Chapter 25**
	LOEP pg. 1	Revised LOEP
	TOC pgs. 1 & 2	Revised Table of Contents
	25-00-00 pgs. 3 & 4	Revised Figure 25-1 Revised Figure 25-2
	25-11-00 pgs. 2, 3 & 4	25-8.a. Renumbered subparagraphs.
	25-12-00 pg. 1	25-20.a.2. Deleted paragraph on cupholder and renumbered following paragraphs. Deleted Figure 25-17 Cupholder.
	25-12-00 pg. 2	25.20.a.6. Changed "Fuel Selector" to Fuel Tank Selector." 25-20.a.8.a. Changed "two flat head 10-32 screws" to "three flat head 10-32 screws." 25-20.a.8.c. Added subparagraph for protection of splined rod connecting fuel tank selector to fuel selector knob.
	25-30-00 pg. 1	25-23.a.1 through 25-23.a.3. Revised paragraphs to describe presence of differing swivel lights. Added (S/N 42001 through 42003) to title of Figure 25-19. 25-23a.2. Changed "10-32" to "6-32." Changed "Figure 25-19" to "Figure 25-19 or Figure 25-20." 25-24.a.5. Changed "Figure 25-19" to "Figure 25-19 or Figure 25-20."
	25-30-00 pgs. 2 through 8	Added Figure 25-20 and renumbered following figures. Revised Figure 25-21. 25-26.a.2. Changed "8-32 screws" to "6-32 screws and washers." 25-27.a.2. Changed "8-32 screws" to "6-32 screws and washers." Revised all figure references.
	25-31-00 pg. 1	25-37.a. Revised to indicate there is only one forward wing access panel which is located under the right side front seat pan. Added A/C Evaporator Access Panel to Figure 25-24.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
	25-31-00 pg. 2	Added paragraph 25-42 A/C Evaporator Access Panel.
D		**Chapter 27**
	LOEP pgs. 1 and 2	Revised LOEP
	TOC pg. 1	Revised Table of Contents
	27-10-00 pg. 2	27-10.c. Compressed word spacing on "remove the push-pull tubes for inspection."
	27-10-10 pg. 4	Moved Figure 27-15 to better fit on the page.
	27-20-00 pgs. 1 through 4	27-15.b. Revised procedure for rudder cable installation and adjustment. 27-15.d. Revised procedure for rudder cable removal.
	27-30-00 pgs. 6 and 7	Revised subparagraph numbers to be consecutive.
	27-60-00 pgs. 1 and 2	Added speed brake provisions.
D		**Chapter 28**
	LOEP pgs. 1 and 2	Revised LOEP.
	28-20-00 pg. 1	Renumbered first subparagraph from "b" to "a."
	28-20-00 pg. 2	Corrected Latest Revision Date.
	28-20-00 pg. 3	Changed "RADUIS" to "RADIUS" in Figure 28-18.
	28-20-00 pg. 7	Renumbered first subparagraph from "c" to "b."
	28-20-00 pg. 9	28-15.a.1. Changed "pilot's" to "co-pilot's." 28-15.a.5. Changed "four self-locking nuts" to "four countersunk 6-32 screws, self-locking nuts." Changed "fuel selector panel" to "fuel selector plate."
	28-20-00 pg. 10	28-15.b.1. Changed "Wiring Manual" to "Electrical Manual." 28-15.b.8. Changed "four self-locking nuts and washers on the studs mounted" to "four countersunk 6-32 screws, self-locking nuts, and washers." Changed "Torque Nuts 10 in.-lbs." to "Torque to 10 in.-lbs."
	28-20-00 pg. 12	Revised Figure 28-30.
	28-20-00 pgs. 13 and 14	Added subparagraph 28-16.b.2. "Hand tighten the flared tube fittings connecting the fuel lines to the fuel pump until the assembly is complete and then tighten 150-250 in.-lbs. Added subparagraph 28-16.b.3. "Prior to tightening the hose clamps around the fuel pump ensure that the fuel lines are not pre-stressed. Tighten hose clamps 10-15 in.-lbs." Revised subparagraph "28-15.b. General Leak Check for Each Line, Tube, or Fitting" to Subparagraph "28-15.c. General Leak Check for Each Line, Tube, or Fitting" and renumbered following subparagraphs. Replaced photograph of Primer Switch in Figure 28-31.
	28-20-00 pg. 15	Reformatted paragraph under Figure 28-32.
	28-20-00 pg. 16	Revised Figure 28-33.

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Revision Level	**Page No.**	**Comment**
	28-20-00 pg. 22	Revised Figure 28-37 title.
	28-20-00 pgs. 6 through 20.	Revised page numbers.
D		**Chapter 32**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Changed Par. No. 32-1 Main Landing Gear – General to Par. No. 32-2. Revised page numbers.
	32-10-00 pgs. 6, 7, and 8	Revised page numbers.
	32-20-00 pgs. 2 and 3	32-7.a.7 and 32-7.a.8 Changed "AN526C1032R8" to "AN526C1032 or MS51958." 32-7.a.10 Added sentence "Hand tighten the MS21207 screws holding the internal rib to the nose strut fairing snug." And removed MS21207 from sentence indicating torque of 30 to 36 in.-lbs. Changed "AN526C1032" to "AN526C1032 or MS51958." Figure 32-8 Revised screw type. 32-8.a.3. Deleted safety wire, changed bolts and washers to screws, and added blocks. 32-8.b.5. Changed "plates" to "Blocks."
	32-20-00 pg. 4	Figure 32-9. Deleted nut on inside left of the fork.
	32-20-00 pg. 5	32-8.b.8. Revised sentence.
	32-20-00 pg. 6	32-9.a.5. and 32-9.a.6. Changed "castellated" to "self-locking."
	32-41-00 pg. 1	32-14.a.4. Changed "slide the brake" to "slide the brake and spacer". Figure 32-15. Revised the figure to show the spacer.
D		**Chapter 33**
	LOEP pg. 1	Revised LOEP
	TOC pg. 1	Revised Table of Contents
	33-00-00 pgs. 1 through 4	33-1.c. Indicated differing lighting conditions between S/N 42001 to 42003 and S/N 42004 and on. Figure 33 -1 Revised to indicate differing lighting conditions between S/N 42001 to 42003 and S/N 42004 and on.
	33-10-00 pg. 1	33-4.d.1. and 33-4.e.3. Added reference to Figure 33-4.
	33-10-00 pg. 2	33-5.a. Indicated differing lighting conditions between S/N 42001 to 42003 and S/N 42004 and on. Added "reading lights" to definition of overhead swivel lights. Added reference to Figure 33-4. Figure 33-3. Added "(S/N 42001 to 42003)" to title
	33-10-00 pgs. 3 and 4.	Added Figure 33-4 and renumbered following figures. 33-5.b. Indicated differing removal procedures between S/N 42001 to 42003 and S/N 42004 and on. 33-5.c. Indicated differing installation procedures between S/N 42001 to 42003 and S/N 42004 and on. 33-5.d. Indicated differing troubleshooting procedures between S/N 42001 to 42003 and S/N 42004 and on.
	33-10-00 pgs. 3 through 10.	Revised figure titles. Updated figure references. Deleted 33-7.e. and Figure 33-7 "Compass Light." Added 33-10-00 pages 9 and 10.

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Revision Level	Page No.	Comment
	33-40-00 pgs. 1, 3 through 6.	Revised figure titles. Updated figure references.
D		**Chapter 34**
	All Pages	Revised entire chapter to include Avidyne Entegra Primary Flight Display and Multi-Function Flight Display. Deleted magnetic compass for S/N 42001, deleted Figure 34-9, and renumbered following figures and cross references.
D		**Chapter 53**
	LOEP pg. 1	Revised LOEP
	53-10-00 pg. 3	Figure 53-2. Indicated A/C system bay rib on A/C ready aircraft only.
	53-60-00 pg. 1	53-8.c. Removed "replaceable" from the last sentence. Added sentence "Replacement of the sleeves may only be performed with replacement of the cables."
D		**Chapter 56**
	LOEP pg. 1	Revised LOEP
	TOC pg. 1	Revised Table of Contents
	56-10-00 pg. 1	56-3.a. Added "See Figure 56-2" after the first sentence. Deleted heat-chisel method and revised entire paragraph. 56-3.b.5. Added "For S/N 42020 and on, there is glue on the beveled edge that must be removed (see Figure 56-2)." Changed "polyethylene" to "polyurethane glue between the windshield or window and the fuselage flange." Changed "original glue" to "original glue on the fuselage flange."
	56-10-00 pg. 2	56-4.a.1. Changed "flange" to "flange and bevel the edge as shown in Figure 56-3." Added Figure 56-2 and renumbered following figures.
	56-10-00 pg. 3	56-4.a.10. Added sentence "It is not necessary for the beveled surface to pass the water-break test." 56-5.a.1. Changed "flange" to "flange and bevel the edge as shown in Figure 56-3." 56-5.a.11. Added sentence "It is not necessary for the beveled surface to pass the water-break test.
	56-10-00 pg. 4	Added Figure 56-3 and renumbered following figures.
	56-10-00 pg. 5	56-7.a.5. Change "place, excess…" to "place, ensure that adhesive fills the bevel to provide a smooth transition between fuselage and window. Excess…" 56-8.a.5. Added reference to Figure 56-2. 56-8.a.6. Change "place, excess…" to "place, ensure that adhesive fills the bevel to provide a smooth transition between fuselage and windshield. Excess…"
	56-10-00 pg. 6	Revised Figure 56-3 and renumbered to Figure 56-5. 56-9.a. Deleted "Due to the out-gassing of the curing adhesive used to bond the windows in place," and changed "cracked paint" to "cracked paint and filler."
	56-10-00 pgs. 1 through 8	Revised all figure references.
D		**Chapter 57**
	LOEP pg. 1	Revised LOEP
	57-00-00 pg. 2	Revised page footer.
	57-10-00 pgs. 1 through 10	Revised footer and page numbers. Deleted pages left intentionally blank. 57-3.m. Changed "30" to "28."

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Revision Level	Page No.	Comment
D		**Chapter 61**
	LOEP pg. 1	Revised LOEP
	61-10-00 pg. 1	61-4.a.2. Added subparagraph 2 "Disconnect the battery wires attached to the left alternator." Renumbered following subparagraphs.
	61-10-00 pg. 2	61-4.b.2. Changed "60 to 70 ft.-lbs." to "70 to 80 ft.-lbs." 61-4.b.5. Changed "20 lb.-ft." to "15 to 25 lb.-ft." Changed "After final belt adjustment has been made torque jam nut on both ends of adjustment rod and nut attaching the adjustment rod to the alternator 30 to 35 lb.-ft." to "After final belt adjustment has been made torque jam nut on both ends of adjustment rod, and nuts attaching the adjustment rod to the alternator and to the bracket attached to the engine 15 to 20 lb.-ft." 61-4.b.6. Added warning statement "Ensure that there is a minimum clearance of ½" between the upper cowling skin and the alternator case, fan, belt, and pulley." 61-4.b.7. Added subparagraph 7. "Connect the battery wires to the alternator." And renumbered following subparagraphs.
	61-20-00 pgs. 1 and 2	61-5.a.2. Added subparagraph 2 "Disconnect the battery wires attached to the left alternator." Renumbered following paragraphs. 61-5.b.6. Changed "20 lb.-ft." to "15 to 25 lb.-ft." Changed "After final belt adjustment has been made torque jam nut on both ends of adjustment rod and nut attaching the adjustment rod to the alternator 30 to 35 lb.-ft." to "After final belt adjustment has been made torque jam nut on both ends of adjustment rod, and nuts attaching the adjustment rod to the alternator and to the bracket attached to the engine 15 to 20 lb.-ft." 61-5.b.7. Added warning statement "Ensure that there is a minimum clearance of ½" between the upper cowling skin and the alternator case, fan, belt, and pulley." 61-5.b.9. Added subparagraph 9. "Connect the battery wires to the alternator." Renumbered following subparagraphs.
	61-20-00 pg. 5	61-7.a. In Solution No. 4 deleted sentence "This Page Intentionally Left Blank."
D		**Chapter 71**
	LOEP pg. 1	Revised LOEP
	71-00-00 pg. 9	Revised Figure 71-11.
	71-60-00 pg. 1	71-7.7. Added "and discard." to the end of the sentence. 71-7.9. Changed "Reinstall the filter" to "Install the new filter element".
D		**Chapter 76**
	LOEP pg. 1	Revised LOEP
	76-20-00 pg. 1	Revised full rich spring in Figure 76-5.
	76-20-00 pg. 5	76-4.b.8. Added procedure.
	76-20-00 pg. 9	Replaced Figure 76-14 Induction Heat Control Cable Connection to Valve.
E		**Part 0 Administrative**
	Pgs. ix through xxviii	Revised the Narrative Discussion of Revisions and renumbered subsequent pages.
E		**Chapter 4**
	Revisions Pg. 1	Revised Log of Revisions
	LOEP pg. 1	Revised LOEP.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
	04-00-00 pg. 5	For the System Battery, revised location of the battery. For the No. 2 Alternator, changed "20 to 30" to "13 to 17."
	04-10-00 pg. 1	Deleted "and will be completed by the end of 2003."
E		**Chapter 5**
	LOEP pg. 1	Revised LOEP.
	05-10-00 pgs. 1 and 3	Item 7. Added "or biennially." Item 34. Indicated no clock battery replacement if Avidyne FlightMax Primary Flight Display installed in the aircraft.
	05-20-00 pgs. 8 through 13	Item 88. Added "if the rods show any sign of corrosion, replace with new rods." Other text shifted on pages.
E		**Chapter 11**
	LOEP pg. 1	Revised LOEP.
	11-30-00 pg. 7	Revised Oxygen Distribution Manifold placard.
E		**Chapter 20**
	LOEP pgs. 1 and 2	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	20-70-00 pgs. 6 through 16	Removed and replaced Checkout Description table.
	20-90-00 pgs. 2, 4, and 6	Revised page numbers.
E		**Chapter 21**
	LOEP pg. 1	Revised LOEP.
	21-60-00 pgs. 3 and 4	Add "(S/N 42003 to 42023)" to Figure 21-9. Added Figure 21-10. Renumbered pages
E		**Chapter 22**
	LOEP pg. 1	Revised LOEP.
	22-11-00 pgs. 1 and 2	22-15.b. Changed "NAS1149F0363P" to "NAS1149F0332P" 22-15.b.3. changed "NAS1149F036P" to "NAS1149F0332P" in two places.
	22-11-00 pgs. 2 and 3	22-16.b. Changed "NAS1149F0363P" to "NAS1149F0332P" 22-16.b.3. changed "MS24694-C50" to "MS24694-S50 " and "NAS1149F036P" to "NAS1149F0363P"
E		**Chapter 24**
	LOEP pg. 1	Revised LOEP.
	24-00-00 pg. 6	Revised page number.
	24-30-00 pg. 3	24-3.e.5. Changed "15 to 25 lbs.-ft." to "13 to 17 lbs.-ft." 24-3.g.2. Changed "30 to 35 lbs.-ft." to "40 to 45 lbs.-ft."
E		**Chapter 25**
	LOEP pgs. 1 and 2	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
	All	Renumbered paragraphs and updated figure and paragraph cross references.
	25-00-00 pg. 2	25-1.3. Changed “Figure 25-2 ” to “Figure 25-2 and Figure 25-3.”
	25-00-00 pgs. 4 through 6	Added Figure 25-3 and renumbered following figures. Added page 6.
	25-10-00 pgs. 1 and 3	Renumbered figure and revised all cross references.
	25-11-00 pgs. 1 through 11	25-5.a. Changed “Figure 25-2” to “Figure 25-2 and Figure 25-3.” 25-9.a. Added provision for removal and installation of the flight instrument panel with the Entegra Option. Revised Figure 25-11
	25-12-00 pgs. 1 through 4	Revised Figure 25-17. 25-20.a.2. Added provision for Drink Holder and renumbered following subparagraphs. Added Figure 25-18 and renumbered following figures.
	25-13-00 pgs. 1 and 2	Renumbered figure and revised all cross references.
	25-30-00 pgs. 1 through 6	Revised title of Figure 25-20 and Figure 25-21. Added Figure 25-22 and renumbered following figures. Revised all figure cross references.
	25-31-00 pgs. 1 and 2	Renumbered figure and revised all cross references.
	25-60-00 pg. 2	Renumbered figure and revised all cross references.
	25-90-00 pg. 1	Renumbered figure and revised all cross references.
E		**Chapter 27**
	LOEP pg. 1	Revised LOEP.
	27-30-00 pg. 2	Figure 27-19. Revised text.
	27-60-00 pgs. 1 through 4	27-25. Added new figure cross reference and revised removal and installation procedure concerning screws and bottom mount bar. Added new Figure 27-33. 27-25.e. Added paragraph for servicing and maintenance.
E		**Chapter 28**
	LOEP pgs. 1 and 2	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	28-20-00 pgs. 1 through 26	28-14.a.5. Added “, except the torque on -6 aluminum tube must be 75 to 125 in.-lbs. and on -8 aluminum tube must be 150 to 250 in.-lbs.” to the end of the first sentence. 28-14.c.2. Revised torque values. Added “(S/N 42003 to 42040)” to title of Figure 28-16. Added Figure 28-17 and renumbered following figures and page numbers. Removed and replaced Figure 28-26 and Figure 28-27. Removed and replaced Figure 28-30. 28-15.b.7 Added alignment instructions. 28-15.b.8. Changed “10 in.-lbs.” to “4 to 6 in-lbs.” 28-15.b.10. Changed “pilot’s” to “co-pilot’s”. Revised all figure cross references. Added pages 25 and 26.
E		**Chapter 31**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.

NARRATIVE DISCUSSION OF REVISIONS

Revision Level	Page No.	Comment
	31-00-00 pg. 1	31-1. Indicated the Voltmeter/OAT/Clock will not be installed on the aircraft if the Avidyne Entegra Primary Flight Display (PFD) is installed on the aircraft.
E		**Chapter 32**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	32-10-00 pgs. 2 through 7	32-3.a.14. Changed "50 to 58 in.-lbs. to "23 ft.-lbs." Revised Figures 32-4, 32-5, and 32-6. Added Figures 32-7 and 32-8. 32-4.b. Revised Wheel Toe-in and Camber procedures.
E		**Chapter 33**
	LOEP pgs. 1 and 2	Revised LOEP. Revised footer on pg. 2.
	TOC pg. 2	Revised footer.
	33-00-00 pg. 1	33-1.c. Changed "S/N 42001 to 42023" to "S/N 42001 to 42003 and "42024 and on" to "42004 and on."
E		**Chapter 34**
	LOEP pgs. 1 through 4	Revised LOEP.
	TOC pgs. 1 and 2	Revised Table of Contents
	34-50-00 pgs. 7 and 9	34-18. Indicated the KG 102A remote gyro and KI525A HIS indicator will not be installed in the aircraft if the Avidyne Entegra Primary Flight Display (PFD) is installed on the aircraft. 34-19.a. Added "or Figure 34-42." 34-19.a.2. Added "for S/N 42003 to 42023. Remove the 6-32 flat head screws for S/N 42024 and on."
	34-50-00 pgs. 11 through 16	Retitled Figure 34-41, added Figure 34-42, and renumbered following figures.
	34-60-00 pgs. 12 through 14	Added cross reference to Figure 34-51 in 34-23.a.
	34-60-00 pgs. 20 through 34	Added Figure 34-51. 34-23.f. Deleted "Primary" from title. 34-24.a.3. Revised location description of Magnetometer/OAT. 34-26. Deleted "AND MFD" from title. 34-27. Added provision for removal and installation of MFD power converter. Revised Figure 34-53. Added Figure 34-54. Renumbered following paragraphs, figures, and pages.
E		**Chapter 51**
	Title Page pg. 2	Revised Footer.
	LOEP pg. 1	Revised LOEP.
	51-70-00 pg. 9	Revised Figure 51 – 3.
E		**Chapter 53**
	LOEP pg. 1	Revised LOEP.
	53-10-00 pg. 3	Removed and replaced Figure 53-2.
	53-30-00 pg. 1	Removed and replaced Figure 53-3.

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Revision Level	Page No.	Comment
E		**Chapter 56**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	56-10-00 pgs. 1 through 10	56-3.b.5. "For S/N 42020 and on" changed to "For 42030 to 42050." 56-4.a.1. Deleted "and bevel the edge as shown in figure 56-3." Revised Figure 56-2. 56-4.a.10. Deleted "It is not necessary for the beveled surface to pass the waterbreak test." 56-5.a.1. Changed "Bevel the edge as shown in Figure 56-3" to "For S/N 42001 to 42049 do not bevel the edge. For S/N 42050 and on bevel the edge as shown in Figure 56-3." Revised Figure 56-3. Revised Figure 56-5.
E		**Chapter 61**
	LOEP pg. 1	Revised LOEP.
	61-20-00 pg. 2	61-6.b. Revised propeller governor adjustment procedures.
E		**Chapter 71**
	LOEP pgs. 1 and 2	Revised LOEP.
	71-00-00 pgs. 2 through 12	Removed and replaced Figure 71-1 through Figure 71-7 Added Figure 71-8 through 71-11. Revised page numbers.
E		**Chapter 76**
	LOEP pg. 1	Revised LOEP.
	76-10-00 pgs. 2 and 3	76-3.a.3. Reformatted paragraph and renumbered following subparagraphs. Renumbered Figure cross references as required.
	76-20-00 pgs. 1 through 12	76-4. Added S/N effectivity. Changed title of Figure 76-5. Added Figure 76-6 and renumbered following figures. Revised Figure 76-7. 76-4.a. Added ", full rich spring (S/N 42003 to 42044)." Added 76-4.a.2. Changed "spacer" to "spacers" and "Figure 76-6" to "Figure 76-8." Removed and replaced Figure 76-7 Fuel Mixture Control Assembly. 76-4.b.8. Removed and replaced procedure. 76-5.b.4. Added "Hand tighten cable stand-off nut snug, do not over tighten.**"**
E		**Chapter 77**
	LOEP pg. 1	Revised LOEP
	TOC pg. 1	Revised Table of Contents
	77-00-00 pg. 1	77-1.c. Added subparagraph for MFD.
	77-20-00 pgs. 1 through 6	77-5. Revised paragraph to account for EMAX considerations. 77-6. Revised paragraph to account for EMAX considerations. 77-11. Deleted Digital Engine Scanner paragraph and subparagraphs in their entirety. Added figures for CHT Adapter Probe and EGT Probe and renumbered following figures.
E		**Chapter 78**
	LOEP pg. 1	Revised LOEP.
	78-10-00 pgs. 1 and 2	78-3. Revised text formatting and renumbered following paragraphs. 78-3.4. Revised text formatting and renumbered following paragraphs.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
F		**Part 0 Administrative**
	Title Page Pgs. i, ix through xxxiv	Revised Title Page. Revised the Narrative Discussion of Revisions
F		**Chapter 3**
	LOEP pg. 1	Revised LOEP.
	03-00-00 pg. 5	Figure 3 – 1. Deleted asterix from 35.8 Ft. dimension.
F		**Chapter 4**
	Revisions Pg. 1	Revised Log of Revisions
	LOEP pg. 1	Revised LOEP.
	04-00-00 pg. 3	For Bonding of the Top and Bottom Wings, Bonding of Wing Skin to Ribs and Spar, Wing Flaps, Ailerons, and Rudder changed “Section ”20-26” to “Section 20-27.” For Wing Flap Actuator changed “Section 27-23” to Section “27-22.” For Elevator Trim Tab changed “Section 27-19, 27-20, and 27-21” to “Section 27-19 and 27-20.”
	04-00-00 pg. 5	Revised belt replacement of the No. 2 (Left) Alternator. Added replacement and calibration of the CO Detector.
F		**Chapter 5**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	05-10-00 pgs. 1 through 3	Deleted Item 7. Deleted Item 19. Changed Item 27 to Item 26 and changed “Check distribution bus” to “Inspect power grid.” Added Items 28 through 32 for the Oxygen system. Deleted Item 35. Repaginated and Renumbered items.
	05-20-00 pgs. 5, through 13	Deleted Item 46. Deleted two sentences from Item 76. Added inspection for corrosion and replacement of drive rods for Item 81. Added “See Figure 27-1” to Item 88. Changed “Locktite” to “Loctite” in Item 94. Added cleaning of strainer to Item 102. Added Item 127 for the Oxygen system. Added Items 160 through 165 for the Automatic Climate Control System (ACCS). Repaginated and renumbered items.
F		**Chapter 11**
	LOEP pg. 1	Revised LOEP.
	11-30-00 pgs. 6 and 7	Revised fuel selector knob placard. Deleted placard on oxygen lines
F		**Chapter 12**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents
	12-10-00 pgs. 3 and 4	Added paragraph 12-14 Servicing of Oxygen System and renumbered following paragraphs.
	12-30-00 pgs. 1 through 8	12-18. Replaced nose strut servicing paragraphs. Renumbered paragraph numbers. Added pages 7 and 8.

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Revision Level	**Page No.**	**Comment**
F		**Chapter 20**
	LOEP pgs. 1 and 2.	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	20-70-00 pgs. 5 through 12.	Renumbered paragraph for inspection and repair after lightning strike.
	20-80-00 pg. 1.	Renumbered paragraph 20-24 to 20-25.
	20-90-00 pg. 1.	Renumbered paragraph 20-25 to 20-26.
	20-100-00 pg. 1.	Renumbered paragraph 20-26 to 20-27.
F		**Chapter 21**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	21-40-00 pgs. 1 and 2.	21-11.a.4. Changed “connecting linkage” to “plunger seal.” 21-11.b.5. Added adjustment of servo motor. Revised Figure 21-5.
	21-100-00 pgs. 1 through 32.	Added Automatic Climate Control System (ACCS) Section.
F		**Chapter 23**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	23-40-00 pgs. 7 and 8	Replaced Figure 23-8. 23-9.c.2. Changed “Figure 23-8 Headphone/Microphone Jack Installation” to “Figure 23-8.” 23-9.d, 23-9.e, and 23-9.f. Added removal and installation of Bose jack cover. 23-10. Added paragraph “Stereo Audio Input Jack, Optional” and renumbered following paragraphs. Added Figure 23-9 and renumbered following figures.
	23-60-00 pg. 1	Renumbered paragraph and figure.
F		**Chapter 24**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	24-00-00 pgs. 1 through 4	24-1 Revised paragraphs e, f, g, h, i, and j. Revised Figures 24-1 and Figure 24-2.
	24-30-00 pgs. 1 through 12	Revised and reformatted paragraph 24-3 for new No. 2 alternator design. Added and rearranged figures. Revised cross references. Added pages 9 through 12.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
	24-40-00 pgs. 1 and 2	24-6.b. Revised paragraph title. Revised Figure Nos. and cross references.
	24-50-00 pgs. 1 and 2	Revised Figure 24-11. Added paragraph 24-9 Power Grid. Revised Figure Nos. and cross references.
F		**Chapter 25**
	LOEP pgs. 1 and 2	Revised LOEP.
	TOC pgs. 1 and 2	Revised Table of Contents.
	25-11-00 pgs. 5 through 12.	25-10.a. Indicated two self-locking nuts and washer may be nuts with integral toothed washer. Revised Figure 25-11 25-12.a. Revised cross reference. 25-13., 25-14., 25-15., and 25-17. Indicated nuts and washers or nuts with integral toothed washer. Revised Figure 25-13. 25-19.a. Revised removal and installation of Acknowledgment switch or Acknowledgement/Traffic switch. Added paragraph 25-20 Carbon Monoxide Detector. Deleted Figure 25-14 Acknowledge Switch Installation. Added Figure 25-16 Carbon Monoxide Detector. Renumbered following paragraphs and figures. Revised cross references.
	25-12-00 pgs. 1 and 2	25-21.a.1. Added “An optional stereo audio input jack may be installed within the pencil box and must be disconnected before removal of the armrest or the console.” 25-21.a.9. Revised paragraph. Renumbered paragraphs
	25-13-00 pg. 1	Renumbered paragraphs. Revised cross references.
	25-20-00 pg. 1	Renumbered paragraphs. Revised cross references.
	25-30-00 pgs. 1 through 10	25-24.a.2. Added cross reference to Figure 25-24. 25-24.a.3. Added disconnection of flexible oxygen hose. 25-25.a.5. Added cross reference to Figure 25-24. Revised title of Figure 25-22 FCOLP &ACOLP (S/N 42045 and on), Added Figure 25-23 and Figure 25-24 and renumbered following figures. Renumbered paragraphs. Revised cross references.
	25-31-00 pgs. 1 and 2	Renumbered paragraphs. Revised cross references.
	25-60-00 pgs. 1 and 2	Renumbered paragraphs. Revised cross references.
	25-80-00 pg. 1	Renumbered paragraphs. Revised cross references.
	25-90-00 pg. 1	Renumbered paragraph. Revised cross references.
F		**Chapter 27**
	LOEP pgs. 1 and 2	Revised LOEP.
	27-00-00 pg. 3	27-2.d. Revised paragraph to indicate two green LED’s.
	27-10-00 pg. 1	27-10.b.7. Revised gap tolerance between aileron and flaps.

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Revision Level	**Page No.**	**Comment**
	27-20-00 pgs. 2 through 6	27-15.b. Revised installation of rudder cable paragraphs. Added Figure 27-18 and Figure 27-19. Renumbered following figures and cross references. Added pages 5 and 6.
	27-30-00 pgs. 1 through 3, 5, and 6	Renumbered figures and cross references.
	27-50-00 pgs. 1 through 7	27-22.b.4. Indicated use of SS2-32 or AN960-416 washers. 27-22.b.5. Indicated use of SS2-32 or AN960-416 washers. 27-22.b.8. Revised gap tolerance between aileron and flaps. Renumbered figures and cross references.
	27-60-00 pgs. 1 through 3	27-24. Added note indicating that cartridges and asymmetric logic control unit must be returned to the manufacturer for repair or replacement and the cartridges must be returned as a pair. 27-25.a.3. Changed “flat head screws” to “screws.” 27-25.b.4. Changed “flat head screws” to “screws.” Renumbered figures and cross references.
F		**Chapter 28**
	LOEP pgs. 1 and 2	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	28-10-00 pgs. 1 through 10	28-4.c. Revised paragraph to show two different hose clamping conditions. Added Figure 28-3 and renumbered following figures and cross references. Added pages 9 and 10.
	28-11-00 pgs. 2 through 4	Renumbered figure nos. and cross references.
	28-20-00 pgs. 1 through 6	Renumbered figure nos. and cross references. 28-14.4. Changed the word “Scotchbright” to “Scotchbrite.”
	28-20-00 pgs. 8 through 17	Renumbered figure nos. and cross references. Revised Figure 28-33. Added boost pump pressure switch description.
	28-20-00 pgs. 19 through 24	28-18.c. Revised Boost Pump Pressure Switch Removal and Installation. Deleted Figure 28-36 and renumbered following figures. Deleted pages 28-20-00 pages 25 and 26. Revised figure nos. and cross references.
	28-40-00 pgs. 1 and 2	28-20. revised paragraph. 28-24. Deleted paragraph on digital fuel management system.
F		**Chapter 32**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	32-20-00 pg. 1	32-6.a. Revised paragraph.

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Revision Level	**Page No.**	**Comment**
	32-20-00 pgs. 5 through 10	32-8.b.9. Added instruction for attaching nose wheel assembly to base of the nose strut. Renumbered following paragraph. 32-9. Replaced paragraph. 32-10. Replaced all occurrences of “oil” with “oil leakage.”
F		**Chapter 33**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	33-40-00 pgs. 3 through 8	33-12.a. Added “See Figure 33-10 or Figure 33-11.” Revised first Caution note. Revised title of Figure 33-10. Added Figure 33-11 and Figure 33-12. 33-12.d. Added paragraph for Xenon landing light ballast. Revised Figure 33-11 Light Adjustment Reference Card. Revised Figure 33-13 Light Adjustment 33-13. Revised Caution note. 33-13.b. Added 29 in. dimension for xenon light. 33-13.f.2. Added 29 in. dimension for xenon light. Added pages 7 and 8. Revised figure nos. and cross references.
F		**Chapter 34**
	LOEP pgs. 1 through 3	Revised LOEP.
	TOC pgs. 1 and 2	Revised Table of Contents.
	34-00-00 pg. 5	34-1.5.m. Changed “BFGoodrich Model 1100 Attitude Indicator” to “Attitude Indicator”.
	34-10-00 pg. 1	34-2. Added indication of pitot heat thermal device. Revised Figure 34-6.
	34-10-00 pg. 8	Revised Figure 34-13.
	34-30-00 pg. 2	Revised Figure 34-23.
	34-30-00 pgs. 4 through 6	34-9.b. Revised marker beacon antenna tuning adjustment procedures.
	34-50-00 pg. 7	34-18. Changed “the the” to “the”.
	34-60-00 pgs. 25 through 28	34-26.a. Added removal of XM weather receiver. 34-27. Revised paragraph. Removed and replaced Figure 34-54. 34-28.c.3. Indicated removal of screws. 34-28.d.1. Indicated screws must go through the gasket. Revised Figure 34-56.
	34-60-00 pgs. 34 through 44	34-30. Added ORBCOMM antenna removal and installation. 34-31. Added Ryan TCAD processor removal and installation. 34-32. Added Ryan TCAD antenna removal and installation. 34-33. Added XM Receiver removal and installation. 34-34. Added XM antenna removal and installation. Added Figure 34-58 through Figure 34-65. Added pages 35 through 44.

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Revision Level	**Page No.**	**Comment**
F		**Chapter 35**
	All pages	Replaced Chapter
F		**Chapter 51**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	51-80-00 pgs. 1 through 6.	Added paragraphs for expanded metal mesh repair.
F		**Chapter 53**
	LOEP pg. 1	Revised LOEP.
	53-00-00 pg. 1	Changed “Material Specification Data Sheet” to “Material Safety Data Sheet” in the Warning note.
	53-10-00 pgs. 3 and 4	Revised Figure 53-2. 53-5.n. Added oxygen bracket paragraph.
	53-30-00 pgs. 1 and 2	Revised Figure 53-3. Deleted Figure 53-4. 53-6.e. Revised paragraph. 53-6.f. Revised paragraph. 53-6.j. Revised paragraph. Repaginated page.
	53-60-00 pg. 1	Renumbered Figure and revised cross reference.
	53-70-00 pgs. 1 and 2	53-9.e Repaginated paragraph. 53-9.f.1 and 53-9.f.2 Revised paragraphs for removal and installation of steps secured with nutplates.
F		**Chapter 57**
	LOEP pg. 1	Revised LOEP.
	57-00-00 pg. 1	57-1. In the last sentence of the second paragraph indicated that forward rib at WS 151.0 will not be present on aircraft in which the wing was constructed with an integrated cuff.
	57-10-00 pg. 6	57-6.a.3. Indicated forward rib at WS 151.0 will not be present in all aircraft.
F		**Chapter 61**
	LOEP pg. 1	Revised LOEP.
	61-20-00 pgs. 1 and 2.	61-5.b.4. Revised torque values and added anti-seize lubricant. Repaginated following page.
F		**Chapter 71**
	LOEP pg. 1	Revised LOEP.
	71-10-00 pg. 1	71-5.a.2. Changed “fusealge” to “fuselage.”

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Revision Level	**Page No.**	**Comment**
	71-10-00 pg. 3	71-5.b.6. Indicated that pin holes not present on all aircraft. Revised Figure 71-16.
	71-60-00 pg. 1	71-7.10. Added "Ensure that clearance between the face of the filter frame and the alternator is at least 3/4 inch."
F		**Chapter 77**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	77-10-00 pgs. 1 through 4	77-3. and 77-4. Revised paragraphs to account for differing combinations of manifold pressure/fuel flow gauges depending on avionics equipment installed in the aircraft. Added paragraph 77-5 Fuel Flow Transducer and renumbered following paragraphs. Added Figure 77-1 and renumbered following figures. Added pages 3 and 4 to 77-10-00.
	77-20-00 pgs. 1 through 6	77-6. Revised CHT paragraphs to reflect possibility of two different styles of probes and adaptor probes at cylinder head #2. Deleted paragraph 77-10 Engine Boost Pump Pressure Switch. Deleted Figure 77-4 Engine Fuel Pressure Switch Assembly.
G		**All Pages**
	All	Revised header and footer of all pages.
G		**All Chapters**
	LOEP	Revised LOEP.
		Changed all occurrences of "Lancair" to "Columbia Aircraft Manufacturing Corporation". Miscellaneous paragraph numbering revisions for consistency.
G		**Part 0 Administrative**
	Title Page, Pgs. i, xxxi through xliv	Revised Title Page. Revised the Narrative Discussion of Revisions. Added Ice and Rain Protection Ch. 30 to the Maintenance Manual Table of Contents.
G		**Chapter 1**
	TOC pg. 1	Revised Table of Content.
	01-00-00 pgs. 1 through 6	01-2. Added Item 7, Garmin G1000 System Maintenance Manual and Item 8, Required Equipment List Garmin G1000. 01-6. Added Ice and Rain Protection, Ch. 30. 01-10.b. Changed " i through xxxiii" to " with roman numerals". Repaginated pages.
G		**Chapter 2**
	02-00-00 pgs. 1 through 4	02-1. Deleted "on page xxxiii" from the 1st sentence. Deleted the 3rd sentence. Deleted "specific" from the 4th sentence. 02-7. Deleted Lancair part nos. from list. Added Garmin G1000 document nos.
G		**Chapter 3**
	03-00-00 pg. 1	03-2.a. Changed "strength" to "strength and stiffness" in the last sentence of the paragraph.

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Revision Level	Page No.	Comment
	03-00-00 pgs. 2 to 5	03-5.a. Revised for addition of Garmin G1000 option. 03-8.a. Revised main gear tire size. 03-11. Numbered subparagraphs. Indicated differing voltage depending upon aircraft S/N and Garmin G1000 annunciation in last subparagraph, and numbered it 03-11.e. Revised empty weight of 2200 to 2400 and maneuvering speed of 149 to 158.
G		**Chapter 4**
	Revisions pg. 1	Indicated Rev. G.
	All pgs.	Revised header.
G		**Chapter 5**
	TOC pg. 1	Revised Table of Contents.
	05-10-00 pgs. 1 to 3	Revised Advisory Circular (AC) and FAR references for Item nos. 3 and 4. Added Items 17 and 18 for Garmin GFC autopilot system and renumbered following items. Revised Item 21. Moved Items 27, 28, and 29 after the "Recommended Replacements" title and renumbered to Items 34, 35, and 36.. Moved Items 30, 31, 32, and 33 in front of the "Recommended Replacements" title and renumbered to Items 30, 31, and 32. Added Item 36 and renumbered following Items. Revised Item 37 from PFD to MFD, added Garmin G1000 MFD, and renumbered to Item 40. Added inspection of engine mount bushings to Item 39 and renumbered to Item 42.
	05-20-00 pgs. 1 through 13	05-4. Deleted "Lancair" from the paragraph title. Deleted "Lancair" and changed "Check for fuel stains on floor" to "Check for fuel stains on underside of aircraft" in Figure 5-2. Deleted "Lancair" from the Instructions for Continued Airworthiness checklist, and revised Item 6, Item 16, Item 27, Item 31, Item 139, and Item 147. Added Chapter 30 to Engine and Propeller title. Added Item 53 Propeller Heat/De-ice System and renumbered following items. Revised Item 60 Battery box and renumbered to Item 61. Revised Item 75 Exterior emergency door release and renumbered to Item 76. Deleted Belleville washers from Item 77 Rudder and renumbered to item 78. Added Item 103 Garmin GFC 700 Autopilot System. Revised item 126 Oxygen system and renumbered to item 128. Added item 140 Garmin G1000 System. Renumbered following items. Revised Item 139 Fuel selector valve and renumbered to Item 142 Revised Item 151 MLG box support structure and renumbered to Item 154.
	05-22-00 pgs. 1 and 2	Added pitot drain to items accessible through the access panel on the underside near left wing root (Fuselage Exterior). Revised Fuselage Interior access panel items for avionics. Added power plug and pitch servo under hat rack. Deleted access panel in right side of rudder from the Controls area.
G		**Chapter 6**
	TOC pg. 1	Revised Table of Contents.
	06-00-00 pg. 1	06-1.g. Deleted. Renumbered 06-1.h. to 06-1.g. 06-3.e. Revised low and high propeller blade angle.
	06-00-00 pg. 4	06-12.f. Revised main gear tire size.

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G		**Chapter 8**
	TOC pg. 1	Revised Table of Contents.
	08-00-00 pgs. 1 to 8	08-2. Added Caution and example regarding specific weight of Aviation Gasoline. Repaginated pages. 08-3. Changed “Smart Levels” to “digital inclinometers”.
G		**Chapter 11**
	11-20-00 pgs. 2 and 3	Revised Avgas Only placard. Indicated S/N 42001 to 42500 for 12 v ground power placard. Added 24 v ground power placard for S/N 42501 and on. Changed “On Interior of Gascolator” to “On Interior of Gascolator Door”.
	11-30-00 pgs. 1 through 10	Added placards unique to Garmin G1000 option avionics and retitled placards unique to Basic or Avidyne option avionics. Revised Center Console Below Radios placard. Revised Above 12,000 Ft (PA) Reduce: placard. Revised Manifold Pressure placard. Revised Alt. Static placard. Added Panel Light Dimmer placard. Revised Fuel Selector Knob placard. Revised Maximum Fuel Imbalance placard. Added CO Detector placard. Added placard for sealing the air conditioning system bay access cover. Repaginated pages. Added pages 9 and 10.
G		**Chapter 12**
	12-00-00 pg. 1	12-2.b.3. Revised procedure. 12-2.b.4. Changed “line” to “fitting”.
	12-30-00 pg. 2	12-18.a. Added “and Figure 12-2” to paragraph title.
	12-30-00 pgs. 4 through 8	Added Figure 12-2 Columbia Aircraft Manufacturing Corporation Nose Strut Service. 12-20. Indicated two different battery manufacturers depending upon aircraft S/N. 12-21.b. Added reference to Chapter 51 for approved window cleaning materials and expanded procedures. Updated figure titles and cross references. Repaginated pages.
G		**Chapter 20**
	TOC pg. 1	Revised Table of Contents.
	20-10-00 pg. 7	20-9.c. Added paragraph specifying torquing with cotter pin installation of castellated nuts.
	20-50-00 pg. 2 and 3	Changed “Lancair” to “CAM” in Figure 20-31 and Figure 20-32.

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Revision Level	**Page No.**	**Comment**
	20-70-00 pgs. 1 through 26	20-22.d.2. Added cross reference to Figure 20-34. 20-22.d.5. Revised cross reference. Added Figure 20-34 and renumbered following figures. 20-23.k. Indicated a checkout procedure for both Avidyne avionics and the Garmin G1000 system. Added Garmin G1000 system Checkout Description table. In the Checkout Description table, Indicated differing voltages depending upon aircraft S/N for Item nos. 4 and 5. In the Checkout Description table, Indicated differing voltages depending upon aircraft S/N for Item no. 39. 20-24.c.1(d) Revised paragraph to add Eccobond 64C A/B. 20-24.c.1(e) Changed “400” to “200”. Added “Do not abrade through the lightning protection mesh of the aircraft.” 20-24.c.1(j) Deleted “Do not exceed 300°F. If an equivalent conductive adhesive is used, follow the manufacturer’s instruction for minimum cure temperature and cure time.” 20-24.c.1(l) Changed “less than 0.5” to “0 to 2.0”. 20-24.c.2(c) Revised paragraph to add Eccobond 64C A/B. 20-24.c.2(e) “less than 0.5” to “0 to 2.0”. 20-24.c.2(i) Deleted “Do not exceed 300°F. If an equivalent conductive adhesive is used, follow the manufacturer’s instruction for minimum cure temperature and cure time”. 20-24.f.4 Changed “0.5” to “2.0”. Repaginated pages. Added pages 17 through 26.
G		**Chapter 21**
	21-00-00 TOC pg. 1	Revised Table of Contents.
	21-00-00 pgs. 1 and 2	21-1.a. Revised to indicated ECS or ACCS option for environmental control. Revised description of system. 21-1.c. Deleted the first sentence and the last two sentences. 21-1.d. Revised description of eyeball vents in the instrument panel. 21-1.e. Revised grammar. 21-1.f. Deleted last sentence dealing with vent operation. 21-1.g. Revised description of rear passenger eyeball vents. Revised Figure 21-1. Repaginated pages.
	21-20-00 pgs. 1 through 8	Revised sections to describe differing conditions due to Garmin G1000 system. Added Figure 21-5. Revised cross references. Repaginated pages. Renumbered paragraphs. Added pages 7 and 8.
	21-40-00 pgs. 1 and 2	Deleted Figure 21-6. Added Figure 21-8. Revised cross references. Renumbered paragraphs. Deleted pages 3 and 4.
	21-60-00 pgs. 1 through 4	21-12. Indicated one of three types of temperature and air control units possible in the aircraft. Renumbered figures and revised cross references. Renumbered paragraphs.

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Revision Level	**Page No.**	**Comment**
	21-100-00 pgs. 1 through 34	21-13.c. Revised to indicate Diagnostics Fault Code display. 21-13.c.1. Added advanced diagnostics. 21-13.c.2. Added firmware version. 21-13.c.3. Added Parameter function and NOTE. 21-13.d.1. Added cabin sealing requirements. Renumbered following paragraphs. 21-13.d.3 i) Revised to include oil coat to o-rings and added trinary switch torque. 21-13.d.3 m) Added paragraph. 21-13.e. Indicated differing locations of control head due to Garmin G1000 system. 21-13.e.2.b) 3) Changed “Tighten snug” to “Torque 275 to 325 in.-lb. 21-13.e.3 a) ii) Changed “panel” to “access cover”. 21-13.e.3 b) Revised to include additional procedures to seal the air conditioning system bay and cover. 21-13.e.6. Revised Receiver/Dryer with Trinary Switch paragraph to indicate the trinary switch may be removed without evacuating the system. Added torque of 7 ft.-lbs. for the trinary switch. 21-13.e.7. Added location of cabin temperature sensor with G1000 system installed. Added Figure 21-28. 21-13.e.8 a) ii) Changed “panel” to “access cover”. 21-13.e.8 b) Revised to include additional procedures to seal the air conditioning system bay and cover. 21-13.e.9. Added cross reference to Figure 21-31. Revised title to Figure 21-27. Added Figure 21-28. 21-13.e.10. Added cross reference to Figure 21-33. Revised title to Figure 21-28 and renumbered it to 21-32. Added Figure 21-33. 21-13.e.11. Added cross reference to figure 21-33. 21-13.e.12 a) Revised for alternate location of ECU for S/N 42502 and on. 21-13.i.12. Changed “2 lbs. (907 grams)” to “24 oz. (680 grams)”. 21-13.i.13. In NOTE, changed “2 lbs. (907 grams)” to “24 oz. (680 grams)”. Repaginated pages. Added Pages 31 through 34. Renumbered paragraphs.
G		**Chapter 22**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	22-00-00 pgs. 1 through 14	Added S-Tec title to page. 22-1. Revised paragraph title. Revised page numbering and repaginated pages.
	22-01-00 pgs. 1 through 10	Added paragraphs for Garmin GFC 700 Autopilot.
G		**Chapter 23**
	23-00-00 pg. 1	23-1. Added indication that Garmin GMA 1347 audio panel may be installed in the aircraft.
	23-40-00 pgs. 5 through 8	23-8. Indicated Nav/Com bypass switch not installed with Garmin G1000 option. 23-9. Indicated location of headphone jacks for Garmin G1000 option. 23-9.e.1. Indicated only one panel for Garmin G1000 option. 23-10. Revised Stereo Audio Input Jack to include Garmin G1000 option. Repaginated pages.
G		**Chapter 24**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.

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Revision Level	**Page No.**	**Comment**
	24-00-00 pgs. 1 through 3	24-1. Revised section to add Garmin G1000 system. Deleted "60-amp (continuous output)". 24-1.f. Added air conditioning circuit breaker. 24-1.g. Added air conditioning circuit breaker. Revised Figure 24-1. Added Figure 24-2 and Figure 24-4. Revised title of Figure 24-3. Repaginated pages and added pages 7 and 8.
	24-30-00 pgs. 1 through 16	24-4. Added "(S/N 42001 to 42500)" to paragraph title. Revised Figure 24-5. Revised the titles of Figure 24-8 and Figure 24-9 and renumbered to Figure 24-10 and 24-11. Revised Figure 24-10 and renumbered it to 24-12. 24-5. Added Normal Battery System (S/N 42501 and on). 24-5.c. Added Caution that battery cables must be tied to tie blocks to keep from chafing against the cowling. 24-5.d. Revised title to indicate (S/N 42001 to 42500) and stated that this voltmeter is not present in aircraft S/N 42501 and on. Renumbered paragraph to 24-6.d. Renumbered following paragraphs. 24-6. Indicated difference due to Garmin G1000 system. Added pages 13 through 16.
	24-40-00 pgs. 1 and 2	24-7. Revised paragraph to indicate new design of ground power plug. Added new figure for Ground Power Plug Installation (S/N 42501 and on). Renumbered paragraphs. Revised figure nos. and cross references.
	24-50-00 pgs. 1 and 2	Revised Figure 24-11 and renumbered to Figure 24-16. Revised Figure 24-12 and renumbered to Figure 24-14. Renumbered paragraphs. Revised figure nos. and cross references.
G		**Chapter 25**
	TOC pgs. 1 and 2	Revised Table of Contents.
	25-00-00 pg. 2	25-1.e. Revised Engine, Flight Instrument, and Electrical Panels paragraph for Garmin G1000 Option.
	25-00-00 pgs. 4 through 8	Revised titles of Figure 25-2 and Figure 25-3. Added Figure 25-4 for Garmin G1000 Option and renumbered following figures. Added pages 7 and 8.
	25-10-00 pgs. 1 and 3	Renumbered Figure 25-4 to 25-5 and Figure 25-5 to 25-6.
	25-11-00 pgs. 1 through 16	25-6. Revised Glare Shield removal and installation depending upon avionics option. 25-7. Revised Instrument Panel removal and installation depending upon avionics option. 25-8, 25-9, 25-10, 25-11, 25-14, 25-15, 25-18, and 25-19. Added (Basic or Avidyne Option) to the title. 25-13. Revised Master Switch Panel removal and installation depending upon avionics option. 25-17. Revised Flap Panel removal and installation depending upon avionics option. 25-20. Revised to indicate no test/reset switch or annunciator panel light for Garmin G1000 option. Revised removal and installation to include Garmin G1000 configuration. Revised CO detector testing to indicate Basic or Avidyne option and Garmin G1000 option. 25-21. Added tower removal and installation. 25-22 Added floor duct removal and installation. Added Figure 25-19 and renumbered following figures and cross references. Repaginated pages. Added pages 13 through 16.

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Revision Level	**Page No.**	**Comment**
	25-12-00 pgs. 1 through 6	25-21. Revised Center Console removal and installation depending upon avionics option and renumbered it to 25-23. 25-23.a.1(a) Changed “flat head” to “truss head” in the armrest assembly paragraph. Repaginated pages.
	25-13-00 pgs. 1 and 2	25-21.b.3 Revised to indicate mount at aircraft centerline. 25-21.b.4 Added for mount at front side panel. Renumbered figures. Revised cross references.
	25-30-00 pgs. 1 through 12	Revised Figure 25-28 and renumbered it to 25-32. 25-26. Added Left Instrument Panel (Garmin G1000 Option). 25-27. Added Right Instrument Panel (Garmin G1000 Option). 25-28. Added Lower Instrument Panel (Garmin G1000 Option). Added Figure 25-29 and renumbered following figures. 25-30.a.5. Changed “six” to “five”. Repaginated pages. Added pages 11 and 12.
	25-31-00 pgs. 1 and 2	Renumbered figures. Revised Figure 25-35 Access Panel Locations. Revised cross references.
	25-60-00 pgs. 1 through 3	25-46. Revised removal and installation of the ELT remote switch depending upon avionics option and renumbered to 25-51. Renumbered figures. Revised cross references.
	25-90-00 pg. 1	Renumbered figures. Revised cross references.
G		**Chapter 27**
	27-00-00 pgs. 2 through 4	27-2. Revised to describe trim system under Garmin G1000 system. Added Figure 27-3. 27-22.b.4. Deleted reference to part number SS2-32. 27-22.b.5. Deleted reference to part number SS2-32.
	27-01-00 pgs. 1 through 12	27-8.f. Indicated Figure 27-12 is for basic or Avidyne option only. For Garmin G1000 option the rudder limiter test button is in the overhead console. Revised to indicate differences due to Garmin G1000 system. Revised title of Figure 27-12. Renumbered Figure 27-3 through 27-12 to Figure 27-4 through 27-14. Revised cross references.
	27-10-00 pgs. 1 through 6	27-10.b.7. Changed “0.250” to “0.375”. Renumbered Figure 27-14 through 27-17 to Figure 27-15 through 27-18. Revised cross references.
	27-20-00 pg. 2 through 4	Revised Figure 27-18 and Figure 27-19. Renumbered Figure 27-18 and 27-19 to Figure 27-19 and 27-20. Revised cross references.
	27-30-00 pgs. 1 through 6	Renumbered Figure 27-20 through 27-23 to 27-21 through 27-24. 27-18.b.12. Revised torque for jam nuts. Revised cross references. Repaginated pages.
	27-50-00 pgs. 1 through 7	27-22.b.8. Changed “0.250” to “0.375”. Renumbered Figure 27-24 through 27-32 to Figure 27-25 through 27-33. Revised cross references.
	27-60-00 pgs. 1 through 3	Renumbered Figures 27-33 through 27-35 to Figure 27-34 through 27-36. Revised cross references.
G		**Chapter 28**
	TOC pg. 1	Revised Table of Contents.
	28-00-00 pgs. 1 through 6	28-1. Revised to describe differences due to Garmin G1000 system. Added Figure 28-2. Added pages 5 and 6.

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Revision Level	Page No.	Comment
	28-10-00 pgs. 1 through 9	28-4.c.1. Deleted snap-grip clamp. 28-4.c.2. Deleted snap grip clamp. 28-4.c.3. Added torquing requirement. 28-5.c.4. Revised torquing requirements. Renumbered Figure 28-2 through 28-12 to Figure 28-3 through 28-14. Replaced Figure 28-3 and renumbered to Figure 28-4. Revised cross references.
	28-11-00 pgs. 2 through 4	28-12.a.1(c) Revised PR 1428 manufacturer. Renumbered Figure 28-14 through 28-16 to Figure 28-15 through 28-17. Revised cross references.
	28-20-00 pgs. 1 through 26	28-14.a.4. Changed "Scotchbrite" to "Scotch-Brite™". 28-15. Added removal and installation of the fuel selector knob. 28-15.h. Added 500 and 1000 hour inspection criteria for the fuel selector valve. 28-18. Revised to indicate differences due to Garmin G1000 system. 28-18.b.2 a) 1) ii), 28-18.b.2 a) 4) ii), and 28-18.b.2 a) 4) iii) Revised paragraph to indicate 12 or 24 volts depending on aircraft S/N. 28-18.b.2 a) 5) i) 1) Revised paragraph to indicate ± 12 or ± 24 volts depending on aircraft S/N. Revised title of Figure 28-18 and renumbered to 28-19. Added Figure 28-19 and renumbered to 28-21. Deleted Figure 28-37. Added Figure 28-38. Revised figure nos. and cross references. Repaginated pages. Added pages 25 and 26.
	28-40-00 pg. 1	28-20. Indicated differences due to Garmin G1000 system. 28-21. Indicated Basic or Avidyne Option only. 28-22. Revised due to Garmin G1000 system.
G		**Chapter 30**
	All	Added chapter.
G		**Chapter 31**
	TOC pg. 1	Revised Table of Contents.
	31-00-00 pgs. 1 through 3	31-1. Indicated voltmeter/OAT/clock not installed with Garmin G1000 system. 31-2. Indicated differing Hobbs locations depending upon avionics installed. 31-2.c. Revised location of air switch if Avidyne or Garmin G1000 is installed. Revised Note. Repaginated pages.
	31-30-00 pgs. 1 through 4	31-5. Revised removal and installation of the hour meter. 31-7. Revised removal and installation of the air switch. Repaginated pages. Added pages 3 and 4.
G		**Chapter 32**
	TOC pg. 1	Revised Table of Contents
	32-10-00 pgs. 7 and 8	Added paragraph 32-5 Main Landing Gear Leg Abrasion Repair. Renumbered following paragraphs. 32-5.b. Changed "Scotchbrite" to "Scotch-Brite™" and renumbered to 32-6.b.
	32-20-00 pgs. 2 and 3	32-7.a.5. Indicated S/N 42001 to 42089. 32-7.a.6. Added removal of heel plate for S/N 42090 and on. Renumbered following paragraphs. 32-7.a.10. Added screws for heel plate attachment to nose strut fairing.
	32-20-00 pgs. 7 through 10	32-9.a.3. a) 3) and 32-9.b.3. a) 3) Revised to remove weight off the nose gear to allow rotation of the nose wheel assembly. Repaginated pages. 32-10. Changed "takeoff" to "landing" and "should" to "must" in the first sentence.

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Revision Level	**Page No.**	**Comment**
	32-40-00 pgs. 1 through 4	32-11. Revised main gear tire size. Revised paragraph numbers.
	32-41-00 pgs. 1 and 2	Revised paragraph numbers.
G		**Chapter 33**
	TOC pg. 1	Revised Table of Contents.
	33-00-00 pgs. 1 through 6	33-1. Revised to indicate difference due to Garmin G1000 option. 33-3.a. Deleted last two sentences. Added Figure 33-2. Repaginated pages. Added pages 5 and 6.
	33-10-00 pgs. 1 through 8	33-6. Revised paragraph to indicate LED lighting and also revised inspection, removal, and replacement procedures. 33-7.d. Indicated "(Avidyne Option Only)" in the title. 33-9. Revised paragraph due to Garmin G1000 option. Renumbered Figure 33-2 and 33-3 to Figure 33-3 and 33-4. Renumbered Figure 33-3 through 33-7 to Figure 33-4 through 33-8. Repaginated pages and revised cross references.
	33-40-00 pgs. 1 through 7	33-11.e. and 33-11.f. Revised paragraphs due to Garmin G1000 option. 33-12.c.3 Added application of RTV 162 silicone caulk. Renumbered figures and revised cross references.
G		**Chapter 34**
	TOC pgs. 1 and 2	Revised Table of Contents.
	34-00-00 pgs. 1 through 12	34-1.b.1. Indicated tubing color for the pitot and static systems. Revised cross references. Added Figure 34-3. 34-1.b.9. a) Deleted "oil temperature". Repaginated pages. Added page 12.
	34-10-00 pgs. 1 through 10	34-2.i.2. Revised cross reference. 34-2.j.1. Revised cross reference. 34-2.j.3. Revised connection of the static line. Added Figure 34-11 and renumbered following figures. Added Figure 34-14 and renumbered following figures. 34-2. Changed "altitude encoder" to "altitude encoder (not present if Garmin G1000 installed)" and renumbered paragraph to 34-3. 34-3.c. Added Garmin G1000 option. Revised Figure 34-14 and renumbered it to 34-16.
	34-20-00 pg. 1	34-5. Revised due to Garmin G1000 system. 34-6. Added "(Basic or Avidyne Option Only)" to the title and indicated no turn coordinator installed with Garmin G1000 option.
	34-30-00 pgs. 2 and 3	34-8.e.5. Added removal of RTV white silicone sealant if present. 34-8.f.2. Added installation of RTV white silicone sealant. Renumbered paragraph 34-12 Marker Beacon Antenna to 34-10.
	34-30-00 pg. 5 and 6	Renumbered paragraph 34-14 Instrument Landing System to 34-10 and revised paragraph to indicate differing conditions due to the Garmin G1000 system.
	34-50-00 pgs. 12 to 15	34-20 Revised to indicate Garmin G1000 Option.. 34-21 Revised Garmin Avionics Cooling Fan to describe Garmin G1000 option.

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Revision Level	**Page No.**	**Comment**
	34-60-00 pgs. 23 through 62	34-25.a.5. Indicated four rubber isolators. Revised Figure 34-52. Revised Figure 34-54, indicated (S/N 42001 to 42500) in the title, and renumbered it to Figure 34-55. 34-26. Indicated PFD voltage converters only installed in aircraft S/N 42001 to 42500. 34-27. Indicated MFD voltage converters only installed in aircraft S/N 42001 to 42500. 34-28. Added Garmin G1000 System including Figure 34-58 through Figure 34-67 and renumbered following paragraphs and figures. 34-28 Revised WX-500 Stormscope Processor removal and installation and renumbered paragraph to 34-30. 34-32. Added "(Basic or Avidyne Option Only)" to the title and indicated ORBCOMM antenna not installed with Garmin G1000 system. 34-35. Revised Ryan TCAD Transponder Coupler removal and installation depending upon basic, Avidyne, or Garmin G1000 options. Added Figure 34-68 and renumbered following figures. Revised title of Figure 34-59 and renumbered to 34-67. Added Figure 34-76 and renumbered following figures. Added paragraph 34-29 and renumbered following paragraphs. Added Figure 34-82, Figure 34-83 and Figure 34-84 and renumbered following figures. Repaginated pages. Added pages 45 through 62.
G		**Chapter 35**
	TOC pg. 1	Revised Table of Contents.
	35-00-00 pgs. 1 and 2	35-1. Indicated differing display due to Garmin G1000 system. 35-1.i. Indicated Basic or Avidyne option only.
	35-00-00 pgs. 4 and 5	35-4.b., 35-4.g., 35-4.k., 35-5.a., and 35-5.b. Revised for Garmin G1000 system. Renumbered following paragraphs. Repaginated pages.
	35-10-00 pgs. 3 through 6	35-7. Revised Note indicating wrapping of tapered pipe threads with teflon tape. 35-7.a.4. Added additional cleaning solvents. 35-7.b.2. Revised indicating wrapping of tapered pipe threads with teflon tape.
	35-10-00 pgs. 10 through 12	35-11.a.1. and 35-12.a.1. Revised for Garmin G1000 system. Repaginated pages.
	35-10-00 pg. 14	35-13. Revised title to indicate Basic or Avidyne option only.
	35-10-00 pg. 18	35-17. Revised table for Garmin G1000 system.
	35-10-00 pg. 19	35-18. Added annual inspection procedures with Garmin G1000 installed.
G		**Chapter 51**
	All	Removed and replaced entire chapter. Non structural repair procedures added to chapter. Reformatted and repaginated all pages.
G		**Chapter 53**
	53-10-00 pgs. 3 and 4	Revised Figure 53-2. 53-5.n. Added paragraph for Instrument Panel Mounting Brackets (S/N 42501 and on). 53-5.m. Revised title of paragraph and renumbered following paragraphs.
G		**Chapter 56**
	TOC pg. 1	Revised Table of Contents.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
	56-00-00 pgs. 1 and 2	56-1.a.1) Deleted cleaning procedures and added cross reference to Chapter 51 for approved cleaning materials and procedures. Repaginated pages.
	56-10-00 pgs. 1 through 10	56-4.a.3. Added for bevel on side window. Added Figure 56-3. Renumbered following paragraphs, cross references, and figures. Revised Figure 56-3 and renumbered to 56-4. 56-9.1 Changed "TLC" to "Columbia Aircraft Manufacturing Corporation". Repaginated pages.
G		**Chapter 61**
	TOC pg. 1	Revised Table of Contents.
	61-20-00 pgs. 1 through 8	61-5. Revised paragraph to indicate either an adjustment rod or a banana bracket for alternator adjustment. Added min. 5 thread engagement of governor cable into rod end of cable eyelet, and added cross references to Figure 61-4. Revised Figure 61-2 and 61-3. Added Figure 61-4. Renumbered Figure 61-4 to Figure 61-5. Repaginated pages. Added pages 7 and 8.
G		**Chapter 71**
	71-00-00 pg. 1	71-3. Changed "IO-550-NY2B" to "IO-550-N". Deleted Intake Manifold Pressure at Idle under Operating Limits.
	71-00-00 pgs. 9 through 12	Revised Figure 71-11 Engine. Repaginated pages 10 through 12.
	71-60-00 pg. 1	71-7.10. Changed "3/4" to "1/2".
G		**Chapter 74**
	74-30-00 pg. 1	74-2. Revised description of ignition switch location and added Caution note to the end of the section.
G		**Chapter 76**
	76-10-00 pgs. 1 through 5.	76-3. Revised to indicate friction or vernier throttle control. Added cross reference to Figure 76-2. Revised title of Figure 76-1. Added Figure 76-2. Revised title of Figure 76-3 and renumbered to Figure 76-4. 76-3.a.2. Revised paragraph to depend upon aircraft s/n. Revised Figure 76-4 and renumbered to Figure 76-6. Added Figure 76-5 and renumbered following figures. 76-3.a.4. Added cross reference to Figure 76-15 and Figure 76-16. 76-3.b.2. Added cross reference to Figure 76-15 and Figure 76-16. 76-3.b.5. Revised paragraph to depend upon aircraft s/n. Deleted paragraph 76-3.b.7. 76-3.c.4. Added min. 5 thread engagement of throttle cable into rod end of cable eyelet. Repaginated pages and revised cross references.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
	76-20-00 pgs. 1 through 14	Revised Figure 76-11, Figure 76-12, Figure 76-13, and Figure 76-14. Added Figure 76-15 and Figure 76-17. Renumbered following figures and revised cross references. 76-4.b.2. Added cross reference to Figure 76-15 and Figure 76-16. 76-4.b.8. Added procedure. 76-4.c.4. Added min. 5 thread engagement of fuel mixture control cable into rod end of cable eyelet. Revised Figure 76-21. Repaginated pages. Revised figure nos. and cross references. Added pages 13 and 14.
G		**Chapter 77**
	TOC pg. 1	Revised Table of Contents.
	77-00-00 pg. 1	77-1. Revised to indicate Basic, Avidyne, and Garmin G1000 system options.
	77-10-00 pgs. 2 through 4	77-4.c.2 Revised paragraph to indicate 5 or 10 VDC depending on aircraft S/N. 77-4.d., and 77-4.e. Added cross reference to Figure 77-2. 77-5. Revised for Garmin G1000 system. Revised title of Figure 77-1. Added Figure 77-2.
	77-20-00 pgs. 1 through 6	77-6. Revised to include Garmin G1000 system. Revised to indicate tube nut or bayonet attachment type CHT probes. 77-7. Indicated Garmin G1000 system has six exhaust gas temperature probes. Figure 77-4 Revised Title and renumbered to 77-5. Renumbered figures. Renumbered paragraphs. Repaginated pages.
G		**Chapter 78**
	78-10-00 pgs. 5 and 6	78-6.7. Revised to indicate flange nuts are wet torqued with 50W aviation oil 100 to 110 in.-lbs. rather than torqued 72-84 in.-lbs. 78-6.8. Added 0.38 in. clearance between tailpipes and heat shield on fuselage. Repaginated pages.
H		**Part 0 Administrative**
	Title Page, Pgs. ii, xlii to xlvi	Revised Title Page. Revised Log of Revisions. Revised Narrative Discussion of Revisions.
H		**Chapter 4**
	Ch. 4 Log of Revisions pg. 1	Updated page.
	LOEP pg. 1	Revised LOEP.
	04-00-00 pg. 5	04-3. Revised ELT battery replacement requirement.
H		**Chapter 5**
	LOEP pg. 1	Revised LOEP.
	05-10-00 pg. 1	Item 2. Revised ELT battery replacement requirement.
H		**Chapter 6**
	LOEP pg. 1	Revised LOEP.
	06-00-00 pg. 2	06-6.c. Revised the Maximum Empty Weight from 2580 lbs. to 2568 lbs.
H		**Chapter 20**
	LOEP Pgs. 1 and 2	Revised LOEP.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
	20-70-00 pg. 14	Revised Item 2. Perform tests with the crosstie open. With only the right bus energized verify the starter does not engage.
H		**Chapter 21**
	LOEP Pgs. 1 and 2	Revised LOEP.
	21-100-00 pgs. 1 to 36	21-15.a. Added optional electrically driven compressor and description of operation using ground power. 21-15.c. Changed “evaporator” to “evaporator temperature sensor” in the second paragraph. 21-15.d.3.i) Added FVC68D (PVE) for electric compressor. 21-15.e.1 and 21-15.e.1.a) Indicated two types of refrigerant oil depending on compressor installed. 21-15.e.2 Added optional electrical compressor and reformatted section. Revised the titles of Figure 21-19 and Figure 21-20. Renumbered following figures and cross references. Repaginated pages. Added pages 35 and 36.
H		**Chapter 23**
	LOEP pg. 1	Revised LOEP.
	23-40-00 pg. 8	23-10.b.2.c) Added Loctite 243.
H		**Chapter 24**
	LOEP pgs. 1 and 2	Revised LOEP.
	TOC Pg. 1	Revised TOC.
	24-30-00 pg. 1	24-3. Added indication of third alternator for optional equipment.
	24-30-00 pgs. 4 to 20	Added Figure 24-9. Renumbered following figures and cross references. 24-3.b.3 and 24-3.b.4. Added Alternator Pulley and Drive Pulley removal and replacement. 24-3.d and 24-3.e. Renumbered to 24-3.c.3 and 24-3.4. 24-3.d. Added Accessories Alternator. Revised 24-3.f. and 24-3.g .for the differing belt adjustment conditions and renumbered paragraphs to 24-3.e. and 24-3.f., respectively. Repaginated pages. Added pages 17 to 20.
	24-50-00 pgs. 2 to 4	Added 24-11 Electric A/C Interlock Assembly. Added pages 3 and 4.
H		**Chapter 25**
	LOEP pgs. 1 and 2	Revised LOEP.
	TOC pg. 2	Revised Table of Contents.
	25-60-00 pgs. 1 to 4	25-49 to 25-52. Revised for addition of the Artex ME406 ELT.
	25-90-00 pg. 1	25-54. Updated Figure Nos. and cross references.
H		**Chapter 34**
	LOEP pgs. 1 to 3	Revised LOEP.
	34-60-00 pg. 22	34-24.a.8. Changed “screws” to “nuts”. 34-24.b. Added requirement to tighten nuts snug.
	34-60-00 pg. 35	34-28.j.2.b) Revised paragraph and added requirement to tighten nuts snug.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
H		**Chapter 52**
	LOEP pg. 1	Revised LOEP.
	52-10-00 pg. 8	52-10.a.8. Added item to allow AN690-516 washer between lock and lock tab if required.
I	All	Removed all occurrences of Columbia, Columbia Aircraft Manufacturing Corporation, etc. and replaced with Cessna as applicable. Repaginated pages.
J		**Part 0 Administrative**
	Title Page, Pgs. ii, xxxiii to lii	Revised Title Page. Revised Log of Revisions. Revised Narrative Discussion of Revisions. Revised Maintenance Manual Table of Contents.
J		**Chapter 2**
	LOEP pg. 1	Revised LOEP.
	02-00-00 pg. 3	Changed the publication number for the G1000 Cockpit Reference Guide from "190-00567-00" to "190-00567-01". Deleted "-2/USA" from publication number AWBCMM0001 under Chapter 32 – Wheels and Brakes.
J		**Chapter 4**
	Ch. 4 Log of Revisions pgs. 1 and 2	Updated pages.
	LOEP pg. 1	Revised LOEP.
	04-00-00 pgs. 1, 4, and 5	04-1.b. Revised paragraph and indicated 25,200 TIS hours for Major Airframe Inspection instead of four Airframe Service Cycles. 04-1.d. Revised the example at the end of the paragraph. 04-2. Added heading to the table at pg. 4. 04-3. Changed the airplane life limit from "12,000 hours" to "25,200 flight hours".
	04-10-00 pg. 1	04-4. Changed "12000" to "25,200".
J		**Chapter 5**
	LOEP pg. 1	Revised LOEP.
	05-10-00 pgs. 2 and 3	Added Item 15, Inspect aileron linear bearing, and Item 35, Garmin GDC 74A, to Figure 5-1. Revised following item nos. Item 21. Indicated testing location and renumbered to Item 22.
	05-20-00 pg. 8	Item 81. Revised inspection of trim servo.
	05-22-00 pgs. 1 and 2	05-5. Added Aileron linear bearing access panel to the Left Wing and the Right Wing.
J		**Chapter 6**
	LOEP pg. 1	Revised LOEP.
	06-00-00 pg. 2	Revised the Fuel Quantity row in Figure 6-1.
J		**Chapter 11**
	LOEP pg. 1	Revised LOEP.
	11-20-00 pg. 2	Revised the Near Fill Cap Of Fuel Tank placard.
	11-30-00 pgs. 4 to 8	Revised the On Bottom of Baggage Compartment Door Opening placard. Revised the Compass Calibration placard. Revised the Engraved On Fuel Selector Knob and Upper Plate placard. Repaginated pages

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
J		**Chapter 12**
	LOEP pg. 1	Revised LOEP.
	12-10-00 pg. 1	12-7. Revised the fuel capacities.
	12-30-00 pg. 6	12-20. Revised the list of installed batteries.
J		**Chapter 20**
	LOEP pgs. 1 and 2	Revised LOEP.
	20-50-00 pgs. 2 and 3	Added Dow Corning DC 7 to Figure 20-31. Added Loctite 660 to Figure 20-32.
	20-70-00 pg. 5	20-22.f.3. Changed “less than 100 ohms” to “0 to 500 ohms” and split paragraph to make 20-22.f.4.
J		**Chapter 21**
	LOEP pgs. 1 and 2	Revised LOEP.
	TOC pgs. 1 and 2	Revised Table of Contents.
	21-20-00 pgs. 3 and 8	21-6.a.2.a) and 21-6.b.1. Added for alternative defrost/floor valve assembly to defrost channel configuration. Revised Figure 21-6.
	21-40-00 pg. 1	Revised Figure 21-7.
	21-100-00 pgs. 7 to 40	21-15.d.1.a) Indicated there may be one plug rather than two clear panels. 21-15.e.2.b) Revised statement regarding the trinary switch and system pressures. 21-15.e.6.a)4) Changed “v” and “vii” to “5” and “7”, respectively. 21-15.e.7. Changed “is controlled by” to “provides input data to”. Added alternative location of cabin temperature sensor for the Garmin G1000 option. 21-15.e.7.a) Revised removal procedures for the cabin temperature sensor. 21-15.e.9. Revised for alternative defog/floor vent valve assembly to defrost channel configuration. 21-15.j. Added System Flushing paragraphs. Changed all occurrences of “21-10” and “21-13” to “21-15”. Revised Figure 21-15, Figure 21-28, Figure 21-31, and Figure 21-33. Repaginated pages. Added pages 37 to 40.
J		**Chapter 22**
	LOEP pg. 1	Revised LOEP.
	22-01-00 pg. 1	22-17. Changed all occurrences of “GDU 1042” to “GDU 1042/GDU 1044” and “GIA 63” to “GIA 63/GIA 63W”.
	22-01-00 pg. 4 and 6	Revised page footer.
	22-01-00 pg. 9	22-24. Changed “GIA 63” to “GIA 63/GIA 63W”.
J		**Chapter 23**
	LOEP pg. 1	Revised LOEP.
	23-40-00 pg. 3	23-5 Revised for two types of speaker installation. Revised Figure 23-3.
J		**Chapter 24**
	LOEP pg. 1	Revised LOEP.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
	TOC pg. 1	Revised Table of Contents.
	24-00-00 pg. 4	Revised Figure 24-2.
	24-30-00 pgs. 9 to 20	24-3.d.2.d) Revised installation criteria for alternator to adjustment rod bolt. 24-4.c.3. Added Note to not seal holes or gaps in battery box, or space between grommet and wiring. 24-5.a. Added description of battery charging circuit. 24-5.c.3. Added torque of nut or bolt. 24-5.c.4. Added Note to not seal space between grommet and wiring. 24-5.d. to 24-5.g. Revised battery charging, testing, maintenance, and reconditioning procedures. Revised Figure 24-13 and 24-14.
	24-50-00 pgs. 1 to 3	24-8. Revised for voltage regulators located under the insturment panel. 24-11. Revised the description of the electric A/C interlock. Repaginated pages.
J		**Chapter 25**
	LOEP pgs. 1 and 2	Revised LOEP.
	TOC Pg. 2	Revised Table of Contents.
	25-10-00 pgs. 3 and 4	25-4.a.4. Added retention clip.
	25-11-00 pgs. 1 and 5	25-6.b.1.a) and 25-7.b. Changed "GDU 1042" to "GDU 1042/GDU 1044".
	25-11-00 pgs. 13 and 14	25-20.b Added statement that CO detector may be in the tower on Garmin G1000 aircraft. 25-20.b.1 Indicated removal of the tower if applicable.
	25-30-00 pgs. 10 and 11	25-40 Added statement that baggage floor carpet on some aircraft has a flap allowing removal of the access panel beneath without removing the carpet.
	25-100-00 pgs. 1 and 2	25-55 Added Cupholder (Garmin G1000 Option).
	25-110-00 pgs. 1 and 2	25-56 Added Footwell.
J		**Chapter 27**
	LOEP pgs. 1 and 2	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	27-00-00 pg. 1	Revised Figure 27-1.
	27-01-00 pgs. 3 to 11	27-6.a.9. Revised weight modification for mass balance procedure. 27-6.a.10. Added paragraph. Revised Figure 27-7 and Figure 27-9. Repaginated pages.
	27-10-00 pgs. 1 to 8	27-10.a.3. Added step to disconnect the trim servo wire. 27-10.b Revised aileron rigging and adjustment procedures. 27-10.c. Added inspection of aileron linear bearing. 27-13.b.3. Changed the servo tab neutral position from 3º to 2º. Revised Figure 27-15. Revised Figure 27-18. Added pages 7 and 8.
	27-20-00 pgs. 1 to 6	27-15.a.5. Added clearance between rudder actuator and rudder attachment assembly. 27-15.b.11.a) 14) Cross referenced Chapter 22 if autopilot is installed. 27-15.b.11.b) 18) Cross referenced Chapter 22 if autopilot is installed. Added Figure 27-19, Rudder Actuator Clearance. Revised Figure nos. and cross references. Repaginated pages.

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Revision Level	**Page No.**	**Comment**
	27-30-00 pgs. 1, 2, 3, 5, and 6	Revised Figure nos. and cross references.
	27-50-00 pgs. 1 to 7	27-22.c. Changed "OKS 475B" to "MIL-PRF-23827 Type II". Revised Figure nos. and cross references.
	27-60-00 pgs. 1 to 3	Revised Figure nos. and cross references.
J		**Chapter 28**
	LOEP pgs. 1 and 2	Revised LOEP.
	28-00-00 pg. 4	28-1.f. Indicated quantity of usable fuel available by aircraft S/N when the low fuel annunciators activate.
	28-10-00 pg. 7	28-8.b.3. Revised number of bolts securing the slosh box lid.
	28-20-00 pgs. 12 to 14	28-15.b. Added CAUTION about alignment of fuel selector shaft. 28-15.b.5. Revised to note and record position of the fuel selector shaft or to mark the shaft to ensure correct alignment during reassembly. 28-15.c.7. Revised to ensure alignment and position are maintained for the fuel selector shaft after assembly. Revised Figure 28-34.
J		**Chapter 30**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.
	30-60-00 pgs. 5 to 18	30-5. Indicated SMR Technologies or Hartzell propeller heater boots installed on aircraft. Added removal and installation procedures per Hartzell Propeller Manual 182 (61-12-82). Added address for Hartzell. 30-5.a.3. and 30-5.b.3. Added Hartzell Propeller Manual 182. 30-5.b.3 to 30-5.b.10. Added installation procedures for Hartzell heater boots. 30-5.c. Deleted sentence to remove the bracket. 30-5.d. Added Hartzell Propeller Manual 182. Revised the title of Figure 30-7. Added Figure 30-8 and renumbered following Figures and cross references. Added pages 17 and 18.
J		**Chapter 32**
	LOEP pg. 1	Revised LOEP.
	32-10-00 pg. 2	32-3.b. Added cross reference to Chapter 57 for anti-chafe tape removal and installation.
	32-10-00 pgs. 7 and 8	32-6.b. Changed "Loctite 675" to "Loctite 660". Repaginated following page.
	32-40-00 pg. 1	32-13. and 32-14.a.5. Revised reference to indicate the "latest issue" of Cleveland Maintenance Manual AWBCMM0001. Deleted "-2/USA".
	32-41-00 pg. 2	32-15.b.3 Revised reference to indicate the "latest issue of Cleveland Maintenance Manual AWBCMM0001".
J		**Chapter 33**
	LOEP pg. 1	Revised LOEP.
	TOC pg. 1	Revised Table of Contents.

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Revision Level	**Page No.**	**Comment**
	33-40-00 pgs. 3 to 8	33-12.a. Revised to indicate xenon taxi lights. Deleted reference to Figure 33-12. Deleted Figure 33-12. Revised Figure 33-11. 33-12.b.2. Changed “four” to “the mounting”. 33-12.c.2. Changed “four” to “the mounting”. 33-12.d. Indicated taxi light ballast. 33-12.d.3. Revised locknuts and washers. 33-13.f.1. and 33-13.f.2. Added Precise Flight Taxi and Landing Light requirements. Revised Figure 33-13 and Figure 33-14 and renumbered to Figure 33-12 and Figure 33-13. Revised Figure nos. and cross references.
J		**Chapter 34**
	LOEP pgs. 1 to 3	Revised LOEP.
	TOC pg. 2	Revised Table of Contents.
	34-20-00 pg. 3	34-7.b. Added Magnetic compass base re-installation. Revised Figure 34-23.
	34-60-00 pgs. 28 to 33	34-28.b. Changed “GDU 1042” to “GDU 1042/GDU 1044 MFD” in the paragraph title. Indicated the GDU 1044 required for WAAS utilization. Revised Figure 34-59. 34-28.d. Added GIA 63W to the paragraph title. Revised the paragraph to indicate the GIA 63W is required for WAAS utilization. Changed “GIA 63” to “GIA 63/GIA 63W” in the removal and replacement procedures. Added “, if installed” to WX-500 StormScope and Ryan TCAD. 34-28.e. and 34-28.f. Changed “GIA 63” to “GIA 63/GIA 63W” and “GDU 1040 MFD” to “GDU 1042/GDU 1044 MFD”. 34-28.g. Added pitot and altimeter testing required every 2 years.
	34-60-00 pg. 35	34-28.k. Revised the paragraph to indicate differing antenna conditions due to WAAS.
	34-60-00 pgs. 37 to 62	34-28.n. Added GA 35 Antenna (WAAS) removal and installation procedures. 34-28.o. Added GA 37 Antenna (WAAS) removal and installation procedures. Renumbered following paragraphs. 34-28.p.2.a) Revised the paragraph title to include the GDU 1044 MFD and renumbered to 34-28.r.2.a). 34-28.p.2.c) Changed “GIA 63” to “GIA 63/GIA 63W” and renumbered to 34-28.r.2.c). Repaginated pages.
J		**Chapter 51**
	LOEP pgs. 1 and 2	Revised LOEP.
	TOC Pg. 1	Revised Table of Contents.
	51-70-00 pgs. 2 to 24	51-6.f.2. c) Revised proportions to “Mix per manufacturer’s instructions”. 51-6.g.2. d) Revised proportions to “Mix per manufacturer’s instructions” and revised hardener and thinner table. 51-6.g.3. c) and 51-6.g.3. g) Revised proportions to “Mix per manufacturer’s instructions”. 51-6.i.1. Changed “3/8” to “1/2”. 51-6.i.3. a) Revised list of approved paints. 51-6.k. Revised paragraph and added cross reference to Figure 51-11. Added Figure 51-11 Approved Paint Colors. 51-6.k.1 and 51-6.k.2. Added paragraphs for use of Zone C paint in a Zone B area. Added pages 23 and 24.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
	51-80-00 pgs. 1 to 4	Revised Figure nos. and cross references.
J		**Chapter 52**
	LOEP pgs. 1 and 2	Revised LOEP.
	TOC Pg. 1	Revised Table of Contents.
	52-10-00 pgs. 5 to 9	52-8.a.3. Revised to indicate surround clips may be rotated. Repaginated pages.
	52-10-00 pgs. 14 to 16	52-14 Added Remote Keyless Entry System. Added pages 15 and 16.
	52-20-00 pgs. 1 to 6	52-15. Added cross reference to new Figure 52-20. Revised title of Figure 52-19. added Figure 52-20. 52-18. Added cross reference to Figure 52-20. Deleted number of electrical connections and added pressure switch setting of 10 psi for pump type B. Added removal/installation of pump type B. Revised paragraph nos., Figure nos. and cross references.
J		**Chapter 53**
	LOEP pg. 1	Revised LOEP.
	53-30-00 pg. 2	53-6.i. Cross referenced footwell removal and installation procedures to chapter 25.
J		**Chapter 56**
	LOEP pg. 1	Revised LOEP.
	56-10-00 pgs. 8 to 10	Revised Figure 56-7. Repaginated pages.
J		**Chapter 57**
	LOEP pg. 1	Revised LOEP.
	57-00-00 pg. 1	57-1. Added absence of the aft rib at WS 151.0 on aircraft built at the end of 2007 and later.
	57-10-00 pgs. 7 to 10	57-6.a.5. Added absence of the aft rib at WS 151.0 on aircraft built at the end of 2007 and later. 57-7.a. Changed “14” to “16”. 57-7.b. Added item 9 to the list of access panels. Revised Figure 57-9. 57-10. Revised paragraph to add horizontal stabilizer, vertical stabilizer, cowling near the latch, door tread, tow bar attachment point, wheel pants, gear leg fairings, and landing light lens. Repaginated pages.
J		**Chapter 71**
	LOEP pg. 1	Revised LOEP.
	71-00-00 pgs. 10 to 12	71-4.a.25. Added heat shield if present. 71-4.c.1. Added heat shield (if applicable), and clearance criteria. 71-4.c.21. Added application of high temperature silicone.
	71-10-00 pgs. 1 to 4	71-5.a.1. Added use of drill/driver and added Note on drill speed and torque. 71-5.b. Revised cowling installation procedures. Repaginated pages.
	71-20-00 pgs. 1 and 2	71-6.c. Revised to allow an additional washer under the engine mount bolt head to provide a minimum of 2 and a maximum of 3.5 threads to extend through the nut after torquing. Revised Figure 71-17.

NARRATIVE DISCUSSION OF REVISIONS		
Revision Level	**Page No.**	**Comment**
J		**Chapter 76**
	LOEP pg. 1	Revised LOEP.
	76-00-00 pg. 1	76-1. Added NOTE about cosmetic shrink wrap on control cables. Added Note to apply high temperature silicone between hoses, wires, or lines touching the engine induction tubes, oil radiator, or engine mount.
	76-20-00 pgs. 7 and 8	76-4.c.7. Expanded control arm motion criteria. 76-4.c.8. Added clearance requirements. Repaginated pages.
J		**Chapter 77**
	LOEP pg. 1	Revised LOEP.
	77-00-00 pg. 1	77-1. Added NOTE to apply high temperature silicone between hoses, wires, or lines touching the engine induction tubes, oil radiator, or engine mount.

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CHAPTER 1

INTRODUCTION

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Chapter 1

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01-1. GENERAL

a. This manual is intended to assist mechanics and other authorized personal in maintaining the Cessna 350 (LC42-550FG). The manual provides detailed descriptions of systems, troubleshooting procedures, procedures for removal and installation of parts and components, and applicable maintenance procedures and techniques.

b. General features, characteristics, and limitations of the manual are described below.
   1. **Manual's Intended Use** – This document is intended for use as an *on-aircraft* manual, i.e., one that provides instructions for installation and removal of components.
   2. **Related Publications** – Detailed information associated with airplane parts, complex engine maintenance, and complex propeller maintenance, are contained in the applicable publications described in the Related Publications Section below.
   3. **Applicability** – The manual is applicable to airplane serial numbers 42001 and higher.

c. The Airworthiness Limitations section (Chapter 4) is FAA approved and specifies maintenance required under §§ 43.16 and 91.403 of the Federal Aviation Regulations unless an alternative program has been FAA approved.

01-2. RELATED PUBLICATIONS

a. The following manual and publications should be used in conjunction with the *Cessna 350 (LC42-550FG) Airplane Maintenance Manual*.
   1. *The Pilot's Operating Handbook* – Cessna publication number RB050000
   2. *The Cessna 350 (LC42-550FG) Parts Catalog* – Cessna publication number pending
   3. *The Cessna 350 (LC42-550FG) Electrical Manual* – Cessna publication number RB24000X
   4. *IO-550-N Teledyne Continental Maintenance Manual* – TCM publication number X30634A
   5. *IO-550-N Teledyne Continental Parts Catalog* – TCM publication number IPC550N
   6. *IO-550-N Teledyne Continental Overhaul Manual* – TCM publication number X30568A
   7. *Garmin G1000 System Maintenance Manual*, P/N 190-00577-03
   8. *Required Equipment List Garmin G1000 in Cessna Models LC41 and LC42*, Cessna publication number RB011002.

01-3. WARNING, CAUTION, AND NOTE

a. It is important to follow appropriate safety standards attendant with maintenance of the Cessna 350 (LC42-550FG). In some instances, specific safety precautions are described in the various chapters. In addition, the terms Warning, Caution, and Note are used to identify important issues. The format and meaning of these words are shown below.

WARNING

The use of a Warning symbol means that information which follows is of critical importance and concerns an operating procedure, inspection, repair, or maintenance practice, which could cause or result in personal injury or death if not carefully followed.

CAUTION

The use of a Caution symbol means that information which follows is of significant importance and concerns an operating procedure, inspection, repair, or maintenance practice, which could cause or result in damage to or destruction of equipment if not carefully followed.

NOTE

The use of a Note symbol means that an operating procedure, inspection, repair or, maintenance practice is essential to emphasize.

01-4. MANUAL OVERVIEW

a. The manual is set up according to guidelines established by the Air Transport Association of America (ATA) Specification 100. In addition, supplemental standards mandated by the General Aviation Manufacturers Association (GAMA) are included. The manual is divided into five major parts. With the exception of Part 0, all parts of the manual are further divided into chapters. Each chapter deals with a specific topic or system relevant to its part. Chapters are divided, when necessary, into subsystems or sections that are sometimes divided into a unit or subject. This method is discussed below under Chapter/Section Numbering.

01-5. MAJOR PARTS AND CHAPTERS

a. The topic of a particular chapter is established from a standardized ATA list, and the chapter numbering is not sequential. The ATA specifications provide flexibility and permit manufacturers to incorporate unique subjects in a consistent format by reserving chapter numbers. For the Cessna 350 (LC42-550FG), the chapter numbers are grouped into four major parts and shown in the table below. In addition, the chapters associated with each part are tabulated.

01-6. TABLE OF MAINTENANCE MANUAL CHAPTERS

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01-7. CHAPTER/SYSTEM NUMBERING

a. A chapter is also referred to as a system. It is frequently more convenient and logical to think of chapters as systems and chapter numbers as system numbers. The first two digits of the chapter number refer to a particular airplane system, sometimes called a primary system. Here the term "system" is used in a broad sense. The center two numbers refer to a subsystem (also called a section) of the primary system. Finally, subsystems can have a unit or subject. While there are obvious advantages associated with describing chapters as systems, the industry generally uses chapters, hence chapters are the form of reference in this manual.

01-8. NUMBERING SYSTEM TABLE

1	**26**-00-00 The number 26 refers to the fire protection system.	Information which is applicable to the system as a whole. This number is established by ATA.
2	26-**20**-00 The number 20 refers to the fire extinguishing subsystem.	Information which is applicable to the subsystem as a whole. This number is established by ATA.
3	26-**22**-00 The second "2" in the number 22 refers to the engine fire extinguisher.	Information applicable to the sub-subsystem as a whole. This number is assigned by the manufacturer.
4	26-22-**03** The number 03 refers to a particular type of extinguisher.	Information which is applicable to a specific unit of the subsystem. This number is assigned by the manufacturer.

Figure 1-1

a. The table above, Figure 1-1, describes the numbering system used in this manual using the example number 26-22-03, which classifies the fire protection system. While the fire

protection system in the Cessna 350 (LC42-550FG) is not this sophisticated, it does provide a clear demonstration of how a four-stage method of classification functions.

01-9. CHAPTER TOPICS

a. The arrangement of topics within each of the chapters are presented in a particular order. The three main topical categories and order of presentation are Description and Operation, Troubleshooting, and Maintenance Instructions. Maintenance Instructions are further divided into six additional topics. A summary of the arrangement of topics in this manual is shown below.

Description and Operation
Troubleshooting
Maintenance Instructions

1. Servicing
2. Removal and Installation
3. Adjustments/Tests
4. Checking/Testing
5. Cleaning/Painting
6. Repairs

b. The number system and topic presentation order discussed above in paragraphs 01-7 through 01-9 are suggested guidelines for all types of aircraft manufacturers and the scope, by necessity, is somewhat broad. In many areas, a procedure applicable to the Cessna 350 (LC42-550FG) is not as complex as for a commercial transport type airplane. For this reason some of the chapters in this manual are only two or three pages long. In this instance, to provide continuity in presentation and avoid pages with only one or two paragraphs, all the discussion is provided in a single section.

01-10. NUMBERING FIGURES AND PAGES

a. There are numerous exhibits or figures in the manual. The tables, charts, drawings, graphs, illustrations, etc. for each chapter are numbered sequentially beginning with number “1” and preceded by the chapter number and the word “Figure.” At the beginning of a new chapter the numbering sequence is restarted at “1.” For example, Figure 1-1 is the first figure in Chapter 1.

b. The introduction pages at the beginning of the maintenance manual are numbered with roman numerals. Thereafter the pages of each section are numbered sequentially with each section beginning with the number “1.” For the purpose of this manual, the list of effective pages and the table of contents are considered sections.

01-11. NUMBERING OF PARAGRAPHS

a. While the page numbers within a chapter are restarted at the beginning of each section, the paragraph numbers are sequential throughout the entire chapter. Each major paragraph is prefixed with the number of the chapter. For example, in the case of this paragraph, 01-11**,** the reader knows that this is the eleventh major paragraph item in Chapter 1.

01-12. AFTER-MARKET PARTS

a. Other chapters in Part 1 of this manual will discuss mandatory and recommended inspection intervals, replacement time limits, overhaul time limits, etc. The various time periods are based on the use of approved Cessna parts. If other than approved Cessna parts are used, the various inspection and replacement cycles are no longer applicable. If such parts are used, follow the inspection and replacement cycles established by the manufacturer of the part.

01-13. STC MODIFICATIONS

a. Modifications to the airplane, which are incorporated under a Supplemental Type Certificate (STC), can affect procedures and inspection and part replacement cycles. The holder of the STC will issue appropriate time intervals and procedures for areas of the airplane that are affected by the STC. Sometimes STC modifications can change operating characteristics, component loads, and component stresses. Since the interface of the STC modification with the existing structure can result in a number of unknown elements, the standards established by Cessna may not be valid.

CHAPTER 2

HOW TO USE THIS MANUAL

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02-1. GENERAL

a. To find information about a specific system, the table of contents for the entire maintenance manual is a good place to start. First, decide which part of the manual is applicable, and under the appropriate part, locate the desired chapter. Go to the table of contents of the chapter in question for an expanded itemization of the subject matter in that chapter.

02-2. REVISION SERVICE

a. **Normal Revisions –** The manual is normally revised on a schedule that varies from six months to a year. The controlling factor is the number of changes made to the manual within a given time period. On page i there is a Log of Revisions. The Log of Revision pages are updated by Cessna each time regularly scheduled revisions are issued. The Log of Revisions contains a list of all revisions made to the maintenance manual since its original issue. By comparing the Log of Revisions to the Record of Incorporated Revisions, the user of this manual can verify that all applicable revisions are included. In addition, at the bottom of each page the date of the initial issuance and latest revision dates are shown.

   The Narrative Discussion of Revisions lists each change that was incorporated and gives a brief description of the change and the affected page numbers. This section is located in the introductory material after the Service Bulletin Revision page.

b. **Temporary Revisions** – Temporary revisions are issued to facilitate timely changes of important information and are printed on blue paper. The revisions are numbered sequentially and can affect more than one chapter. Temporary revisions are normally incorporated in the next regularly scheduled revision cycle. Maintaining the Log of Temporary Revision is important. When a temporary revision is received, the first four columns of the log are used to document its incorporation into the maintenance manual. Later, when the temporary revision is superseded by incorporation of a scheduled revision, the temporary revision is removed from the manual and appropriate notations are made in the last two columns of the log.

02-3. SERVICE BULLETINS

a. Service Bulletins and Service Letters are issued by the manufacturer and may require a special inspection or modification to the airplane. At the time of delivery, applicable service bulletins are normally incorporated. When a subsequent Service Bulletin is issued after the delivery date, Cessna will notify the airplane owner concerning details for compliance. The Service Bulletin number, date, title and incorporation should be noted in the log provided on page vii. The Service Bulletin will be incorporated into the next scheduled revision of the maintenance manual. There are three types of Service Bulletins/Service Letters as described below.

   1. Immediate compliance is required.
   2. Compliance is required within a defined time period.
   3. No compliance required. (This is normally a Service Letter.)

02-4. REISSUE OF THE MAINTENANCE MANUAL

a. When the level of revised material comprises about 60% of the manual, the manual will be reissued. In the reissued manual, paragraphs, illustrations, tables, and pages are renumbered as necessary to eliminate numbering suffixes.

02-5. EFFECTIVITY

a. At the beginning of each chapter in the maintenance manual is a list of effective pages and the corresponding date of issue.

02-6. SERIAL NUMBER APPLICABILITY

a. Service Bulletins, Revisions, and Temporary Revisions issued for the Cessna 350 (LC42-550FG) may not be applicable to all the airplanes produced. When a particular change is issued, it will identify applicability by indicating the range of serial numbers affected by the change. If no serial number is noted, then the change is applicable to all Cessna 350 (LC42-550FG) airplanes. The serial number for the airplane is issued at the time of manufacture and remains with that airplane for its life. The airplane serial number, make, model, Type Certificate (TC) number, year of manufacture, and Production Certification (PC) number are contained on the Fuselage Identification Plate on the tail cone of the airplane. The serial number is also listed on the cover page of the FAA Approved Flight Manual.

02-7. LIST OF MANUFACTURERS' TECHNICAL PUBLICATIONS

a. The table below contains relevant technical publications for equipment and accessories installed on the Cessna 350 (LC42-550FG). The list is comprehensive and items shown on the list may not be in a particular airplane.

Manufacturer	Manuf. Part No.	Publication Number	Description of Item
Chapter 22 - Auto Flight			
S-Tec Corp One S-Tec Way Mineral Wells Mun. Airport Mineral Wells TX 96067	S-Tec 55X S-Tec 360	St-852	Autopilot & Autopilot Preselect Installation Manuals
Chapter 23 - Communications			
Garmin Aviation Technologies 2345 Turner Rd. SE Salem, OR 97309	SL15	560-0975-00	Audio Panel with Marker Beacons NAV/COMM
Garmin Aviation Technologies 2345 Turner Rd. SE Salem, OR 97309	SL30	560-0404-00	Audio Panel with Marker Beacons NAV/COMM
Garmin Aviation Technologies 2345 Turner Rd. SE Salem, OR 97309	14H ACU	56-1023-00	Annunciator Control Unit
Garmin International, Inc., 1200 East 151st Street Olathe, KS 66062	GMA 340	190-00149-10	Audio Panel Pilot's Guide

Manufacturer	Manuf. Part No.	Publication Number	Description of Item
Chapter 28 - Fuel			
Shadin Company Inc. 6831 Oxford Street St. Louis Park, Minnesota 55426	Miniflo-L		Digital Fuel Management System
Chapter 32 – Wheels and Brakes			
Parker Hannifin Corporation Aircraft Wheels & Brakes 1160 Center Road Avon, OH 44011	30-61E	AWBCMM0001	Cleveland Brakes Component Maintenance Manual
Parker Hannifin Corporation Aircraft Wheels & Brakes 1160 Center Road Avon, OH 44011	40-77	AWBCMM0001	Cleveland Wheels Component Maintenance Manual
Chapter 34 – Avionics			
II Morrow, Inc 2345 Turner Rd. SE Salem, OR 97309	GX 50/60 GPS & Comm	560-0959-00	Global Positioning System Installation Manual
Garmin Aviation Technologies 2345 Turner Rd. SE Salem, OR 97309	SL70	560-0402-02	ATCRBS Transponder
Mid Continent Instruments W 16585 Sherman Way, A-1 Van Nuys, CA 91406	MD200-306	8017972	Remote Analog Navigation Head with Glideslope
Trans-Cal Industries, Inc. 16141 Cohasset St. Van Nuys, CA	SSD120-30	930005	Encoder/Digitizer Installation Manual
Garmin International, Inc., 1200 East 151st Street Olathe, KS 66062	GTX 327	190-00187-00	Mode C Transponder Pilot's Guide
Garmin International, Inc., 1200 East 151st Street Olathe, KS 66062	GTX 330	190-00207-00	Mode S Transponder Pilot's Guide
Garmin International, Inc., 1200 East 151st Street Olathe, KS 66062	GNS 430	190-00140-00	GNS 430 Pilot's Guide and Reference
Garmin International, Inc., 1200 East 151st Street Olathe, KS 66062	G1000 System	190-00567-01	Cockpit Reference Guide for the Cessna 350/400
Garmin International, Inc., 1200 East 151st Street Olathe, KS 66062	G1000 System	190-00577-03	G1000 Maintenance Manual, Cessna 350/400
AlliedSignal, Inc Commercial Avionics Sys. 400 North Rogers Rd. Olathe, KA 66062-1294	KCS 55A	006-00111-0007	Horizontal Situation Indicator Installation Manual
AlliedSignal, Inc Commercial Avionics Sys. 400 North Rogers Rd. Olathe, KA 66062-1294	KI 256	006-08262-0000	Flight Director Installation Manual

Manufacturer	Manuf. Part No.	Publication Number	Description of Item
Avidyne 55 Old Bedford Road Lincoln, MA 01773	PFD	600-00096-000	FlightMax Entegra Series Pilot's Guide
Avidyne 55 Old Bedford Road Lincoln, MA 01773	EX5000	600-00102-000	FlightMax EX5000 MFD Pilot's Guide
Chapter 61 - Propeller			
McCauley Accessory Division 3535 McCauley Drive Vandalia, OH 45377-5053	20309-39	780401	Propeller Governor
Hartzell Propellers, Inc. One Propeller Place Piqua, OH 45356-2634	Model No. PHC-J3YF-IRF- and F7691d-1	115N (Latest Revision)	Three-blade Constant Speed Propeller
Chapter 71 – Engine			
Teledyne Continental Motors P.O. Box 90 Mobile Alabama 36601	IO-550-N	X305568A	IO-550-N Overhaul Manual
Teledyne Continental Motors P.O. Box 90 Mobile Alabama 36601	IO-550-N	X30569A	IO-550-N Parts Catalog
Teledyne Continental Motors P.O. Box 90 Mobile Alabama 36601	IO-550-N	30634A	IO-550-N Maintenance Manual
J.P. Instruments Inc. PO Box 7033 Huntington Beach, CA 92646	EGT-701	103	Digital Engine Scanner

CHAPTER 3

GENERAL DESCRIPTION OF THE CESSNA 350 (LC42-550FG)

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03-1. GENERAL

a. The Cessna 350 (LC42-550FG) is a pre-molded composite built, semi-monocoque, four seat, single engine, low wing, tricycle gear design airplane. The airplane is certified in the utility category and is used primarily for transportation and related general aviation uses. On the following few pages, a brief overview of the airplane and its basic systems is provided, as well as a three view drawing of the airplane.

03-2. BASIC CONSTRUCTION TECHNIQUES

a. The construction process used to build the shell or outer surfaces of the fuselage, wing, and most control surfaces involves creating a honeycomb sandwich. The sandwich consists of two outer layers of pre-preg fiberglass around a honeycomb interior. The term "pre-preg fiberglass" means the fibrous material is impregnated with catalyzed epoxy resin by the manufacturer of the fiberglass. This process ensures consistency in surface thickness and strength. The honeycomb sandwich is assembled in molds of the wing, fuselage, and control surfaces. A vacuum is used during the heat curing procedure to ensure proper compaction. Other structural components of the airplane, like ribs, bulkheads, and spars, are constructed in the same manner. In areas where added structural strength and stiffness is needed, such as the wing spars, carbon material is added to the honeycomb sandwich.

b. **Fuselage** – The fuselage is built in two halves, the left and right sides; each side contains the area from the firewall back to, and including, the vertical stabilizer. The bulkheads are inserted into the right side of the fuselage through a process known as secondary bonding. The two fuselage halves are bonded together, and the floors are bonded in after the fuselage halves are joined. Before the fuselage is assembled into one unit, cables, control actuating systems, and conduits are added at this point because of the ease in access. To prevent damage to the leading edge of the vertical stabilizer, anti-erosion tape may be installed.

c. **Wings and Fuel Tanks** – The bottom wing skin and two spars are one continuous piece. The spars are placed in the bottom wing and bonded to the bottom inside surface. Next, the ribs are inserted and bonded to the inside surfaces of the bottom wing and to the spars. Finally, after wires, conduits, and control tubes are inserted, the two top wing halves are bonded to the bottom wing and all the spars and ribs. The airplane has integral fuel tanks, commonly referred to as a "wet wing." The ribs, spars, and wing surfaces are the containment walls of the fuel tanks. All interior seams and surfaces within the fuel tanks are sealed with a fuel impervious substance. To prevent damage to the leading edge of the wing, anti-erosion tape may be installed.

d. **Horizontal Stabilizer –** The horizontal stabilizer is two separate halves bonded to two horizontal tubes that are bonded to the fuselage. The shear webs and ribs are bonded into the inside surface of the lower skin and the upper skin is then bonded to the lower assembly. To prevent damage to the leading edge of the horizontal stabilizer, anti-erosion tape may be installed.

03-3. FLIGHT CONTROLS (See Chapter 27)

a. **Ailerons and Elevator** – The ailerons and elevator are of one-piece construction with most of the stresses carried by the control surface. Additional structural support is provided by the end caps and the drive rib which is used to mount the control's actuating

hardware. The aileron and elevator control systems are operated through a series of actuating rods and bellcranks that run between the control surface and the control stick in the cockpit.

b. **Rudder –** The rudder is of one-piece construction with most of the stresses carried by the control surface. Additional structural support is provided by the drive rib which is used to mount the control's actuating hardware. The rudder control system is operated through a series of cables and mechanical linkages that run between the control surface and the rudder pedals in the cockpit.

03-4. RUDDER LIMITER (See Chapter 27)

a. The rudder limiter, which is an integral part of the stall system, is designed to limit the normal full left rudder deflection of 17° to only 11.5°. The rudder limiter system is automatically armed in a relaxed position when the aircraft's electrical power is turned on. The system is activated when two conditions exist, (1) the airplane's stall warning is active, and (2) the engine manifold pressure is more than 12 in. of Hg. When the system is activated, a solenoid near the left rudder pedal moves a cam that physically limits the travel of the left rudder pedal. There is a time delay of approximately one second before the system is activated. This delay feature prevents inadvertent activation of the rudder limiter in turbulent air. A light located in the annunciator panel, triggered by a magnetic sensor located next to the rudder limiter cam acts as a visual indication of when the rudder limiter is engaged.

b. Two points need to be emphasized regarding the operation of the rudder limiter. First, if a left rudder deflection of greater than 11.5° exists before the stall warning is active with a throttle setting greater than 12 in. of Hg, the cam cannot engage. In addition, if a left rudder deflection of greater than 12° exists while the stall warning is active and before the throttle setting is greater than 12 in. of Hg, then the cam cannot engage when the throttle is advanced beyond 12 in. of Hg. Second, if the rudder limiter is activated and pressure is applied to the left rudder pedal so that the rudder limiter cam is engaged and then the conditions which caused the rudder limiter to activate cease to exist so that the rudder limiter action is no longer needed, then the pressure on the left rudder pedal must be released in order for the rudder limiter cam to disengage. In either of these two conditions the cam actuation does not override the rudder input. It should also be noted that should the manifold pressure gauge itself or the stall warning horn become inoperative, the rudder limiter will still be functional provided that the stall warning vane is still operative.

03-5. TRIM SYSTEM (See Chapter 27)

a. The airplane is equipped with a two-axes trimming system. The elevator trim tab is located on the right side of the elevator and the aileron trim tab is on the aileron of the right wing. The aileron and elevator trim tabs are electrically controlled by two switches in the trim assembly panel above the engine controls (basic and Avidyne option) or by the A/P Trim switch to the right of the master switches in the overhead (Garmin G1000 option). The two trim motors are protected by one-amp circuit breakers located in the circuit breaker assembly panel.

03-6. WING FLAPS (See Chapter 27)

a. The airplane is equipped with Fowler flaps. During flap extension, the flaps move out from the trailing edge of the wing which increases both the camber and surface area of the wing. The flaps are powered by a 32-watt motor located under the front passenger's seat and protected by a 10-amp circuit breaker located in the circuit breaker panel.

03-7. DOOR SEAL SYSTEM (See Chapter 52)

a. The airplane is equipped with a pneumatic door seal system that limits air leakage and improves soundproofing. An inflatable gasket around each main door expands when the door seal system is turned on. An electrical motor near the pilot's rudder pedals operates the system, which maintains a differential pressure of 12 to 15 psi. The system is activated by a switch in the rocker switch panel labeled "Door Seal" and is protected by a five-amp circuit breaker. The door seal system should be turned on before takeoff and turned off after landing. However, a limit switch will turn off the pump when the door locking mechanism is disengaged. Similarly, if the door locking mechanism is not properly engaged, the pneumatic door seal system will not operate.

03-8. LANDING GEAR (See Chapter 32)

a. The airplane has tricycle landing gear with the two main wheels located behind the center of gravity (CG) and a nose wheel forward of the CG point. The main gear is made from high quality rod steel that has been gun-drilled. The main gear is attached to a tubular steel gearbox that is bolted to the bottom of the fuselage, just aft of the wing saddle. There are 15x6.00-6 tires (tire width and rim diameter in inches) which are inflated to 55 psi and mounted to the gear with Cleveland disk brakes. Composite wheel fairings are mounted over each main gear tire to reduce drag.

b. The nose gear has a nitrogen and oil filled oleo type strut that is bolted to the engine mount and serves as a shock absorber. The landing or vertical impact is absorbed by forcing oil through orifices in the piston and an internal plug or barrier. A rotation key or vane working within an oil filled pocket contains rotational movements (shimmy dampening). Both of these movements, vertical and rotational, are fully contained within the main cylinder body and under normal usage will require little maintenance.

03-9. ENGINE AND PROPELLER (See Chapter 61)

a. The airplane engine is a Teledyne Continental Motors' Aircraft Engine Model IO-550-N. It is a horizontally opposed, six-cylinder, fuel injected, air-cooled engine that uses a high-pressure wet-type oil sump for lubrication. There is also a full flow, spin-on, disposable oil filter. The engine has top air induction, an engine mounted throttle body, and a bottom exhaust system. On the front of the engine, accessories include a hydraulically operated propeller governor, a gear driven alternator, and a belt driven alternator. Rear engine accessories include a starter, gear driven oil pump, gear driven fuel pump, and dual gear driven magnetos. The airplane is equipped with a Hartzell three-bladed constant speed propeller with a McCauley governor.

03-10. FUEL SYSTEM (See Chapter 28)

a. The fuel system has two tanks that gravity feed to a three-position (Left, Right, and Off) fuel selector valve located between the pilot and copilot seats. The fuel flows from the

selected tank to the auxiliary fuel pump and then to the strainer. From this point it goes to the engine-driven pump where, under pressure, it is sent to the throttle/mixture control unit and then on to the fuel manifold valve for distribution to the cylinders. Unused fuel from the continuous flow is returned to the selected fuel tank. A pressure gauge on the metered side of the fuel manifold valve measures system pressure and displays both the fuel pressure and the equivalent fuel flow reading on the same gauge.

03-11. ELECTRICAL SYSTEM (See Chapter 24)

a. The electrical system in this aircraft consists of two independent buses, which are referred to as the left bus and right bus. Two 60-amp (continuous output) alternators provide charging power for the batteries, as well as system power. The batteries will also provide additional power in the event of an over demand situation where the requirements on the system are greater than what can be provided by the alternator. The left and right buses in turn feed the avionics and essential buses.

b. Five current limiters protect the alternators and bus outputs. In addition, the left and right buses are physically isolated inside a covered area mounted to the firewall. Left and right bus controls, grounds, and outputs are routed through separate holes, connectors, and cable runs so any failure on one bus will not affect the operation of the other bus.

c. Control of the buses is via the master switch panel located on the lower left portion of the instrument panel. There is also a crosstie switch on this panel, which will restore power in the event of failure of the right or left systems. For example, if the alternator or some other component on the left side should fail, the crosstie switch will restore power to the electrical items on the left bus by connecting the left bus to the right bus.

d. As its name may suggest, power to the essential bus is never affected, provided power from at least one bus (left or right) is available. The essential bus is diode fed, i.e., current will only flow in one direction, from both the right bus and the left bus allowing the essential equipment to have two sources of power.

e. There is an alternator switch in the cockpit area that disconnects the alternator and stops the excitation. A red “Alt Out” light in the annunciator panel illuminates (Basic or Avidyne option) or an annunciation message displays on the PFD (Garmin G1000 option) if the alternator becomes inoperative. At 16 volts (S/N 42001 to 42500) or 32 volts (S/N 42501 and on) output, an overvoltage control will stop the excitation to the alternator. The airplane is equipped with a voltmeter that measures bus voltage and an ammeter that measures the charging or discharging of the battery.

03-12. ENVIRONMENTAL CONTROL SYSTEM (See Chapter 21)

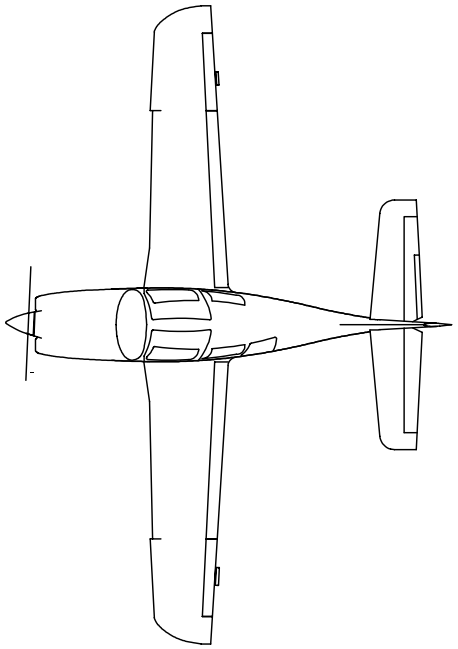
a. The environmental control system (ECS) incorporates the use of an air-to-air heat exchanger, ram intake air, and an electric fan to distribute heated and outside air to various outlets within the cabin. The ECS essentially consists of two subsystems, heated air and fresh air. Heated air is sent to a floor vent system and defroster, and fresh air is ducted through the eyeball vents. The system demand affects the volume of flow to a particular vent. As more vents are opened, the airflow to each vent is decreased.

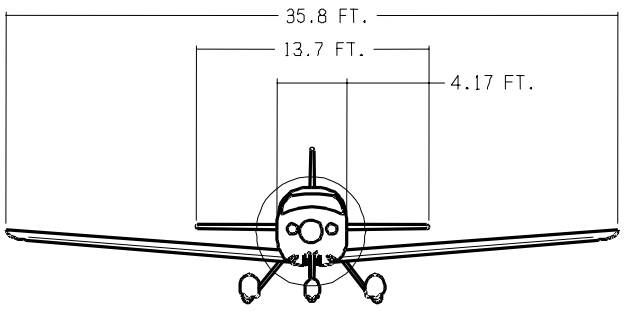
03-13. FLIGHT INSTRUMENTS (See Chapter 34)

a. The airplane is equipped with electrically-driven attitude and heading indicators. The pitot-static system measures pressure differences in ram air from the pitot tube and ambient air from the static air vent. Static air is also used as the pressure source for the vertical speed indicator and the altimeter.

03-14. THREE VIEW DRAWING OF THE AIRPLANE

Dimensions		Weight Data	Speed Data		Specifications
Wing Area	141.2 Sq. Ft.	Empty Weight 2400 ± Lbs.	Stall Speed	57 KIAS	Engine - 310HP IO-550-N Continental
Wing Span	35.8 Ft.		Maneuvering Speed	158 KIAS	
Length	25.2 Ft.	Gross Weight 3400 Lbs.	Cruising Speed	190 KCAS	Propeller - 78 in Hartzell/McCauley
Height	9 Ft.		Never Exceed Speed	235 KCAS	





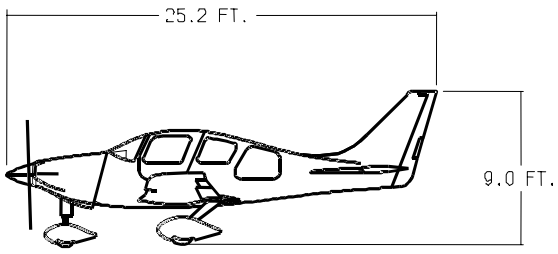


Figure 3 - 1 Cessna 350 (LC42-550FG)

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CHAPTER 4

AIRWORTHINESS LIMITATIONS

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MAINTENANCE MANUAL
LOG OF REVISIONS

The Log of Revisions pages are updated by Cessna each time revisions are issued. The Log of Revisions contains a list of all revisions made to the maintenance manual since its original issue. By comparing the Log of Revisions to the Record of Incorporated Revisions, the user of this manual can verify that all applicable reversions are included.

Revision	Date Issued	Affected pages	FAA Approved By
--	03/12/03	All	N/A
A	03/28/03	All	N/A
B	05/20/03	Revisions pg. 1, Ch. 4 LOEP pg. 1, 04-00-00 pg. 5	N/A
C	05/23/03	Revisions pg. 1, Ch. 4 LOEP pg. 1,	A.J. Paison 05/30/03
D	09/24/03	Revisions pg. 1, Ch. 4 LOEP pg. 1,	A.J. Paison 10/16/03
E	02/10/04	Revisions pg. 1, Ch. 4 LOEP pg. 1, 04-00-00 pg 5, 04-10-00 pg. 1	Jeffrey A. Morfitt 2-26-04
F	02/02/05	Revisions pg. 1, Ch. 4 LOEP pg. 1, 04-00-00 pg. 5	Jeffrey A. Morfitt 2-23-2005
G	06/27/06	Revisions pg. 1, Ch. 4 LOEP pg. 1. Revised header of all pages.	*J. Vincent Massey For Jeffrey A. Morfitt 7/14/2006*
H	10/10/06	Revisions pg. 1, Ch. 4 LOEP pg. 1, 04-00-00 pg 5	*Jeffrey A. Morfitt 11/02/2006*

Revision	Date Issued	Affected pages	FAA Approved By
I	12/07/07	All	*Luann Abrams* *12/07/07*
J	01/08/08	Revisions pgs. 1 and 2, Ch. 4 LOEP pg. 1, 04-00-00 pgs. 1, 4, and 5, 04-10-00 pg. 1.	*Jeffrey A. Morfitt* *ANM-100S* *01/29/08*

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Chapter 4

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04-1. GENERAL

a. The Airworthiness Limitations section (Chapter 4) is FAA approved and specifies maintenance required under §§ 43.16 and 91.403 of the Federal Aviation Regulations unless an alternative program has been FAA approved.

b. **Inspections –** This chapter contains mandatory maintenance tasks including inspections (Section 04-2) and life-limited item replacement requirements (Section 04-3). An airframe inspection in accordance with Section 04-2 is required every 3000 hours. This 3000-hour period is known as an Airframe Service Cycle (ASC). The table below identifies and describes the required maintenance tasks for each ASC. Note: These items are in addition to the items required to maintain the continued airworthiness of the airplane. After 25,200 TIS hours, the FAA requires a Major Airframe Inspection (MAI). Paragraph 04-4 contains the MAI items.

c. **Grace Period** – The required inspections listed in paragraph 04-1 must be performed within 100 hours of the specified interval or at the next annual inspection, whichever occurs first.

d. **Airframe Service Cycle Consistency** – If work is accomplished during the grace period, then the next inspection/part replacement is due at the time in service determined by adding the required interval to the BEGINNING of the grace period. For example, if the first ASC is accomplished during the grace period at 3099 hours TIS, then the next ASC is due at 6000 hours TIS (or within the grace period following 6000 hours). If work is accomplished early, then the next inspection/part replacement is due at the time in service determined by adding the required interval to the TIS at which the work was last accomplished. For example, if the first MAI was accomplished at 24,000 hours TIS, then the next MAI is due at 49,200 hours TIS (or within the grace period following 49,200 hours).

04-2. AIRFRAME SERVICE CYCLE INSPECTION (Every 3000 Hours)

INSPECTION AREA	MAINTENANCE TASK	CHAPTER NUMBER
Fuselage and Vertical Stabilizer		
Bonding of the two fuselage halves	**Visual inspection of the bonding seams:** Tap test seam area around fuselage halves in accordance with Section 20-26. This test detects unbonded areas by the change in audible response when the surface is systematically tapped.	53
Wing Attachment	Replace the wing attachment bolts and nuts with new bolts and nuts. There are a total of 30 bolts, 15 on each side of the wing saddle, that connect the fuselage to the wing as shown in Figure 57-2. The bolts vary in size from AN4-16 to AN4-20. When a bolt is removed, replace it with one that is the same size. The bolts must be replaced one at a time using a systematic process. It is recommended that the left forward bolt be removed and replaced first, and then continuing aft on the left side of the wing saddle. When the left side is complete, start on the right forward position and move aft.	53 & 57

CAUTION

The right aft wing bolt is reversed from the other bolts. That is, the nut is on top rather than the bolt head. This is the correct method of installation and ensures clearance of the bolt from the flap actuating system.

INSPECTION AREA	MAINTENANCE TASK	CHAPTER NUMBER
Wing Attachment (continued)	Use the following steps to remove and install the wing bolts. 1. Remove carpeting and footwell panel in rear seat area, exposing area between front and rear spars. 2. Remove flap assembly panel plate under the aft wing area in the fuselage saddle, exposing area aft of rear spar. 3. Remove inspection cover in the fuel assembly panel, which is located in the left forward saddle area, exposing area forward of main spar. 4. Replace all bolts and nuts. The bolts must be removed and replaced one at a time to prevent misalignment of the wing. See Chapter 57 for torque values. 5. When a bolt is removed, visually inspect the area adjacent to the bolt hole for evidence of damage, failure, and cracking. See Chapter 57 regarding bolt length and torque requirements. 6. The bolts and nuts cannot be reused for any purpose. All bolts and nuts that are removed must be permanently discarded or destroyed, regardless of the time in service.	
Cockpit Frame	Leave rear seat footwell exposed for this inspection. 1. Remove front and rear seats and front floor carpeting. 2. Remove carpeting from baggage floor and shelf. 3. Inspect the exposed area for signs of cracks, delamination, and excessive fatigue.	25
Airframe Paint	Perform a detailed inspection of the airplane's clear-coat and paint. All paint must conform to the latest FAA approved revision of Cessna Document SX512701. Contact the manufacturer for information if paint repair is necessary. Note: The airplane paint has a clear-coat cover that provides protection from UV rays. If the clear-coat is damaged, degradation of the paint and underlying surface is possible. The airframe paint does not have a specific life-limit. Its serviceability depends on the proper care of the airplane as discussed in Section 8 of the AFM/POH and the operating environment. For example, if the airplane is washed and waxed infrequently or the airframe is subject to a hail or dust storm, erosion and/or damage to the clear-coat is possible.	51
Transparencies	Check all transparencies for excessive crazing, UV yellowing, and evidence of cracking. Replace as required.	56
Doors	Remove left and right doors with door hinge blocks and inspect hinge block attachment holes for evidence of damage. Inspect attachment screws of dynamic hinge to the door for tightness	52
Door Openings	Check door openings, particularly the corner of the door openings,	52

INSPECTION AREA	MAINTENANCE TASK	CHAPTER NUMBER
	(including baggage compartment door) for evidence of cracking. Note: Sometimes the paint may crack around a door opening without the underlying fiberglass surface being affected. If paint cracks are observed, the painted surface must be removed by sanding to verify there are no cracks in the fiberglass.	
Wings		
Bonding of the Top and Bottom Wing	Visually inspect the bonding seams. Tap test seam area around leading and trailing edges in accordance with Section 20-27. The tap test detects unbonded areas by the change in audible response when the surface is systematically tapped.	57
Bonding of Wing Skin to Ribs and Spars	Tap test areas where the top and bottom of the skin are bonded to the spars and ribs in accordance with Section 20-27. If the skin is separated, a sponginess will be felt or a dullness in tone will be heard when tapped.	57
Flight Control Surfaces & Systems		
Wing Flaps	Visually inspect bonding seams. Tap test seam area around leading and trailing edges in accordance with Section 20-27. The tap test detects unbonded areas by the change in audible response when the surface is systematically tapped. Check for cracks on drive rib.	27
Ailerons	Visually inspect bonding seams. Tap test seam area around leading and trailing edges in accordance with Section 20-27. The tap test detects unbonded areas by the change in audible response when the surface is systematically tapped. Check for cracks and buckling in area of drive rib.	27
Rudder	Remove rudder and perform visual inspection of the bonding seams. Tap test seam area around leading and trailing edges in accordance with Section 20-27. The tap test detects unbonded areas by the change in audible response when the surface is systematically tapped. Check condition of hardware and cables for cracks, corrosion, and wear.	27
Wing Flap Actuator	Disassemble flap assembly and grease motor and gear train in accordance with Section 27-22. Check hardware for cracks, corrosion, and wear.	27
Aileron Push Tubes and Hardware.	Check for excessive wear and cracks. Inspect tubes for signs of corrosion. Check bellcranks and attachment hardware for signs of cracks and corrosion.	27
Elevator Push Tubes and Hardware.	Check for excessive wear and cracks. Inspect tubes for signs of corrosion. Check bellcranks and attachment hardware for signs of cracks and corrosion.	27
Elevator Trim Tab	Remove elevator trim tab and repack servo motor in accordance with Section 27-19 and 27-20. Inspect hardware for cracks, corrosion, and wear.	27
Elevator	Remove elevator and inspect piano hinge for excessive wear and corrosion.	27
Horizontal Stabilizer	Remove horizontal stabilizer and check stabilizer tube and bolts for cracks, corrosion, and wear. Replace as required.	27

INSPECTION AREA	MAINTENANCE TASK	CHAPTER NUMBER
Fuel System		
Boost Pump	Overhaul per Duke Inc. instructions included in overhaul kit, or replace the boost pump.	28
Landing Gear		
Teflon Bushings In Gear Box	Remove landing gear and check Teflon bushings in gear box for each main gear leg for condition and wear. See Chapter 32 for specific details about the landing gear.	32
Gear Box Attachment Bolts	Remove one attachment bolt from each side of the gear box and check for corrosion. If corrosion is noted, replace all attachment bolts in the gear box. If no corrosion is noted, replace bolts that were removed and check the torque on all bolts in the gear box in accordance with Section 32-4. See Chapter 32 for specific details about the landing gear.	32
Landing Gear Alignment	Once landing gear is reinstalled, check camber and toe-in. See Chapter 32 for specific details about the landing gear.	32

04-3. LIFE-LIMITED PARTS

The items listed below are to be replaced with new items at the times/intervals shown.

Item	Location	Replacement Time/Interval
Emergency Locator Transmitter (ELT) Battery	Aft of the baggage compartment hat rack – See Chapter 25 for more information.	Every 2 years (Artex 200), every 5 years (Artex ME406), or when the battery has been used for more than one hour or 50% of its life.
3-volt Lithium Battery in Avidyne Multi-Function Display (MFD)	On an internal processor board inside the MFD – See Chapter 34 for more information.	Every ten years or when the CMOS memory of the MFD fails to retain configuration data.
System – Dry Sealed Lead-Acid Type Battery (Two)	Just forward of the firewall on the Copilot's side (S/N 42001 to 42022) or on the Pilot's side (S/N 42023 and on).	Alternate replacement every two years – First replace the right battery, after two more years replace the left battery. Alternate replacement every two years. However, if the battery fails to hold a charge, it must be replaced.
No. 1 (Right) Alternator	Just rear of the propeller on the Copilot's side.	Overhaul or replace alternator every 2000 hours.
No. 2 (Left) Alternator	Just rear of the propeller on the Pilot's side.	Overhaul or replace alternator every 2000 hours. Check belt condition and tension every 100 hours or annually. Belt tension range is 13 to 17 lbs.-ft. measured as described in Chapter 24. Replace the belt whenever there is cracking, fraying, or signs of excessive wear, every 5 years, or 2000 hours whichever comes first.
Front and Rear Seat Restraint Devices	At front and rear seats.	Every 3000 hours or 10 years, whichever occurs first.
Wing Attachment Bolts	Wing saddle	Every 3000 hours
SpeedBrakes	Outboard top of wing, each side.	Every 1000 hours return SpeedBrake cartridges to Precise Flight Inc. for clutch lubrication and spring replacement.
SpeedBrakes	Outboard top of wing, each side.	Every 5000 hours return SpeedBrake cartridges to Precise Flight Inc. for drive assembly replacement.
Carbon Monoxide Detector	Mounted to the gusset on the firewall behind the flight instrument panel.	Every 7 calendar years return to CO Guardian LLC for CO Sensor replacement and calibration.
Airplane		25,200 flight hours

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04-4. MAJOR AIRFRAME INSPECTION CHECKLIST (Every 25,200 Hours)

Several pages will be added here that describe all the inspection items that must be performed on the Major Airframe Inspection. This checklist will be added to the manual as part of a revision level. There is currently a life limit on the airplane of 25,200 hours until Section 04-4 is completed and approved by the FAA.

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CHAPTER 5

TIME LIMITS & MAINTENANCE CHECKS

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Chapter 5

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05-1. GENERAL

a. This chapter sets forth recommended scheduled maintenance checks and component overhaul/replacement time periods. These requirements are in addition to those stipulated in Chapter 4. An inspection checklist is included which should be photocopied and utilized during each scheduled inspection. The checklist includes the minimum items recommended by the manufacturer to be inspected and/or serviced. Also included is a list of special inspections and service items, which must be complied with in order to ensure the continued safe operation of this airplane.

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05-2. TIME LIMITS

a. The information in Figure 5-1 identifies either component time limits and/or required special inspections. These inspections must be incorporated into a 100 hour or annual inspection as applicable. If it is anticipated that the component time limit and/or special inspection will occur before the next applicable required inspection, it must be incorporated in the current required inspection.

Time Limits & Special Inspection Items

Special Inspection Item	Time Period	Chapter
1. Remove, inspect, clean, and reinstall fuel nozzles. If ultrasonic cleaning is available, it is the preferred method to use. **Do not** insert anything into fuel nozzle orifice. If cleaning will not remove internal deposits, fuel nozzle must be replaced.	At 1st 100 hr. insp. and every 300 hours thereafter	28
2. Emergency Locator Transmitter (ELT) Battery	Every 2 years (Artex 200) or every 5 years (Artex ME406)	25
3. Altimeter and Static System Inspection – Refer to AC 43-6B, Appendix E of 14 CFR Part 43, and 14 CFR Part 91.	Every 24 calendar months	34
4. Automatic Pressure Altitude encoding System and Transponder inspection and test – Refer to AC 43-6B, Appendix F of 14 CFR Part 43, and 14 CFR Part 91.	Every 24 calendar months	34
5. Perform a compass swing.	Every 24 calendar months	34
6. Repack new wheel bearings (nose and main landing gear) after first 100 hours, then every 400 hours thereafter. This should be done more frequently if experience indicates the need based on the type of operations conducted.	Every 400 hours	32
7. Inspect fuel selector valve.	At 1st 500 hour insp. and every 1000 hours thereafter	28
8. Check fuel gauge accuracy.	Every 500 hours	28
9. Disassemble magnetos and inspect in accordance with TCM Manual X42002-1.	Every 500 hours	71
10. Overhaul magnetos in accordance with TCM Manual X42002-1.	Every 4 years or at engine overhaul, whichever occurs first.	71
11. Remove and inspect oil pressure relief valve in accordance with the TCM Overhaul Manual. Check plunger for scraping and nicks. Check the conical face for roughing. Check limits in accordance with TCM specifications. Replace or reface parts as required. See Chapter 18 of the TCM Overhaul Manual.	Every 500 hours	71
12. Inspect exhaust tone cone.	Every 500 hours	78
13. Check and record the control deflections on the ailerons, flaps, elevator, and rudder.	Every 1000 hours	27
14. Inspect aileron control rods for cracking.	Every 500 hours or biennially	27

Time Limits & Special Inspection Items

Special Inspection Item	Time Period	Chapter
15. Inspect aileron linear bearing for foreign object debris. Ensure there is no debris or adhesive attached to the aileron control rod, bearing housing, bearing race, or ball bearings.	Every 100 hours or annually	27
16. Inspect aileron trim tab friction device for proper sliding friction. Assure device is free from oils, grease, dirt, etc.	Every 500 hours or biennially	27
17. Inspect and service elevator trim motor.	Every 3000 hours	27
18. Verify each Garmin GSM 85 servo mount clutch setting if Garmin GFC 700 autopilot system is installed.	Every 500 hours	22
19. Inspect the GSA 81 servo actuators, GSM 85 servo mounts, and GTA 82 trim adapter if Garmin GFC 700 autopilot system is installed.	Every 1000 hours	22
20. Inspect the alternators to ensure that the brushes are not more than 75% worn and the bearings operate smoothly. If the brushes are more than 75% worn or the bearings do not operate smoothly, then replace the alternator. For alternator No. 1 (right) ensure the clutch resistance is still 140 in.-lbs of torque or greater.	Every 1000 hours	24
21. Remove rudder and check hinges.	Every 1000 hours	27
22. Check resistance of all static wicks between the wick base and the tip. The test requires use of a megohmmeter, sometimes referred to as a Megger. A properly functioning static wick should indicate 1 to 100 megohms of resistance.	Every 1000 hours	20
23. On each main gear, remove the wheel pants, wheels, axles, strut fairings, and underside gearbox access panels. Perform detailed inspection of the landing gear legs. Look for evidence of metal fatigue, corrosion, cracks, and deformation. Pay particular attention to the area where the strut is inserted into the gearbox.	Every 1000 hours	32
24. Service nose strut.	Every 200 hours or annually	12
25. Remove front and rear seats, carpeting, and interior side panels. Inspect the four SAFE strips in the lower fuselage interior for wear, tearing, adhesion, and overall general integrity.	Every 1000 hours	20
26. Conduct exhaust system pressure test.	Every 1000 hours or every 5 years, whichever occurs first	78
27. Inspect exterior emergency door release system. Activate exterior hinge release mechanism to ensure proper operation. Inspect hinge wires for corrosion, kinks, and wear. Replace as required. Move cable up and down to ensure ease of movement. Inspect overhead cabling for chaffing, binding, and frayed cables. To rearm the system, reinsert door hinge wires, and pull both the actuating cables back to their stops. Store the few inches of excess cable in a manner that prevents kinking or binding. See Chapter 52.	Every 1000 hours or every 5 years whichever occurs first	52
28. Inspect power grid.	Every 1000 hours	24
29. Clean door seal system pump filter.	Every 1000 hours	52
30. Purge the oxygen system. Inspect the oxygen lines and fitting for cracks, leaks, and damage. Overhaul the regulator/valve assembly, replace O-rings, and verify the regulator pressure setting. Replace the flexible oxygen lines on the breathing stations. Overhaul the A4 flowmeters.	Every 5 years	35

Time Limits & Special Inspection Items		
Special Inspection Item	**Time Period**	**Chapter**
31. Oxygen Cylinders – Hydrostatically test 5 years after date stamped on the cylinder and every 5 years thereafter.	Every 5 years	35
32. Hard Landing Inspection – In the event of a hard landing, the aircraft should be inspected in accordance with section 32-10.	After Hard Landing	32
33. Lightning Strike – In the event of a lightning strike follow inspection and repair procedures in Chapter 20.	After Lightning Strike	20
34. VCS Compressor – Inspect belt and adjust as required to 50 to 70 lbs. tension.	1st 25 hrs and every 100 hrs thereafter	21
35. Garmin GDC 74A – The static system and altimeter must be verified in accordance with 14 CFR §91.411 and Appendix E of 14 CFR Part 43. Test procedure per the latest revision of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.	Every 2 years	34
RECOMMENDED REPLACEMENTS		
36. Replace oxygen cannulas and/or non microphone oxygen masks.	Every 200 hours of use or every 3 years, whichever occurs first.	35
37. Replace microphone oxygen masks.	Every 500 hours of use	35
38. Replace the O-rings in the CPC connector assembly identified on the breathing stations.	Every 3 years	35
39. Replace or rebuild the gyroscopic attitude indicator (artificial horizon)	Every 1200 hours	34
40. Replace or rebuild the gyroscopic directional indicator (DG)	Every 1200 hours	34
41. Replace or rebuild the turn coordinator	Every 1800 hours	34
42. Replace clock battery. Note: The Avidyne FlightMax Multi-Function Display (MFD) and the Garmin G1000 MFD include a clock function. If the MFD is installed in the aircraft there will be no separate clock requiring battery replacement.	Every 2 years	31
43. Overhaul the propeller. Note: The propeller must be overhauled after 2400 hours of time in service by an authorized propeller overhaul facility. The propeller does not have a specific life limit, however, after extended use its serviceability can be diminished to the point that it cannot be overhauled, and it must be replaced.	Every 2400 hours	61
44. Overhaul the engine. Note: The IO-550-N engine requires a complete overhaul every 2000 hours of time in service by an authorized engine repair facility. There are also overhaul limitations on the magnetos, which is discussed in the Time Limits section of Chapter 5. Inspect the engine mount bushings for corrosion; replace the bushing(s) if corroded.	Every 2000 hours	71
45. Overhaul the propeller governor. Note: The propeller governor requires a complete overhaul every 2000 hours of time in service by an authorized propeller repair facility.	Every 2000 hours	61

Figure 5- 1 Time Limits & Special Inspection Items

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05-3. INSPECTION REQUIREMENTS

a. **Annual Inspection** – Part 91.409 of the Federal Aviation Regulations (FAR's) requires that an airplane must receive an annual inspection each 12 calendar months in accordance with part 43 of the FAR's and be approved for return to service by a person authorized by the requirements of 43.7. For the purpose of the FAR's, 12 calendar months extend from the particular day within the month the inspection is performed to midnight of the last day of the same month, one year later.

b. **100-Hour Inspection –** Part 91.409 of the FAR's require that if an airplane is used to carry passengers for hire or give flight instruction for hire, the airplane must receive an inspection after 100 hours of time in service as specified in part 43 of the FAR's. The 100 hour limitation may be exceeded by not more than 10 hours while en route to reach a place where the inspection can be done. The excess time used to reach a place where the inspection can be done must be included in computing the next 100 hours of time in service.

c. **50 Hour Inspection –** It is recommended that at each 50 hours of time in service the airplane receive a 50-Hour Inspection.

d. **Unscheduled Inspections** – If the limitations of the engine or airframe are exceeded such as a hard or overweight landing, the likely affected components and systems must be inspected as indicated in this chapter. See Chapter 32 for hard landing inspection details. If a situation occurs which is not covered in this chapter, consult Cessna for technical support and recommendations.

e. **Progressive Inspection** – At some point in the future the outline for a progressive maintenance program will be published and included as a revision to the Cessna Maintenance Manual.

f. **Component Inspections** – Time limits and special inspection requirements may exist for equipment manufactured by other suppliers and installed on Cessna Aircraft. These inspections and recommendations are determined by that manufacturer and should be followed to ensure continued airworthiness of that equipment and the aircraft as a whole.

05-4. Cessna 350 (LC42-550FG) Inspection Checklists

a. Pages 2 through 13 of this section contain the inspection items for the required and recommended inspections described in paragraphs 05-1.a through 05-1.c above. Prior to each 50 hour, 100 hour, or annual inspection, an engine run-up should be performed using the checklist in Figure 5- 2. Depending on the specific types of operations, the owner or operator may need to modify and/or increase the frequency of some inspection and/or servicing items.

NOTE

All inspections and maintenance procedures must be in accordance with applicable Federal Aviation Regulations. Please refer to FAR parts 91.403, 91.207, 91.409, 91.411, and 91.413

Cessna 350 Inspection Checklist

Model LC42-550FG

Instructions for Continued Airworthiness

Reg. number	Serial number:	Owner's name:	
Date started	Date completed	Work Order #	**Inspection** 50-hr □ 100-hr □ Annual □
Hobbs	Eng TSN	Prop TSN	Mechanic's Name
TT	Eng TSO	Prop TSO	

ELT battery due:	
Left Battery due:	
Right Battery due:	
Altimeter/Transponder test due:	
O2 bottle hydro. test due:	
Fire extinguisher insp. due:	
Magneto 500-hr insp. due:	
Flexible hose rplcmt. due:	
Misc.:	
Misc.:	
Oil sample? Yes □ No □	
Comments	

Differential compression check: Taken: Hot □ Cold □			
6 -	/80	5 -	/80
4 -	/80	3 -	/80
2-	/80	1 -	/80

Discrepancies Noted During Runup
(Additional space available on last page)

□See last page for additional comments.

RUNUP CHECKS

□ - check (4) if okay. For all others, write actual indication.

System	Pre-inspection	Post-inspection
Boost/primer pump	□	□
High/Lo boost pump cut-off	□	□
Starter	□	□
Oil pressure (cool)	psi	psi
Charging system, Left (1200 RPM)	Amps volts	Amps volts
Charging system, Right (1200 RPM)	Amps volts	Amps volts
Left brake	□	□
Right brake	□	□
Parking brake	□	□
Comm/Nav	□	□
Ignition ground test (idle)	□	□
L. Mag drop (1700 RPM)	RPM Drop	RPM Drop
R. Mag drop (1700 RPM)	RPM Drop	RPM Drop
Prop response (1200 RPM)	□	□
Oil temperature	°F	°F
CAUTION: Ensure CHT and oil temp. are in the green range.		
Static RPM	RPM	RPM
Fuel flow at static RPM	gph	gph
Induct. Heat (1700 RPM)	□	□
Defrost	□	□
Cabin heat	□	□
Oil pressure	psi	psi
Idle RPM	RPM	RPM
CAUTION: Allow engine to cool to 300°F (CHT) before shutdown.		
All annunciator lights are off	□	□
Check for fuel odors in cabin	□	□
Check for fuel stains on underside of aircraft	□	□
Check fuel valve *off* function	□	□
Idle cutoff RPM rise	RPM Rise	RPM Rise

Figure 5- 2 Inspection and Run-up Checklist

Cessna 350 Model LC42-550FG Instructions for Continued Airworthiness Inspection and/or Required Maintenance Checklist	Each 50 hours	Each 100 hours	Annually
POST-RUNUP CHECKS			
1. **Engine cowling** – Remove engine cowling.	❑	❑	❑
2. **Engine oil** – Drain within 30 minutes of engine shutdown. Oil sample should be collected within 15 minutes of engine shutdown. Replenish with appropriate grade of oil. This step may be deleted from the annual inspection, if desired, to keep oil changes on regular 50-hour intervals. However, regulations require that the oil filter still be cut open for inspection.	❑	❑	❑
3. **Engine** – Conduct engine differential compression check. Record on checklist coversheet. (Refer to latest TCM Service Bulletin for permissible drop and differential).		❑	❑
4. **Oil filter** – Replace and torque 16 to 18 ft-lbs. Cut open and inspect old oil filter.		❑	❑
5. **Exterior lights** – Check operation of taxi and landing lights, position lights, and strobe lights.	❑	❑	❑
6. **Pitot heater** – Check operation. **WARNING: Heater gets extremely hot!** Check within 60 seconds after switching on. Do not operate longer than 2 minutes on the ground.		❑	❑
7. **Stall warning system** – Check operation.		❑	❑
8. **Auxiliary lights** – Check operation of courtesy entry lights, entry step lights, and baggage compartment light by opening one door at a time. Leave one door open to check operation of 10-minute lights-off timer.		❑	❑
9. **Interior lights** – Check operation of instrument flood bar, panel lights, instrument lights, overhead lights, electro luminescent panels, and all dimming rheostats.		❑	❑
10. **Annunciator lights** – Check operation.		❑	❑
11. **Aural warning** – Check operation.		❑	❑
12. **Flight controls** – Check for smooth operation of all flight controls with flaps in retracted and extended positions. Leave flaps in extended position when checks are completed.		❑	❑
13. **Door seals** – Check operation of door seal air pump and inspect for air leaks in seals. (Note: Both doors must be latched for system to operate.)		❑	❑
14. **Rudder limiter system** – Check operation. See Chapter 34 of the AFM/POH.	❑	❑	❑
15. **ECS ventilation fan** – Check operation.		❑	❑
16. **Trim actuators** – Check operation of all actuators and accuracy of position indicator LED's. Check switches on trim panel (basic and Avidyne), right of the master switches in the overhead (Garmin G1000), and on pilot's control stick.		❑	❑
17. **Emergency Locator Transmitter (ELT)** – Check operation of remote switch. **NOTE: ELT may be tested only within the first 5 minutes after each hour and should be limited to 3 aural tone sweeps.**		❑	❑
18. **Flight controls** – Operate controls through full range of motion feeling for binding, ratcheting and sloppiness, and listening for unusual noises.		❑	❑
19. **Brakes** – Check for sponginess.	❑	❑	❑

Cessna 350 Model LC42-550FG **Instructions for Continued Airworthiness** **Inspection and/or Required Maintenance Checklist**	**Each 50 hours**	**Each 100 hours**	**Annually**
20. **Aircraft batteries** –Fully charge, clean battery exterior surfaces, and clean battery cables. Check battery box for evidence of leakage.		❑	❑
21. **Access panels, fairings, seats, carpets, covers, side panels, overhead panel (100 hour and annual inspection), and spinner** – Remove for complete access for inspection. Check for missing or stripped screws. Do not remove nose wheel fairing at this point. See paragraph 05-5 for listing of the various access covers in the airplane. **WARNING: Do not remove fuel tank cover plates.**		❑	❑
AIRCRAFT RECORDS			
22. **Aircraft logbooks** – Determine total times, times since overhaul and times since last required or recommended maintenance checks and record on Inspection Coversheet.		❑	❑
23. **Inoperative items** – If any authorized inoperative items have been listed per FAR 91.213, inspect and maintain per FAR 91.405. Verify placard is installed per FAR 43.11. See the equipment list of required items in Section 6, Appendix A, of the AFM/POH.		❑	❑
24. **Airworthiness Directives (AD's) and Service Bulletins** – Check for new and recurring AD's which may need to be complied with during this inspection. Also, check for applicable manufacturer's Service Bulletins.		❑	❑
25. **Aircraft records** – Check for presence and condition of aircraft federal registration form and airworthiness certificate.		❑	❑
26. **Approved Flight Manual / Pilot's Operating Handbook (AFM/POH)** – Verify that all revisions have been installed and that the Equipment List and Weight and Balance forms are current and correct.		❑	❑
ENGINE AND PROPELLER (See Chapters 30, 61 and 71 for Specific Details)			
27. **Cleaning** – Clean engine. Protect magneto vents and cabin fresh air intakes.		❑	❑
28. **Engine** – Inspect for obvious signs of damage, oil or fuel leakage, loose components, and chafing or deterioration of hoses, lines, and wires.	❑	❑	❑
29. **Cylinders, rocker box covers, and pushrod tubes** – Inspect for cracks, broken fins, oil, fuel or exhaust leakage, and discoloration of cylinder head. Check hardware for signs of loss of torque.		❑	❑
30. **Crankcase and sump** – Inspect for cracks, oil leakage, corrosion and signs of loss of torque of hardware. Inspect engine mount supports for condition.		❑	❑
31. **Induction system** – Inspect the induction tubes for security and condition and loose or leaking couplers. Inspect manifold pressure transducer for security of installation and condition of wires. Check operation and sealing of alternate air/heated air door.		❑	❑
32. **Induction air filter** – Inspect for cleanliness and condition of sealing surfaces. Service per instructions in Chapter 12.	❑	❑	❑
33. **Exhaust heater muff** – Remove heat muff sleeve. Conduct pressure check of heat muff/muffler, and inspect for security of studs.		❑	❑

Cessna 350 Model LC42-550FG Instructions for Continued Airworthiness Inspection and/or Required Maintenance Checklist	Each 50 hours	Each 100 hours	Annually
34. **Exhaust system** – Inspect entire system for leaks, cracks, and security. Inspect muffler internal baffles for security.	❑	❑	❑
35. **Fuel metering unit** – Inspect for security and presence of fuel leakage. Check operation of throttle and mixture controls. Ensure levers hit stops before cables reach end of travel. Lube throttle lever with LPS-2.		❑	❑
36. **Fuel manifold valve and distribution lines** – Inspect for security, presence of factory seal installed on safety wire, and evidence of fuel leakage. Inspect distribution lines for security, cracks, and signs of leakage.		❑	❑
37. **Engine-driven fuel pump** – Inspect for security, general condition, and evidence of fuel or oil leakage.		❑	❑
38. **Fuel hoses** – Inspect all flexible fuel hoses for routing, chafing, security, and signs of leakage.		❑	❑
39. **Magnetos** – Inspect for security and condition. Check timing to engine and record: L____°BTDC, R____°BTDC. Timing must be within 22°±1°BTDC. If either magneto must be adjusted, check logbooks for previous magneto adjustments. If the same magneto has already been "bumped up" twice without being removed to check internal timing, that magneto must be removed for internal timing check.		❑	❑
40. **Spark plugs** – Inspect firing end for condition. Clean, gap, reinspect, and test spark plugs following appropriate plug manufacturer's service manual. Reinstall with **new** gaskets and torque to specifications.		❑	❑
41. **Ignition harness** – Inspect for routing, broken shielding, chafing, and security. Inspect spark plug ends for condition and evidence of electrical arcing.		❑	❑
42. **Left Alternator** – Inspect for security and general condition. Inspect electrical connections for security. Inspect belt for tension, fraying, and dryrot. Inspect pulleys for nicks, scratches, warpage, and alignment.		❑	❑
43. **Right Alternator** – Inspect for security, general condition and signs of oil leakage. Inspect electrical connections for security. Check air blast tube for obstructions. Any evidence of alternator malfunction requires removal to conduct drive gear hub slippage inspection.		❑	❑
44. **Electrical cables, wires, and connectors** – Inspect for security, routing, chafing, signs of arcing, and proper termination.		❑	❑
45. **Starter and starter solenoid** – Inspect for security, general condition, and signs of oil leakage.		❑	❑
46. **Oil cooler** – Inspect for security of mounting, cracks, straightness of fins, and evidence of oil leakage around base and fin area. Clean debris from fins. Inspect condition and security of oil pressure switch and temperature bulb and associated wiring.		❑	❑
47. **Engine accessory case** – Inspect accessory case and any additional accessories for security, condition, and evidence of leakage or loss of torque of hardware.		❑	❑
48. **Crankshaft** – Inspect for evidence of oil leakage, corrosion, pitting, cracks, discoloration, or signs of overstressing.		❑	❑
49. **Propeller governor** – Inspect for security, evidence of oil leakage, and tightness of lever on governor shaft. Check operation of propeller control cable. Ensure high-RPM stop is hit before cable reaches end of travel.		❑	❑
50. **Spinner and spinner bulkhead** – Inspect for cracks, security to propeller, and security of any dynamic balance weights. Clean inside of spinner.		❑	❑

Cessna 350 Model LC42-550FG **Instructions for Continued Airworthiness** **Inspection and/or Required Maintenance Checklist**	**Each 50 hours**	**Each 100 hours**	**Annually**
51. **Propeller hub** – Inspect for cracks, corrosion, leakage of oil, excessive loss of grease, security of static balance weights, and security of mounting to crankshaft. Retorque **all** mounting nuts if loss of torque is suspected on any nut in accordance with section 61-4. Service with appropriate grease.		❑	❑
52. **Propeller blades** – Inspect for play, cracks, nicks, dents, pitting, corrosion, and leading edge erosion. Dress propeller and paint back of blade. In operational areas of high humidity, wipe the metal surface with an oily rag.	❑	❑	❑
53. **Propeller Heat/De-Ice System** – Inspect the propeller de-ice boots, slip ring, brushes and brush block, control switch/annunciator, control module, and wiring.		❑	❑
54. **Cooling baffles and seals** – Inspect for cracks, excessive gaps, and security of mounting to engine. Check intercylinder baffles for security.		❑	❑
55. **Engine cowls** – Inspect for cracks, chafing, heat damage, delamination, evidence of exhaust leakage, condition of fastening system, presence of oil filler door placard, and condition of paint.		❑	❑
56. **ECS air inlet scoop** – Inspect for obstruction of inlets and condition of scoop and screen.		❑	❑
57. **Firewall** – Inspect for cracks, buckling, unsealed holes, and other signs of damage. Although the firewall is not structural, problems here may indicate damage to the composite bulkhead underneath and should be closely checked out. Inspect all firewall-mounted accessories for security of mounting.		❑	❑
58. **Engine mount/NLG support** – Raise NLG off of the ground and inspect for cracks, corrosion, loss of torque of mounting hardware, chafing by cables, wires, hoses, etc., and make sure that any flexing item is secured to the mount.		❑	❑
59. **Engine isolation mounts** – Inspect for general condition signs of loss of torque of engine mount bolts. If necessary, torque bolts in accordance with section 71-4. Check safety wiring and security.		❑	❑
60. **Drain and vents lines** – Inspect for security, condition, and obstructions.		❑	❑
61. **Battery box** – Inspect for security of mounting and condition.		❑	❑
62. **Brake fluid reservoir** – Inspect for condition, security, and fluid level. Service as necessary.	❑	❑	❑
63. **Bonding ground straps** – Inspect for condition.		❑	❑
64. **Foreign Objects** – Check engine compartment for foreign objects.	❑	❑	❑
FUSELAGE EXTERIOR & EMPENNAGE (See Chapter 53 and 55 for Specific Details)			
65. **Fuselage exterior surface** – Inspect for obvious signs of damage, including cracks, holes, buckling, and rippling. Check condition of paint and cleanliness. Check drain holes for obstructions.	❑	❑	❑
66. **Exterior placards** – Inspect for presence and condition. See chapter 11, *Placards and Markings*, for required placards.			❑
67. **Windows** – Inspect for cleanliness, condition, and sealing.	❑	❑	❑

Cessna 350 Model LC42-550FG Instructions for Continued Airworthiness Inspection and/or Required Maintenance Checklist	Each 50 hours	Each 100 hours	Annually
68. **Cabin doors** – Inspect for security and fit. Inspect skin, hinges, gas struts, latching mechanisms, door seals, and upholstery. Lubricate hinges and all moving parts with a quality petroleum or silicone-based oil. Check that door seals inflate properly.		❑	❑
69. **Baggage door** – Inspect for security and fit. Inspect door skin, upholstery, seal, hinge, and latching mechanism. Inspect door strut for condition and security. Lubricate hinge and all moving parts.		❑	❑
70. **Entry assist step** – Inspect for security and condition.		❑	❑
71. **Pitot port** – Inspect for obstruction of pitot, signs of damage, which may affect proper airflow, and security of attachment. Remove the pitot port before attempting to clear any obstructions, making sure the port is not enlarged or damaged in any way. Test for proper operation after reinstallation.		❑	❑
72. **Static Port Drain** – Check static port drain for evidence of water. Remove cap from drain line and allow any water within the system to drain. Leak check after reinstallation. **Caution: Do not apply compressed air to the system, since this will result in damage to the static air flight instruments.**			❑
73. **Antennas** – Inspect for security and condition.		❑	❑
74. **Aircraft identification plate** – Inspect for security and legibility.			❑
75. **Fuselage skin** – Perform generalized inspection of fuselage for visible damage and evidence of latent damage.		❑	❑
76. **Exterior emergency door release** – Inspect exterior emergency door release system. Inspect hinge wires for corrosion, kinks, and wear. Replace as required. Inspect overhead cabling for chaffing, binding, and frayed cables. Store the few inches of excess cable in a manner that prevents kinking or binding. Inspect attachment screws of dynamic door hinge to the door for tightness. See Chapter 52.		❑	❑
77. **Vertical stabilizer** – Perform generalized inspection of vertical stabilizer for visible damage and evidence of latent damage. Inspect hinge attach points for security and condition. Ensure leading edge tape is secure.		❑	❑
78. **Rudder** – Inspect for signs of damage, looseness, play in bearings, and condition of hinge attachments and rudder cable horns. Check security of static balance weight. Check for obstruction of drain holes. Inspect bonding strap and static wicks for condition. Lubricate hinges and cable-attach points. Inspect rudder crossover cable for proper cable tension. Inspect rudder crossover cable for wear, alignment, chafing, binding, and fraying especially where the cable rounds a pulley. Remove and inspect rudder crossover cable for wear and fraying at every 1000 hrs and replace as required. Ensure rudder makes full contact with left and right rudder stops		❑	❑
79. **Horizontal stabilizer** – Perform generalized inspection of horizontal stabilizer for visible damage and evidence of latent damage. Inspect hinge attach points for security and condition. Ensure leading edge tape is secure.		❑	❑
80. **Elevators** – Inspect for signs of surface damage, looseness or play in bearings, and condition of hinges and elevator horn. Check security of static balance weights. Inspect bonding straps and static wicks for condition. Check for obstruction of drain holes. Lubricate piano hinges **only** in accordance with section 27-18. Ensure elevator makes full contact with up and down elevator stops.		❑	❑

Cessna 350 Model LC42-550FG **Instructions for Continued Airworthiness** **Inspection and/or Required Maintenance Checklist**	**Each 50 hours**	**Each 100 hours**	**Annually**
81. **Elevator trim tab and actuator** – Inspect for condition, corrosion, and security. If the drive rods show any signs of corrosion, replace with new rods. Lubricate hinge and clevis bolt in accordance with section 27-19. Check trim servo for smoothness of operation; free from binding. Measure and record total trim tab play: _____inches. **Caution: Use special care to ensure trim tab friction devices are not lubricated.**		❑	❑
82. **Rudder Trim** – Check for proper security.		❑	❑
WINGS (See Chapter 57 for Specific Details)			
83. **Wing Saddle Area** – Inspect wing spar for cracks, debonding, and attachment to airframe. Check visible attaching hardware for loss of torque. If necessary, torque hardware in accordance with section 57-4. Inspect aileron bellcranks for cracks and corrosion.		❑	❑
84. **Fuel Tanks** – Check that the airplane's fuel filler neck is grounded. Run a continuity check between each fuel filler neck and one of the exhaust stacks. Fuel cap resistance must be within 0 to 500 ohm. Also, check continuity of all static wicks or static wick attachment points to test the integrity of the lightning protection system.		❑	❑
85. **Wing skins and exterior** – Inspect for obvious signs of damage, including cracks, holes, buckling, and rippling. Check condition of paint and placards. Check drain holes for obstructions. Check fuel tank vents for obstructions. Ensure leading edge tape is secure.		❑	❑
86. **Landing and taxi light assembly** – Inspect for cracks, security of mounting, and cleanliness and condition of lens cover. Operate taxi and landing lights in a dark area and ensure that both lights are properly aimed. If lights are not properly aimed, adjust as required. Refer to Chapter 33 for detailed instructions.		❑	❑
87. **Aileron and flap attach points** – Inspect for security of attachment to wing.		❑	❑
88. **Aileron (left) servo tab and servo tab clevis –** Inspect for damage, looseness, or play in attach bearings and Belleville washers, and condition of rod end attachment. Check security of static balance weights. Inspect bonding straps and static wicks for condition. Check for obstruction of drain holes. Inspect servo tab for proper operation and condition of push-pull rods, bearings, and hinge. If the rods show any signs of corrosion, replace with new rods. Lubricate only servo tab hinge and clevis bolt in accordance with section 27-13. Measure and record total servo tab play: _____inches. See Figure 27-1.		❑	❑
89. **Aileron, right** – Inspect for damage, looseness, or play in attach bearings and Belleville washers, and condition of rod end attachment. Check security of static balance weights. Inspect bonding straps and static wicks for condition. Check for obstruction of drain holes. Inspect trim tab and actuator for condition and security. Lubricate only trim tab hinge and clevis bolt in accordance with section 27-11. Record total trim tab play: _____ inches. **Caution: Use special care to ensure trim tab friction devices are not lubricated.**		❑	❑
90. **Aileron trim tab clevis** – Inspect trim clevis and related hardware for evidence of corrosion and pitting. Check for proper security.		❑	❑
91. **Aileron deflections** – Ensure that both ailerons seat properly on their up and down stops.		❑	❑

Cessna 350 Model LC42-550FG Instructions for Continued Airworthiness Inspection and/or Required Maintenance Checklist	Each 50 hours	Each 100 hours	Annually
92. **Flaps** – Inspect skins for condition and signs of debonding. Check hinges for play and attachment to wing and flap. Check hinges for corrosion .Check flap push-pull rods and rod end bearings for condition, and lubricate. Inspect condition of bonding strap.		❑	❑
93. **Flap Motor and Flap Actuators** – Clean and cycle flaps up and down to check for smooth operation.		❑	❑
94. **Flap deflections** – Ensure that flaps extend equally on each side of the airplane in both the takeoff and landing configurations. Measure the down deflection on each side using neutral ailerons as a reference point. The difference in static deflection should not be greater than $^1/_4$ in.			❑
95. **SpeedBrakes** – Check Cap Strip Cover screws for security, if loose, remove screws and apply Loctite 242, retorque to 8 in-lbs. Check SpeedBrake top attachment screws for security, if loose, remove screws and apply Loctite 242, retorque to 8 in-lbs. Check drain tubes for debris	❑		
96. **SpeedBrakes** – Remove cover Plate. Clean and Inspect unit for damage, corrosion, looseness & proper operation Lubricate worm and worm gear with LUBRIPLATE. Install cover plate.			❑
97. **Fuel Leaks** – Check the security of the fuel tanks by inspecting the outer skin tank areas for evidence of fuel stains. Be sure to open the gascolator access panel and inspect all fuel line and fittings in this area.		❑	❑
98. **Fuel filler and caps** – Inspect for proper locking, condition of o-ring and receptacle, and presence and legibility of placards.		❑	❑
99. **Wing interior** – Inspect wing spar through outer access panel for signs of cracking or debonding. Inspect visible bonded areas of ribs and other structures.		❑	❑
100. **Fuel Vent Hose** – Check for proper connection of the fuel vent hoses to the outboard inspection panels. Check the condition of the hoses.		❑	❑
101. **Flight controls** – Inspect all push-pull rods, rod end bearings, and bellcranks for condition, play, and security of attachment. Ensure all rod end jamnuts are tight.		❑	❑
102. **Contamination Test** – Take fuel samples from both wings and fuel strainer. Inspect for contamination and proper grade of fuel. Clean strainer screen each 100 hours or annually, whichever occurs first.	❑	❑	❑
103. **Garmin GFC 700 Autopilot System –** Inspect for condition and function. See Chapter 22.			❑
LANDING GEAR (See Chapter 32 for Specific Details)			
104. **Nose Gear and shimmy dampener** – Place nose gear on a rotating table (shallow height – sometimes referred to a "Grease Plate") with the normal static weight of the airplane applied to the table. A firm and consistent resistance to the steering rotation should be felt through the full ±60º of wheel movement.		❑	❑
105. **Lifting the Airplane** – Install jacking pads and jack aircraft until all tires clear the floor. (See Chapter 7 for jacking instructions.)		❑	❑
106. **NLG** – Remove nose gear wheel cover and strut fairing. Note that the four bolts attached to the nose gear fork must be removed to remove the nose wheel fairing.		❑	❑
107. **NLG strut** – Inspect from top to bottom for scratches, cracks, corrosion, and signs of overstress. Check for play in strut attaching brackets. Visually inspect for straightness along length of each strut.		❑	❑

Cessna 350 Model LC42-550FG **Instructions for Continued Airworthiness** **Inspection and/or Required Maintenance Checklist**	**Each 50 hours**	**Each 100 hours**	**Annually**
108. **MLG**– Remove main landing gear wheel fairings.		❑	❑
109. **Hydraulic Brake Lines** – Inspect brake lines that are tie wrapped to the main gear strut. Check for security and evidence of chaffing. Check for leaks as evidenced by the presence of hydraulic fluid stains.		❑	❑
110. **Brake calipers and brake lines** – Clean and inspect for condition, fluid leakage, and security of components. Check caliper for small amount of free-floating motion in torque plate. Clean anchor bolts and torque plate if caliper does not free-float. Do not lubricate.		❑	❑
111. **Brake linings** – Inspect brake linings, and replace if the wear lines are not visible.		❑	❑
112. **Wheels and brake discs** – Inspect for cracks and corrosion. Inspect brake discs for cracks, pitting, and signs of overheating. Inspect all hardware for signs of loss of torque in accordance with Cleveland Maintenance Manual AWBCMM0001-2/USA. Measure and record brake disc thickness: Left - _____inches, Right - _____inches.		❑	❑
113. **MLG tires** – Inspect for flat spots, wear, splitting, and dry-rotting. Check tire pressure, and service as necessary.	❑	❑	❑
114. **MLG wheel pants** – Inspect for condition. Clean interior.			❑
115. **MLG wheel assemblies** – Rotate and check for binding, unusual noises, excessive friction, play in bearings, and security of attachment of axles.		❑	❑
116. **NLG assembly** – Verify operation of NLG centering lock with strut fully extended. Remove four attaching bolts from bottom of NLG strut, remove wheel and fork assembly, and remove wheel pant. Remove fairing from strut assembly.		❑	❑
117. **NLG strut and fairing** – Inspect for cracks, corrosion, scratches on piston, fluid leakage, and signs of overstress. Check for security of attachment of strut to engine mount. Check for presence of steering limit markings and condition of strut fairing.		❑	❑
118. **NLG fork assembly** – Inspect for cracks, corrosion, signs of overstress, and side-loading. Inspect towing lug assemblies for cracks, distortion, and security of attachment.		❑	❑
119. **Service Nose Strut** – Check inflation pressure of nose gear, and service as required. See Chapter 12 for servicing instructions.		❑	❑
120. **NLG wheel assembly** – Inspect for cracks and corrosion. Check all hardware for signs of loss of torque in accordance with sections 32-7 and 32-8.		❑	❑
121. **NLG tire** – Inspect for flat spots, wear, splitting, and dry-rotting. Check tire pressure, and service as necessary.	❑	❑	❑
122. **NLG wheel pant** – Inspect for condition. Clean interior.		❑	❑
123. **Wheel bearings (all)** – Clean and inspect for damage, wear, and corrosion. Inspect bearing races for wear and damage. Repack with appropriate grease in accordance with Cleveland Maintenance Manual AWBCMM0001-2/USA. **NOTE: Always wear rubber gloves when handling dry or repacked bearings!**		❑	❑
124. **NLG assembly** – Reinstall strut fairing and wheel pant, then reinstall wheel and fork assembly on strut. Torque four attaching bolts 140 to 180 in-lbs., and apply torque seal.		❑	❑
125. **MLG** – Reinstall outer and inner main gear wheel covers on each wheel.		❑	❑

Cessna 350 Model LC42-550FG Instructions for Continued Airworthiness Inspection and/or Required Maintenance Checklist	Each 50 hours	Each 100 hours	Annually
126. **Lowering airplane** – Ensure all obstructions are clear from under aircraft and landing gear is ready for aircraft weight, then lower aircraft and remove jacking pads.		❑	❑
FUSELAGE INTERIOR (See Chapter 21, 23, 25, 27, 28, 31, 37, 56, 57 for Specific Details)			
127. **Fire extinguisher –** Remove fire extinguisher and inspect. See Chapter 26 for instructions.		❑	❑
128. **Oxygen system –** Inspect flexible lines for security of connections, kinks, or tube discoloration. Reattach or replace lines as necessary. If the Garmin G1000 system is installed see additional procedures in Chapter 35.	❑		❑
129. **Inspection covers** – Remove rear seat, rear seat inspection plates, ELT access cover, right and left console covers, and glare shield.		❑	❑
130. **Flight controls** – Inspect for nicks, scratches and dents in push-pull rods, play in rod end and linear bearings, security of linear bearing mounts and rudder cable guide tubes to fuselage.		❑	❑
131. **Seats** – Inspect all seat structures for cracks, corrosion, and general condition. Check seat controls for positive locking. Inspect cushions and upholstery for condition.		❑	❑
132. **Seat tracks and stops** – Inspect for cracks, wear of latching holes and tracks, and security of tracks and stops.		❑	❑
133. **Restraint systems** – Inspect belts for cuts, fraying, broken stitching, and presence of TSO tags. Check all buckles for proper locking and release. Check inertia reels for proper operation. Check attach points to structure for all belts.		❑	❑
134. **Upholstery and trim** – Inspect for general condition, attachment, and cleanliness.		❑	❑
135. **Placards** – Inspect for presence and condition of all required interior placards. See Chapter 11 for required placards.		❑	❑
136. **Instruments** – Check security of instruments in panel and legibility of markings. Ensure that no magnetic tools are used during this procedure.		❑	❑
137. **Avionics** – Check security of radios, GPS display, indicators and controls on panel and center console, legibility of markings and operation of knobs. Ensure that no magnetic tools are used during this procedure.		❑	❑
138. **Instrument panel, radio rack, and center console** – Inspect for general condition, security of attachment, and cleanliness. Ensure that no magnetic tools are used during this procedure.		❑	❑
139. **Nav/Com Bypass (Basic and Avidyne Option Only)** – Check operation. Check operation of light.		❑	❑
140. **Garmin G1000 System –** Verify operation. See Chapter 34 for annual inspection procedure.		❑	❑
141. **Magnetic compass** – Inspect for security, clarity of glass, and signs of oil leakage. Inspect compass correction card for presence and legibility of all headings. Ensure that no magnetic tools are used during this procedure.		❑	❑
142. **Fuel selector valve** – Operate through full range and verify that valve locks into each detent. Ensure that green LED's in the fuel gauge (Basic or Avidyne option) or blue lights in the fuel gauge on the MFD (Garmin G1000 option) illuminate.		❑	❑
143. **Fuel selector valve** – Inspect fuel valve for evidence of fuel leakage.		❑	❑

Cessna 350 Model LC42-550FG **Instructions for Continued Airworthiness** **Inspection and/or Required Maintenance Checklist**	**Each 50 hours**	**Each 100 hours**	**Annually**
144. **Environmental Control System (ECS) controls** – Check operation of manual controls.		❑	❑
145. **ECS vents** – Check all vents for proper operation of manual controls.		❑	❑
146. **Instrument panel backside** – Remove glare shield and inspect all wires, lines, hoses, control cables, instruments, etc., for any interference, chafing, and loose or stressed connections. Inspect structure of forward bulkhead, engine mount support structures, etc., for cracks, debonding, and general condition. Inspect soundproofing material for condition and adhesion to forward bulkhead.		❑	❑
147. **Reinstall glare shield** - Check security of attachment and condition. The glare shield for the Garmin G1000 option has foil attached and must be grounded correctly with a "U" clip. Check pull-out shield (Avidyne option only) for freedom of movement, condition, and security.		❑	❑
148. **ECS ducts** – Inspect for cracks, holes, routing, and chafing.		❑	❑
149. **Grounding rails** – Inspect for cracks and security along forward length. Pay special attention to bonding straps connecting sections of rails at formers and to engine mount bolts.		❑	❑
150. **Rudder pedals** – Inspect for security, cracks, and play. Check for leakage from brake cylinders. Lubricate rudder cable clevis bolt and brake cylinder attach points. **Do not lubricate rudder pedal torque tube pillow blocks.**		❑	❑
151. **Parking brake valve** – Inspect for security of mounting and signs of leakage.		❑	❑
152. **Seat box structure** – Inspect for cracks, debonding, and any signs of obvious damage.		❑	❑
153. **Wing spar** – Inspect for cracks, debonding, and security of attachment to fuselage.		❑	❑
154. **MLG box support structure** – Inspect for cracks, debonding, and security of hardware. If paint has been removed, metal is exposed, or corrosion is present, clean the surface and apply paint as needed.		❑	❑
155. **Floor and supporting structure** – Inspect for cracks, holes, and security of attachment.		❑	❑
156. **Flight controls (forward fuselage through baggage compartment)** – Inspect for nicks, scratches, and dents in push-pull rods, play in rod end and linear bearings, security of linear bearing mounts, and attachment of rudder cable guide tubes to fuselage.		❑	❑
157. **Baggage compartment floor** – Inspect for cracks, holes, and security.		❑	❑
158. **Hat shelf** – Inspect for cracks, holes, and security.		❑	❑
159. **Emergency Locator Transmitter (ELT)** – Remove from tray and remove battery cover. Inspect for battery corrosion and any obvious internal or external damage to housing. Verify replacement date on battery matches date on housing placard. Reinstall battery cover. Test operation of G-switch in strict accordance with instructions in Chapter 25. Record battery replacement due date: _____		❑	❑
160. **ELT installation and support shelf** – Inspect ELT wiring and antenna cable for security, routing, and chafing. Check connectors for security of pins and proper connection. Inspect ELT tray and support shelf for cracks and security. Replace tray if any cracks are found.		❑	❑

Cessna 350 Model LC42-550FG **Instructions for Continued Airworthiness** **Inspection and/or Required Maintenance Checklist**	**Each 50 hours**	**Each 100 hours**	**Annually**
161. **Empennage interior structure** – Inspect for cracks, debonding, or other signs of damage. Verify that all drain holes are clear of debris.		❑	❑
162. **Access panels and fairings** – Inspect for condition. Check fasteners and receptacles for condition.		❑	❑
163. **Automatic Climate Control System (ACCS) controls** – Check operation of system and of the system control head.		❑	❑
164. **ACCS air conditioning compressor drive belt** – Check belt condition and tension. Replace the belt whenever there is cracking, fraying, or signs of excessive wear.		❑	❑
165. **ACCS air conditioning compressor drive belt pulleys** – Check pulleys for security, tightness, rotational smoothness, and freedom.		❑	❑
166. **ACCS air conditioning refrigerant hoses** – Inspect hoses for chafing, leads, and security.		❑	❑
167. **ACCS air ducting** – Inspect for intake and outlet obstructions, damage, cracks, holes, routing, and chafing.		❑	❑
168. **ACCS/ECS vents**– Check all vents for proper operation of manual controls. Inspect for intake and outlet obstructions, and damage.		❑	❑
INSPECTION WRAP-UP			
169. **Fuselage and wings** – Verify aircraft is free of any tools, parts, and debris, and reinstall all access fairings, panels, carpeting, seats, etc. removed for this inspection.	❑	❑	❑
170. **Engine** – Verify oil sump is properly serviced with oil, battery box is closed, and engine compartment is free of tools, rags, and debris, and install lower cowl.	❑	❑	❑
171. **Engine** – Run engine for no more than two minutes at 1000 to 1200 RPM. After shutdown, check for leaks at oil filter base, fuel nozzles, and any other components removed during this inspection. If no leaks are noted, install top cowling.	❑	❑	❑
172. **Aircraft** – Operate engine at 1000 to 1500 RPM to warm engine. Operate all aircraft systems to verify proper operation. As engine warms, operate engine systems at appropriate engine speeds and complete all checks listed on Inspection Coversheet.	❑	❑	❑
173. **Aircraft records** – Complete entries in logbooks, AD compliance lists, and any other required records and work orders.	❑	❑	❑

ADDITIONAL COMMENTS

ADDITIONAL COMMENTS

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05-5. LIST OF INSPECTION AREAS

CESSNA 350 (LC42-550FG) INSPECTION ACCESS AREAS			
Area	**Type of Access**	**Location**	**Inspection & Maintenance Items**
LEFT WING	Access Panel	Underside, inboard, below fuel tank	Fuel tank, two fuel sending units, slosh box, fuel low annunciator switch, vapor return lines
	Access Panel	Underside, middle, below fuel tank	Fuel tank, one fuel sending unit
	Access Panel	Underside, outboard	Fuel vent
	Access Panel	Underside trailing edge, near right side of aileron	Final aileron drive, mounting bolts to aileron, bellcrank assembly
	Access Panel	Underside towards leading edge, forward of inboard fuel tank access panel	Fuel lines and fuel lines to fuel tank connection fittings
	Access Panel	Aft of inboard fuel tank cover	Final drive for flap bellcranks
	Access Panel	Underside trailing edge, near middle of aileron	Aileron control rod linear bearing
RIGHT WING	Access Panel	Underside, inboard, below fuel tank	Fuel tank, two fuel sending units, slosh box, fuel low annunciator switch, vapor return lines
	Access Panel	Underside, middle, below fuel tank	Fuel tank, one fuel sending unit
	Access Panel	Underside, outboard	Fuel vent
	Access Panel	Underside trailing edge, near left side of aileron	Final aileron drive, mounting bolts to aileron, bell crank assembly
	Access Panel	Underside towards leading edge, forward of inboard fuel tank access panel	Fuel lines and fuel lines/fuel tank connection fittings, outside air temperature probe
	Access Panel	Aft of inboard fuel tank cover	Final drive for flap bellcranks
	Access Panel	Underside trailing edge, near middle of aileron	Aileron control rod linear bearing
FUSELAGE EXTERIOR	Access Panel	Underside center near trailing edge of the flaps	Aileron bellcranks and crossover assembly, flap motor, marker beacon tuning slug
	Access Panel	Underside near left wing root	Auxiliary fuel pump, gascolator, pitot drain, wires and plugs for wing electrical (position/strobe lights, landing/taxi lights, stall horn, fuel quantity, and fuel low annunciator)
	Access Panel	Tail cone right side	Air vent, elevator push-pull rods
	Access Panel	Vertical stabilizer, left side	Antenna connection junction

CESSNA 350 (LC42-550FG) INSPECTION ACCESS AREAS			
Area	**Type of Access**	**Location**	**Inspection & Maintenance Items**
	Engine Cowling – Top and Bottom	Forward of fuselage firewall	All engine and propeller parts and components.
FUSELAGE INTERIOR	Access Panel	Under front seat bench	Fuel and fuel vent lines
	Access Panels (2)	Right side of center console (arm rests)	Fuel selector valve, environmental control system (ECS) heated air ducts
	Cover Panel	Rear seat footwell	Wing attachment bolts
	Access Panels (2)	Under rear seat back	Gear box
	Access Panels (2)	Baggage compartment floor	Gear box
	Access Panel	Floor area of baggage compartment (Avionics Bay)	Special optional avionics equipment (DME, HSI slaving transmitter, remote stormscope sensor)
	Access Panel	Floor area of baggage compartment hat rack	Power plug, pitch servo
	Access Panel	Aft bulkhead of baggage shelf	ELT, elevator push-pull rods
	Cabin, Side-Trim Panels	Pilot's side – runs from control stick to rear seat back	Aileron and elevator push-pull rods, circuit breaker panel, control stick switches wiring, fuel lines, ECS fresh air ducts
	Cabin, Side-Trim Panels	Copilot's side – runs from control stick to rear seat back	Aileron and elevator push-pull tubes, communication wiring, control stick switches wiring, common wire bundle, ECS fresh air ducts
	Baggage Compartment Trim Panels (2)	Left and right sides of baggage compartment	Elevator push-pull rods
CONTROLS	Access Panel	Right elevator, underside, middle	Trim tab servo motor
	Access Panel	Underside, right aileron	Trim tab servo motor

CHAPTER 6

DIMENSIONS & AREAS

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Chapter 6

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06-1. POWERPLANT LIMITATIONS

a. Number of Engines: One (1)
b. Number of Cylinders: Six (6)
c. Engine Manufacturer: Teledyne Continental
d. Engine Model Number: IO-550-N
e. Maximum Power: 310 BHP at 2700 RPM
f. Maximum Manifold Pressure: Full power at sea level
g. Maximum Cylinder Head Temperature: 460°F

06-2. POWERPLANT FUEL AND OIL DATA

a. **Oil Grades Recommended for Various Average Air Temperature Ranges**
   1. Below 40°F: SAE 30, 10W30, 15W50, or 20W50
   2. Above 40°F: SAE 50, 15W50, 20W50, or 20W60

b. **Oil Temperature**
   1. Maximum Allowable: 240ºF
   2. Recommended takeoff minimum: 75°F
   3. Recommended flight operations: 170°F to 200°F

c. **Oil Pressure**
   1. Normal Operations: 30-60 psi (pounds per square inch)
   2. Idle, minimum: 10 psi
   3. Maximum allowable (cold oil): 100 psi

d. **Approved Fuel Grades**
   1. 100LL Grade Aviation Fuel (Blue)
   2. 100 Grade Aviation Fuel (Green)

e. **Fuel Flow and Fuel Pressure**
   1. Normal Operations: 10 to 22 GPH (7 to 16 psi)
   2. Idle, minimum: 1 to 2 GPH (4 psi)
   3. Maximum allowable: 25 GPH (18 psi)

06-3. PROPELLER DATA AND LIMITATIONS

a. Number of Propellers: One
b. Propeller Manufacturer: Hartzell
c. Propeller Hub and Blade Model Numbers: PHC-J3YF-IRF and F7691D-14
d. **Propeller Diameters**
   1. Minimum: 76 inches
   2. Maximum: 77 inches
e. **Propeller Blade Angle at 30 inch Station**
   1. Low: 14.1° ± 0.2°
   2. High: 34.7° ± 1°

06-4. POWERPLANT INSTRUMENT MARKINGS

a. The following table shows applicable color-coded ranges for the various powerplant instruments within the aircraft.

INSTRUMENT	RED LINE Minimum Limit	GREEN ARC Normal Operating	RED LINE Maximum Limit
Tachometer	Minimum for idle 600 RPM (No Placard)	2000 - 2700 RPM	2700 RPM
Manifold Pressure	N/A	18-28 Inches Hg. (No Placard)	
Oil Temperature	Minimum for takeoff 75ºF (No Placard)	160 – 210ºF	240ºF
Oil Pressure	Minimum for idle 10 psi (No Placard)	30 - 60 psi	100 psi (Cold Oil) (No Placard)
Fuel Quantity	"E" or "zero" reading in level flight indicates the remaining four gallons (S/N 42001 to 42567), or two gallons (S/N 42568 and on), in each tank cannot be used safely in flight.	N/A	N/A
Fuel Pressure/Fuel Flow	N/A	10 - 22 GPH 4 – 14 psi (No Placard)	25 GPH (18 psi)
Cylinder Head Temperature	N/A	240 – 460ºF	460ºF

Figure 6 - 1 Powerplant Instrument Markings

06-5. POWER SETTING LIMITATIONS

a. Do not exceed 20 in. of Hg. manifold pressure below 2200 RPM. This requirement is not an engine limitation, but rather a harmonic condition inherent in the Cessna 350 (LC42-550FG).

06-6. WEIGHT LIMITS (Utility Category)

a. Maximum Ramp Weight: 3400 lbs.
b. Maximum Zero Fuel Weight: FS (103) 2725 lbs. – FS (110) 3228 lbs.
c. Maximum Empty Weight: 2568 lbs.
d. Maximum Takeoff Weight: 3400 lbs.
e. Minimum Flying Weight: FS (103) 2240 lbs. – FS (110) 2500 lbs.
f. Maximum Landing Weight: 3230 lbs.
g. Maximum Baggage Weight: 120 lbs.

06-7. CENTER OF GRAVITY LIMITS

a. The following table, Figure 6 - 2, specifies the center of gravity limits for utility category operations. The variation along the arm between the forward and aft datum points is linear (straight line). The straight-line variation means that at any given point along the arm, an increase in moments (expressed as inch-pounds) changes directly according to the variations in weight and distance from the datum.

CATEGORY	FORWARD DATUM POINT AND WEIGHT	AFT DATUM POINT AND WEIGHT	VARIATION
Utility Category	103 inches 2240 to 2500 lbs.	110 inches 2500 to 3400 lbs.	Straight Line

Reference Datum: The reference datum is located at the tip of the propeller spinner. This location causes all arm distances and moments (the product of arm and weight) to be positive values.

Figure 6 - 2 Center of Gravity Limits

06-8. CABIN AND ENTRY DIMENSIONS

a. Maximum Cabin Width: 50 inches (Outside Dimension)/49 inches (Inside Dimension)
b. Maximum Cabin Length: (Firewall to aft limit of baggage compartment): 139.6 inches
c. Maximum Cabin Height: 47 inches
d. Minimum Entry Width: 33 inches
e. Minimum Entry Height: 28 inches/Clearance with Door Open: 46 inches

06-9. SPACE AND ENTRY DIMENSIONS OF BAGGAGE COMPARTMENT

a. Maximum Baggage Compartment Width: 38.5 inches
b. Maximum Baggage Compartment Length: 52 inches
c. Maximum Baggage Compartment Height: 34.5 inches
d. Maximum Baggage Entry Width: 27 inches

06-10. SPECIFIC LOADINGS

a. Wing Loading: 24.08 lbs./ft.^{2}
b. Power Loading: 11.3 lbs./hp.

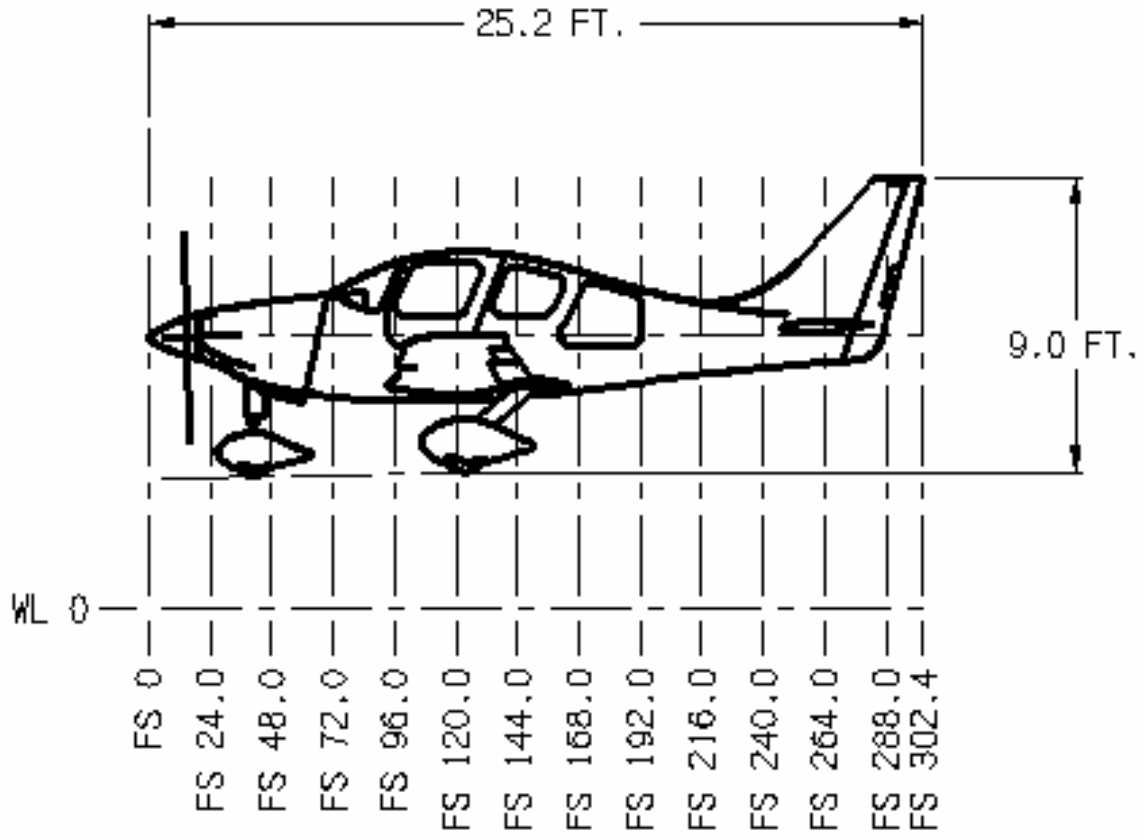
06-11. OTHER SPECIFICATIONS

a. Wing Area: 141.2 ft.
b. Wing Span: 35.8 ft.
c. Angle of Incidence Wing: 0º
d. Angle of Incidence Horizontal Stabilizer: -2.0º
e. Dihedral: 4.5º
f. Aspect Ratio: 9.18
g. Mean Aerodynamic Chord: 48.275 inches
h. Fuselage Length: 25.7 ft.
i. Height of Vertical Stabilizer: 9 ft.
j. Total Horizontal Stabilizer Area: 36.4 ft.^{2}
k. Stabilizer Span: 14.1 ft.
l. Vertical Stabilizer Area: 15.68 ft.^{2}
m. Total Aileron Area: 5.75 ft.^{2}
n. Rudder Area: 4.46 ft.^{2}
o. Total Flap Area: 20.36 ft.^{2}
p. Total Elevator Area: 11.48 ft.^{2}

06-12. LANDING GEAR

a. Nose Strut Pressure: 250 psi (Nitrogen)
b. Landing Gear Type: Fixed (Steel)
c. Landing Gear Track Width: 89.5 inches
d. Nose Tire Size: 5.00-5 (10 Ply)
e. Nose Tire Pressure: 88 psi
f. Main Gear Tire Size: 15x6.00-6 (6 Ply)
g. Main Gear Tire Pressure: 55 psi

06-13. FUSELAGE STATIONS



06-14. WING AND HORIZONTAL STABILIZER STATIONS

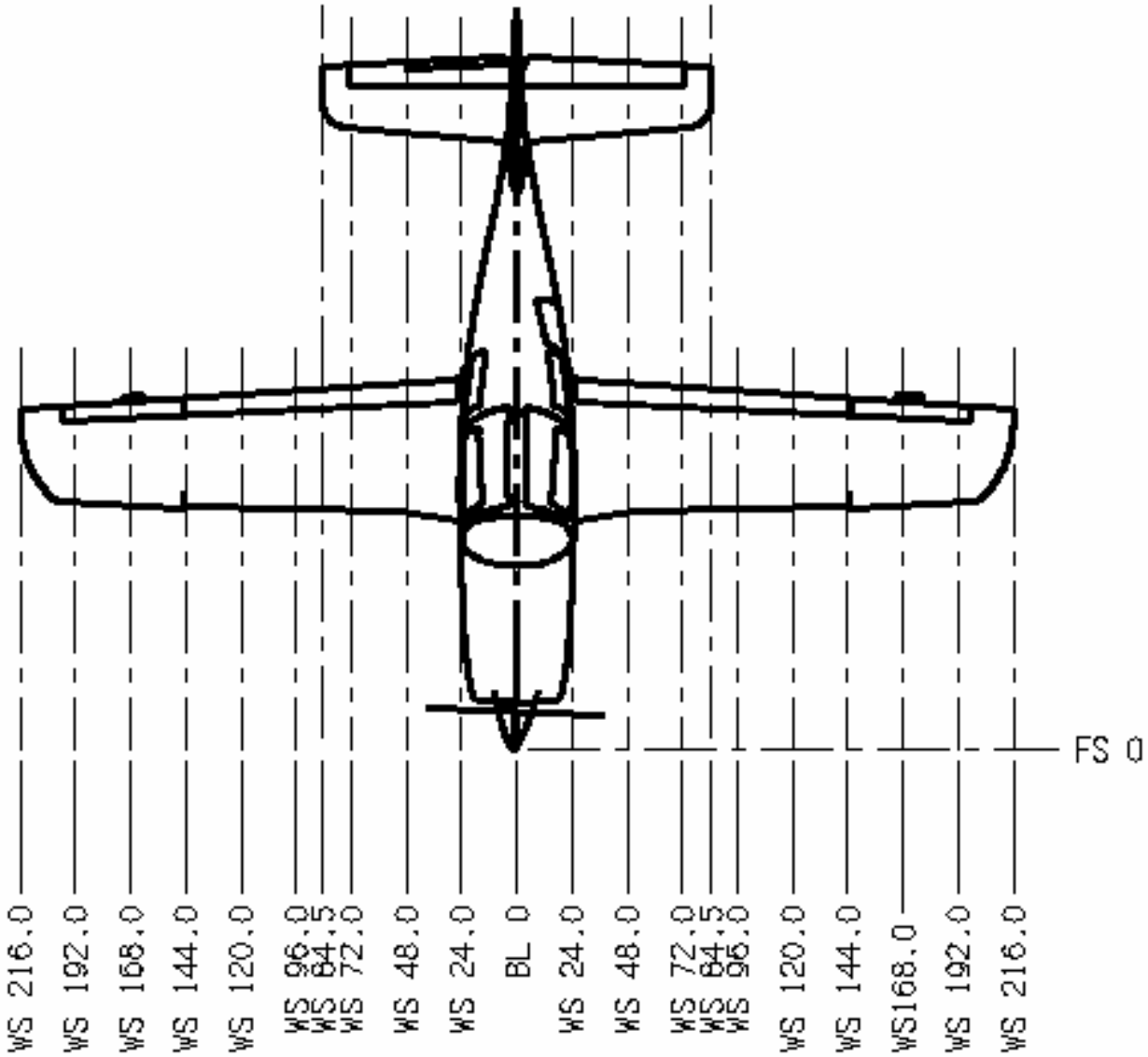


Figure 6 - 4 Wing Stations (WS)

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CHAPTER 7

LIFTING & SHORING

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07-1. JACKING THE AIRPLANE

a. **Lifting Points** – There are two jack points under each wing proximate to the wing saddle. The points are near the center of mass of the longitudinal axis, and great care must be used when jacking the airplane. The tailskid is used as a third point of stabilization.

b. **Raising One Wing Only** – If only one wing is raised, as when changing a single tire, the airplane can be safely jacked by one person using steps 1 through 5 below.

1. Perform operation in a level area, such as an airplane hangar.
2. Set the parking brake and chock the nose tire and the main gear tire that is not raised.
3. Place 50 pounds of ballast (usually sandbags) on the engine cowling, near the propeller.
4. Place a jack under the jack point of the wing to be lifted and raise the jack up to the wing jack point. Take extra precaution to ensure the jack is properly stabilized, the base is locked in position, and the jack is lifting vertically. Be sure the raising point of the jack is properly inserted into the jack point on the wing.
5. Slowly raise the jack until the desired ground clearance is achieved. The clearance between the bottom of the tire and lifting surface (ground or hangar floor) must not exceed three inches.

c. **Raising Both Wings** – If the airplane is simultaneously lifted by both jacks, then the procedures established in steps 1 through 8 below should be followed.

1. Perform operation in a level area, such as an airplane hangar.
2. Use a minimum of three trained individuals when jacking the airplane, one person for each jack and one observer.
3. Because the jack points are near the airplane's center of gravity, 50 pounds of ballast must be placed on the horizontal stabilizer. Sandbags are perfect for this operation since they distribute pressure evenly and will not mar the surface.
4. Attach the tail to a stabilizing stand.
5. Place the sandbags or other ballast on the horizontal stabilizer as close as possible to the longitudinal axis.
6. Place a jack under each jack point of the wing. Raise the jacks up to the wing jack point. Take extra precaution to ensure the jacks are properly stabilized, the bases are locked in position, and the jacks are lifting vertically. Be sure the raising point of the jacks are properly inserted into the jack points on the wing.
7. Slowly raise the jacks until the desired ground clearance is achieved. The observer must ensure the jacks are lifting vertically and there is no tail wobble. As the airplane is lifted off the surface, the observer must connect the stabilizing stand to the tail.
8. Review all of the caution items on the next page.

CAUTION

There are several points that must be considered when raising the airplane with two jacks.

1) **It is best to jack the airplane in an enclosed area. Wind gusts will destabilize the airplane and cause it to fall off the jacks.**
2) **The airplane must be jacked in a level area on a concrete surface.**
3) **Raise the jacks simultaneously in short increments to ensure the airplane stays as level as possible.**
4) **If the jacks have locking pins, they must be inserted once the desired jacking height is achieved. Otherwise, the airplane must be continuously observed to ensure the jacks do not bleed off pressure. If the jack loses pressure while the tail is attached to the tail stand, damage to the tail will result.**
5) **Once the airplane is on jacks, do not attempt to enter the airplane.**
6) **When lowering the airplane, remove the tail stand support and bleed pressure evenly from jacks in short increments.**

CHAPTER 8

LEVELING & WEIGHING

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08-1. GENERAL

a. To determine the empty weight and center of gravity of the airplane, the airplane must be in a level area and in a particular configuration.

08-2. AIRPLANE CONFIGURATION (Empty Weight)

a. The airplane empty weight includes eight quarts of oil (dipstick reading), unusable fuel, hydraulic brake fluid, and installed equipment.

b. Defuel the airplane per instructions in Chapter 12 of this manual.

c. Ensure the oil sump is filled to eight quarts (cold engine). Check the reading on the dipstick and service as necessary.

d. Place the pilot's and front passenger's seat in the full aft position.

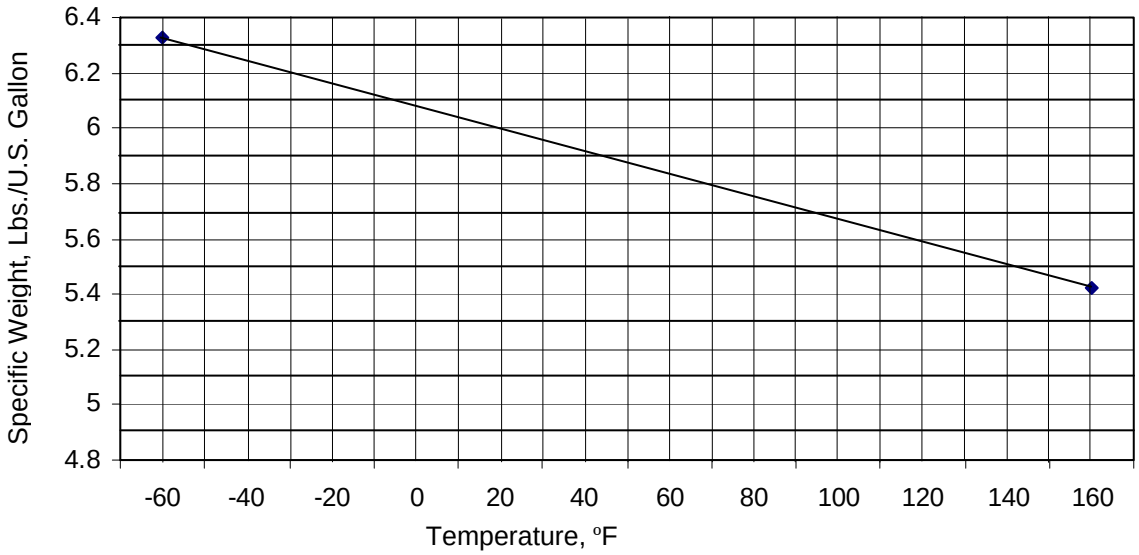
e. Retract the flaps to the cruise or 0° position.

f. Center the controls to the neutral static position.

g. Ensure all doors, including the baggage door, are closed when the airplane is weighed.

CAUTION

It is not recommended to weigh an airplane with full fuel and subtract the weight of the fuel to obtain empty weight because the weight of fuel varies with temperature. If this method of weight determination is used, fuel weight should be calculated conservatively. Use the specific weight of fuel at ambient temperature. See table and example below.



Average Specific Weight of Aviation Gasoline (Mil-F-5572 Grade 100/130 Type) Versus Temperature

The following is offered as an example only. It is important to remember that the aircraft weight in the example does not apply to a specific airplane.

Example:

Unconservative Calculation

Conventionally used fuel specific weight (6 lbs./U. S. gal.)

Total Aircraft weight with fuel	= 3038 lbs.
Weight of fuel (98 gal. x 6 lbs./U. S. gal.)	= 588 lbs.
Airplane empty weight (3038 lbs – 588 lbs.)	= 2450 lbs.

Conservative Calculation

Fuel specific weight at 60 ºF (5.83 lbs./U. S. gal.)

Total Aircraft weight with fuel	= 3038 lbs.
Weight of fuel (98 gal. x 5.83 lbs./U. S. gal.)	= 571 lbs.
Airplane empty weight (3038 lbs – 571 lbs.)	= 2467 lbs.

08-3. AIRPLANE LEVELING

a. Since there are no perfectly level reference areas on the airplane and the use of digital inclinometers is not common, the airplane is leveled by use of a plumb bob suspended over a fixed reference point in the baggage compartment. Moreover, since the use of jacks with load cells is not prevalent, the wheel scales method is described in this manual. The following steps specify the procedures for installing the plumb bob and leveling the airplane. These steps must be completed before taking readings from the wheel scales.

1. The airplane must be weighed in a level area.
2. Remove the left rear seat cushion and place in the footwell. When the cushion is removed, a small washer, which is bonded to the bottom of the seat frame, will be exposed.
3. Using a string with a plumb bob attached to it, run the string over the gas strut door flange between the flange ball and the point where the baggage door gas strut attaches to the ball, and tie the string off around the front seatbelt bracket. See Figure 8 - 1.

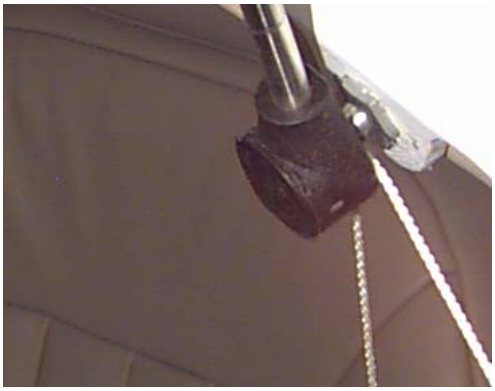


Figure 8 - 1 Attaching String to Gas Strut

4. The pointed end of the plumb bob, in a resting state, will be near a $^{3}/_{16}$ in. washer bonded into the seat frame.
5. Using the two jack method (raising both wings) discussed in Chapter 7, position the two main tires and the nose tire of the airplane on three scales. Ensure the brakes are

set before raising the airplane off the floor. When all of the airplane's weight is on the three scales, move the jacks to a location that is not under the wings.

6. It will be necessary to either deflate the nose tire or strut and/or main tires to center the plumb bob point over the washer. When the pointer of the plumb bob is over any part of the washer, the airplane is level.
7. Once the airplane is level, release the brakes.

08-4. USING THE PERMANENT REFERENCE POINT

a. To determine the empty weight center of gravity of the airplane, it is more convenient to work with the permanent reference. The permanent reference point on the airplane is located at the forward part of the wing bottom, in the center of the wing saddle and is 97.05 inches aft of the datum. The location is shown in Figure 8 - 2. There is a pronounced seam at the point where the fuselage is attached to the wing, and the leading edge of the wing bottom is easy to identify.

b. Suspend a plumb bob from the permanent reference point in the exact center as shown in Figure 8 - 2 and Figure 8 - 4.

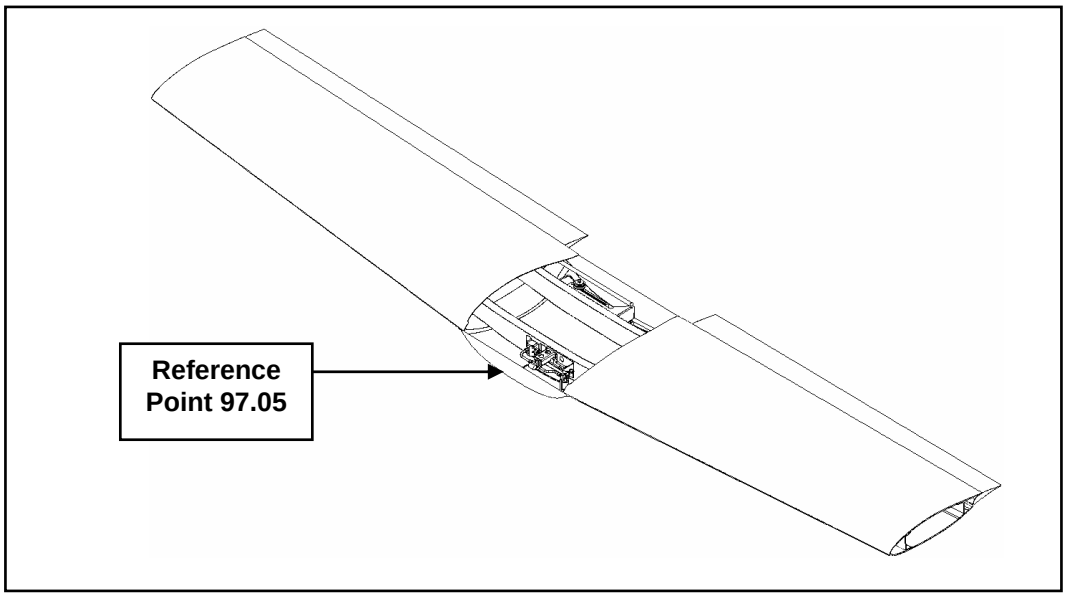


Figure 8 - 2 Reference Point for on Wing Saddle

c. Determine the center point on each tire and make a chalked reference mark near the bottom where the tire touches the floor. On the main gear tires, the mark should be on the inside, near where the arrows point in Figure 8 - 3.

d. Create a lateral reference line between the two main gear tires. This can be accomplished by stretching a string between the two chalk marked areas of the tires, snapping a chalk line between these two points, or laying a 7.3 foot board between the points.

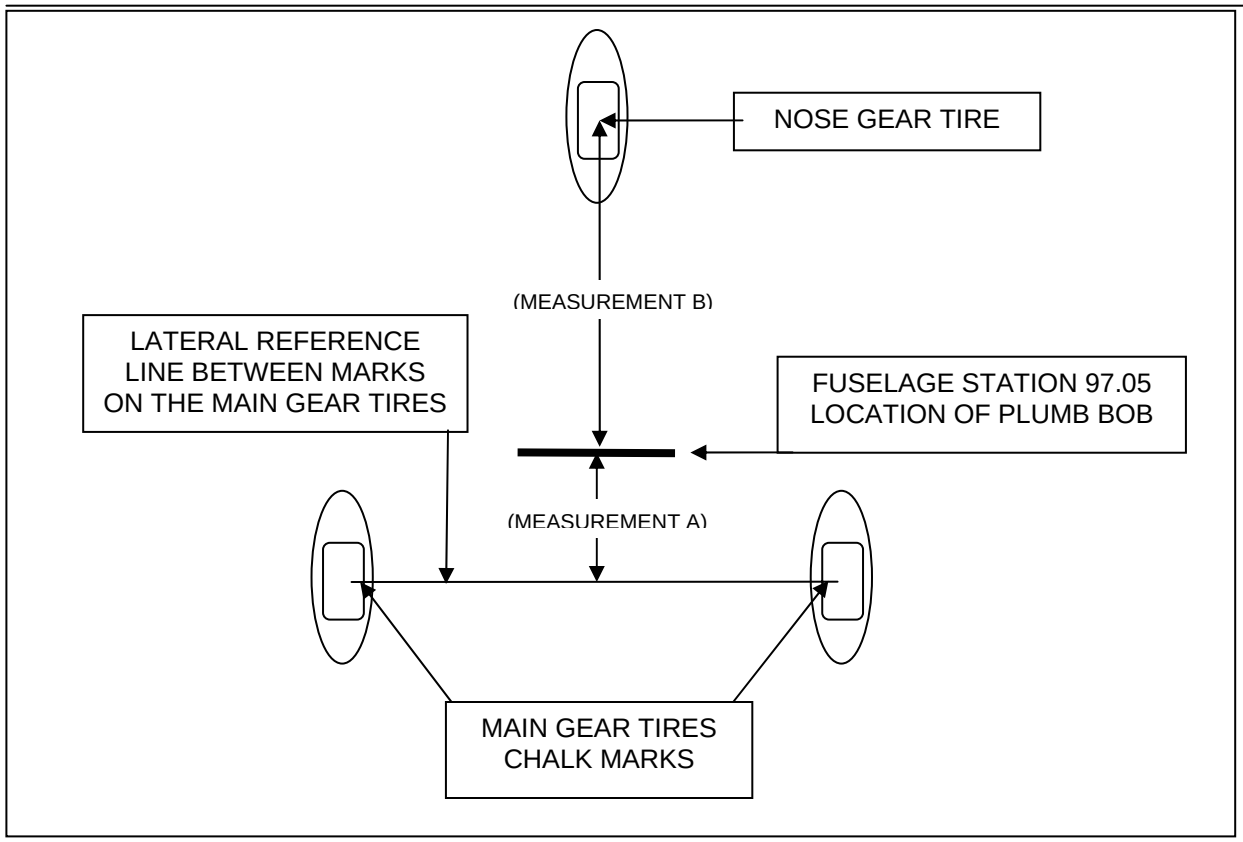


Figure 8 - 3 Diagram of Measurements for Weighing

08-5. MEASUREMENTS

a. Measure the distance along the longitudinal axis from the permanent reference point (tip of the plumb bob) to the lateral reference line between the two tires of the main gear. This is Measurement A in Figure 8 - 3 and Figure 8 - 4.

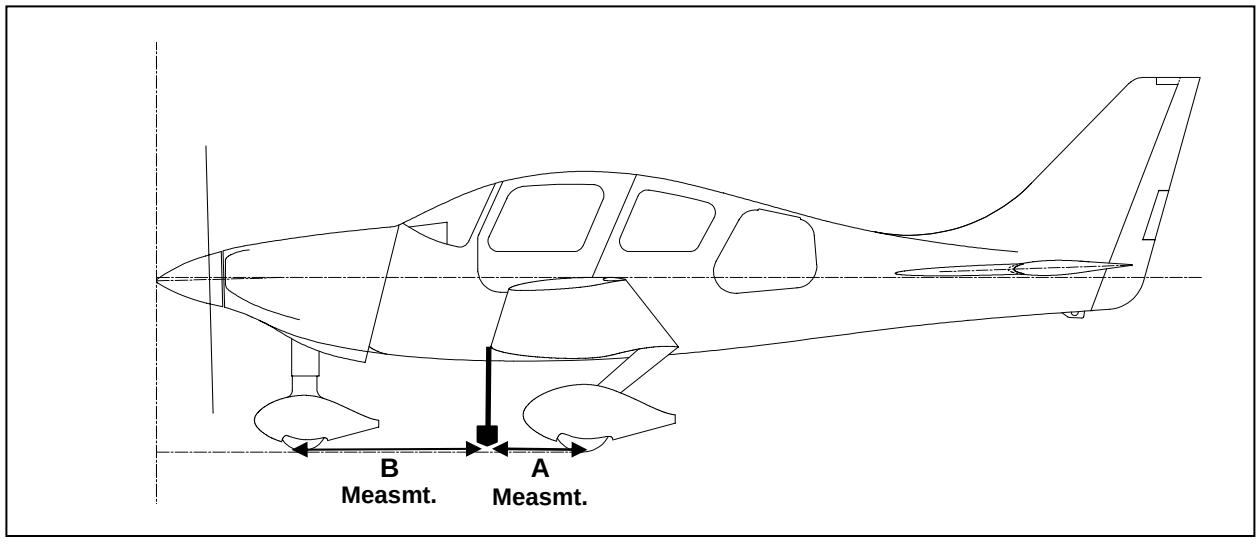


Figure 8 - 4 Diagram of Measurements for Weighing

b. Measure the distance along the longitudinal axis between the plumb bob to the mark on nose tire. This is Measurement B in Figure 8 - 3 and Figure 8 - 4.

08-6. CONVERTING MEASUREMENTS TO ARMS

a. To convert Measurement A and B distances to arms, use the formulas shown in Figure 8 - 5 and Figure 8 - 6, respectively.

MAIN GEAR

Measurement A Distance + 97.05 inches = Main Gear Arm

Figure 8 - 5 Determining Main Gear Arm

NOSE GEAR

97.05 inches - Measurement B Distance = Nose Gear Arm

Figure 8 - 6 Determining Nose Gear Arm

08-7. WEIGHTS AND COMPUTATIONS

a. The main gear scale should be capable of handling weight capacities of up to 1200 lbs., while the nose gear scale needs a capacity of at least 750 lbs.

b. Computing the total weight and moments requires seven steps or operations. These seven operations are shown below and also identified in Figure 8 - 7.

	Operation No. 1	Operation No. 2	Operation No. 3		Operation No. 4		Operation No. 5
Scale Location	**Weight Reading (lbs.)**	**Tare or Scale Error**	**Corrected Weight (lbs.)**	**X**	**Arm (Inches)**	**=**	**Moments (lbs.- inches)**
Right Main Gear	Right Scale Reading	Scale Error	Right Scale Wt. ± Error	X	Main Gear Arm	=	Right Gear Moment
Left Main Gear	Left Scale Reading	Scale Error	Left Sale Wt. ± Error	X	Main Gear Arm	=	Left Gear Moment
Nose Gear	Nose Scale Reading	Scale Error	Nose Scale Wt. ± Error	X	Nose Gear Arm	=	Nose Gear Moment
Total Empty Weight and Empty Moments			**Total Corrected Weight** Operation No. 6				**Total Moments** Operation No. 7

Figure 8 - 7 Determining Weight and Moments

1. **Operation No. 1** – Enter the weight for each scale into the second column.
2. **Operation No. 2** – Next, enter the scale error. The scale error is sometimes referred to as the tare and is entered in the third column for each scale.
3. **Operation No. 3** – Add or subtract the respective error for the three scales and enter the result into the forth column. This is the correct weight.

4. **Operation No. 4** – Using the formulas shown in Figure 8 - 5 and Figure 8 - 6, determine the arm for the main gear and nose gear. Enter this information into the fifth column.
5. **Operation No. 5** – Multiply the corrected scale weights times their respective arms to determine the moments for each location. Enter the moments for each computation in the sixth column.
6. **Operation Nos. 6 and 7** – Sum the weights in the fourth column and the moments in the sixth column. Note: The areas of primary calculations have a double outline.

c. The final step, which is to determine the empty center of gravity, is to divide the total moments by the total corrected weight. A detailed example of this computation is shown in Figure 8 - 9.

08-8. EXAMPLE OF EMPTY CENTER OF GRAVITY (CG) DETERMINATION

a. The following is offered as an example problem to aid in understanding the computation process. It is important to remember that the weights, arms, and moments used in the example problem are for demonstration purposes only and **do not** apply to a specific airplane.

b. For the example problem, assume the following.
   1. Scale Weights
      a) Right Main Gear – 924 pounds
      b) Left Main Gear – 923 pounds
      c) Nose Gear – 490 pounds
   2. Scale Error (Tare)
      a) Right Main Gear Scale is –2 pounds
      b) Left Main Gear Scale is –1 pound
      c) Nose Gear Scale is + 3 pounds
   3. Measurements
      a) Measurement Distance A is 24.05 inches
      b) Measurement Distance B is 56.15 inches

c. These uncorrected scale weights and tares are shown in Figure 8 - 8. Note that after correcting for scale error, the right, left, and nose gear weights are 922.0, 922.0, and 493.0 pounds, respectively.

d. The arm for the main gear is computed as follows using the formula in Figure 8 - 5.

Measurement distance A + 97.05 inches = Main Gear Arm (MGA)
or
24.05 inches + 97.05 inches = 121.1 inches MGA

e. The arm for the nose gear is computed as follows using the formula in Figure 8 - 6.

97.05 inches – Measurement Distance B = Nose Gear Arm (NGA)
or
97.05 inches – 56.15 inches = 40.9 inches NGA

f. The main and nose gear arms, as computed, are shown in Figure 8 - 8.

g. The corrected weights of 922 pounds are then multiplied with the 121.1 inch main gear arm, which produces total moments of 111,654.2 lbs.-inches. In this example, the moments are the same for both the right and left gear since the weights are the same. However, it is not uncommon for the right and left gear weights to vary a few pounds.

h. Next, the corrected 493.0 nose gear weight is multiplied times its 40.9 inch arm, which produces a moment value of 20,163.7 lbs.-inches.

i. Finally, the total moments and corrected weight are summed. In the example below, the total weight is 2,337.0 pounds and the total moment is 243,472.1 lbs.-inches. All this information is summarized in Figure 8 - 8. All required data for determining the empty center of gravity are now available.

Scale Location	Weight Reading (lbs.)	Tare or Scale Error	Corrected Weight (lbs.)	X	Arm (Inches)	=	Moments (lbs.- inches)
Right Main Gear	924	- 2	922.0	X	121.1	=	111,654.2
Left Main Gear	923	-1	922.0	X	121.1	=	111,654.2
Nose Gear	490	+3	+493.0	X	40.9	=	+20,163.7
Total Empty Weight and Empty Moments			2,337.0				243,472.1

Figure 8 - 8 Sample Calculation of Weight and Moments

The formula for determining empty weight center of gravity is expressed as follows:

$$\frac{Total\ Moments}{Empty\ Weight} = Center\ of\ Gravity \quad or \quad \frac{243{,}472.1\ lbs. - inches}{2{,}337\ lbs.} = 104.2\ inches$$

Figure 8 - 9 Determining Center of Gravity

j. In the above example the empty center of gravity of the airplane is at fuselage station (FS) 104.2.

08-9. CHANGES IN THE AIRPLANE'S CONFIGURATION

a. **Determining Location (FS) of Installed Equipment in Relation to the Datum –** If equipment is installed in the airplane, the weight and balance information must be updated. Individuals and companies who are involved with equipment installations and/or modifications are generally competent and conversant with weight and balance issues. These individuals or companies must be aware that the fixed datum is located at fuselage station (FS) 97.05. Please see Figure 8 - 2 and paragraph 08-4 for more information.

b. **Weight and Balance Forms** – There is a form in Appendix A of Chapter 6 of the *Pilot's Operating Handbook* that is used to track changes in the configuration of the airplane. When equipment is added or removed, these pages or an appropriate approved form must be updated. In either instance the required information is similar.

c. **Updating The Form –** Fill in the date the item is added or removed, a description of the item, the arm of the item, the weight of the item, and the moment of the item. Remember to multiply the weight times the arm of the item to obtain the moment. Finally, compute the new empty weight and empty moment by adjusting the running totals. If an item is removed, subtract the weight and moment of the item from the running totals. If an item is added, add the weight and moment of the item to the running totals.

CHAPTER 9

TOWING & TAXIING

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09-1. GENERAL

a. The following pages contain instructions for towing and taxiing the airplane. It is important to remember that the airplane has a free castoring nose wheel with stops at the 60º left and right positions. Personnel unfamiliar with the techniques required when taxiing an airplane with a free castoring nose wheel should be given appropriate instruction or a taxi checkout.

09-2. GROUND HANDLING

a. Towing – A locking, hand-held tow bar is provided with the airplane and stored in the baggage compartment. The tow bar is inserted into two small holes in the nose wheel fairing, forward of the nose wheel axle. The tow bar must be locked in place before attempting to move the airplane. The tow bar is collapsible for storage by removing the locking pin. With the pin removed, slide the handle in and reinsert the locking pin. It is recommended that the airplane only be maneuvered during towing by use of the hand-held tow bar. If it is necessary to tow with a vehicle, extreme care is required to ensure the rotation limits of the nose wheel (60° left and right) are not exceeded. Since the rotation of the nose gear is limited by physical stops, rotating the gear beyond 60° will damage the airplane.

b. It is a good idea to have another person serve as a spotter when towing the airplane, particularly if a vehicle is used. Remember that the airplane has vertical limitations as well as horizontal restrictions. The vertical stabilizer is frequently overlooked as an airplane is being pushed into a hanger with most of the attention directed towards the wingtips. When moving the airplane over uneven surfaces, remember that small up and down oscillations of the nose strut result in amplified movement of the vertical stabilizer. Finally, keep in mind that the height of the vertical stabilizer is affected by inflation levels of both the nose tire and strut. A flat tire or low nose strut will increase the height of the vertical stabilizer.

CAUTION

Do not attempt to move the airplane by pushing or pulling on the propeller. This a common practice for airplanes with fixed pitch propellers, however, it is not recommended for constant speed propellers, since pressures applied to the propeller blades are transmitted to moving parts within the propeller hub. Over time, these forces could cause damage to the propeller.

09-3. TAXIING

a. When taxiing, minimize the use of the brakes. Since the airplane has a free castoring nose wheel, steering is accomplished with light breaking. Avoid the tendency to ride the brakes by making light steering corrections as required and then allowing the feet to slide off the brakes and the heels to touch the floor.

b. Avoid taxiing in areas of loose gravel, small rocks, etc., since it can cause abrasion and damage to the propeller. If it is necessary to taxi in these areas, maintain low propeller speeds. If taxiing from a hard surface through a small area of gravel, obtain momentum before reaching the gravel.

c. The airplane is stable on the ground. The low wing design minimizes the tipping tendency from strong winds while taxiing. Still, the proper positioning of control surfaces during taxiing will improve ground stability in high wind conditions. The following table summarizes control positions that should be maintained for a given wind component.

Wind Component	Aileron Position	Elevator Position
Left Quartering Headwind	Left Wing Aileron Up (Move Aileron Control to the Left)	Neutral (Hold Elevator Control in Neutral Position)
Right Quartering Headwind	Right Wing Aileron Up (Move Aileron Control to the Right)	Neutral (Hold Elevator Control in Neutral Position)
Left Quartering Tailwind	Left Wing Aileron Down (Move Aileron Control to the Right)	Down Elevator (Move Elevator Control Forward)
Right Quartering Tailwind	Right Wing Aileron Down (Move Aileron Control to the Left)	Down Elevator (Move Elevator Control Forward)

Figure 9 - 1 Control Positions for Wind Components

CHAPTER 10

PARKING, MOORING, STORAGE, AND RETURN TO SERVICE

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10-1. PARKING

a. During parking operations, it is best to head the airplane into the wind if possible. Normally, setting the parking brake is recommended, however, there are two situations where doing so is not a good idea.

1. Do not set the parking brake if the brakes are overheated, which might result from a short field landing or extensive taxiing. It is best to not set the parking brake until the brakes have had a sufficient cooling period. A brake pad clamped to a hot chrome disk can cause uneven cooling of the brake disc, which could potentially warp it.
2. Do not set the parking brake in cold weather. Accumulations of freezing rain, ice, and snow can freeze-weld the brake pad to the disk. Landing or taxiing in standing water at near-freezing temperatures can cause similar problems if the parking brake is set when the airplane is parked.

10-2. SECURING THE AIRPLANE

a. In any event, whether the brakes are set or not set, the airplane should be chocked and the following items should be accomplished to secure the airplane.

b. Install the control lock.

c. Chock the main gear tires with chocks on both sides of each tire.

d. Attach a rope or chain to each tie-down point, and secure the rope or chain to a ramp tie-down point. There are three tie-down points, one on each wing and one on the tail. The ropes or chains should have a tensile strength of at least 750 lbs.

e. Install the pitot tube cover.

10-3. STORAGE

a. **Overview** – The storage of an airplane mostly deals with engine related items. Very little needs to or can be done to preserve the airframe, particularly for flyable and temporary storage. The best protection for the exterior is, of course, to hangar the airplane. If the airplane cannot be hangared, then a coat of wax using the material and techniques described in Section 8 of the *Pilot's Operating Handbook* should be applied to all exterior surfaces. In addition, all typical items associated with securing the airplane should be done. These include: (1) installing the pitot tube cover, (2) chocking all wheels and tying the airplane down with the parking brakes released, (3) installing the control lock, (4) topping off the fuel tanks, (5) cleaning the bolts and nuts on the brakes and applying a non-stick preservative like graphite or silicone, and (6) installing other owner-option protection devices. There are three types of storage categories, flyable, temporary, and indefinite. The time period and applicable storage procedure for each type is discussed below.

b. **Flyable Storage (7 to 30 days)** – If the airplane is to be maintained in flyable storage, then it should be flown for a minimum of 30 minutes every 30 days; ground running the engine is <u>not</u> a substitute for flying the airplane. During flyable storage, the propeller should be rotated by hand every seven days. This operation should include at least six complete revolutions of the engine. Stop the propeller 45º to 90º from its original position. For maximum safety use the following procedures:

1. Ensure that the ignition switch is set to the OFF position.
2. Set the throttle to the CLOSED position.

3. Set the mixture to IDLE CUT OFF.
4. Set the parking brake and chock the wheels.
5. Ensure that airplane tie-downs are secure.
6. Open the cabin door on the pilot's side of the airplane.
7. Always assume the propeller could start when moving it manually and use an appropriate technique for hand turning the propeller.
8. Release the parking brake when the operation is completed.

WARNING

Always assume that the engine could start when rotating the propeller by hand. Remain clear of the arc of the propeller blades at all times.

c. **Temporary Storage (up to 90 days)** – Use the following procedures to preserve the engine for temporary storage. See the Airframe Preservation for Temporary or Indefinite Storage heading in paragraph e below for airframe preservation items.

1. Remove the top spark plug from each of the six cylinders and apply an atomized injection of preservation oil, MIL-L-46002, Grade 1. As the oil is injected into each cylinder, the piston should be near bottom dead center, and the preservation operation should be done at room temperature.
2. When Step 1 is complete, and with none of the pistons at dead center, re-spray each cylinder thoroughly making sure to cover all interior surfaces.
3. Install spark plugs.
4. Spray approximately two ounces of preservation oil through the oil filler tube.
5. Seal all engine openings exposed to the atmosphere with suitable plugs or moisture resistant tape.
6. Tag engine, cowling, and other appropriate areas with the statement, "Do not turn propeller, engine preserved."

d. **Indefinite Storage (Over 90 Days)** – If the airplane is to be stored for a long period, follow the procedures listed below to preserve the engine. See the Airframe Preservation for Temporary or Indefinite Storage heading in paragraph e below for airframe preservation items.

1. Drain the engine oil and refill with MIL-C-6529 Type II preservation oil. Start the engine and operate until normal temperature ranges are achieved. Fly the airplane for about 30 minutes and then allow the engine to cool to the ambient temperature.
2. Follow steps 1, 2, and 4 above for Temporary Storage.
3. Install dehydrator plugs MS27215-1 or -2, in each of the top spark plug holes. Ensure the dehydrator plug is blue when installed. Protect and support the spark plug leads with AN-4060 protectors.
4. Place a bag of desiccant in the exhaust pipes and seal the openings with moisture resistant tape.
5. Seal the induction system with moisture resistant tape.
6. Seal the engine breather by taping a dehydrator plug, M527215-2, in the lower end. Seal the "whistle hole" vent in the breather tube with moisture resistant tape.
7. Tag engine, cowling, and other appropriate areas with the statement, "Do not turn propeller, engine preserved."

8. Install plugs in engine cowl inlets, wing leading edge air inlets, and all other openings. **Do not plug or seal tank vents on the bottom of each wing.**

NOTE

During the storage period, FAA Airworthiness Directives and manufacturer's service bulletins may apply which require action based on calendar dates, not operating hours. These items must still be complied with even though the airplane is in storage.

NOTE

The dehydrator plugs must be visually checked every 30 days to verify that the color has not changed. Bad dehydrator plugs should be replaced. If more than half of the plugs change color, the bad plugs and all the desiccant bags on the engine should be replaced.

Every six months the dehydrator plugs should be replaced and the cylinders re-sprayed with preservation oil. When removing the plugs, check the cylinder interior. If rust stains are noted, spray the cylinder with preservation oil, turn the prop through six revolutions, and then re-spray all cylinders.

e. **Airframe Preservation for Temporary and Indefinite Storage** – If the airplane is to be stored for over 30 days, some or all the procedures below may be applicable, depending on the anticipated storage time period.

1. Ensure the tires are free of grease, oil, tar, and gasoline. The presence of these items accelerates the aging process. Sunlight and static electricity convert oxygen to ozone, a substance that accelerates the aging process. Special tire covers can be installed which retard the erosion process.
2. It is best if the weight of the airplane is removed from the tires to prevent flat spots. If the airplane cannot be blocked or set on jacks, then every 30 days each wheel should be rotated about 90º to expose a new tire pressure point.
3. If the airplane does not have a recent coat of wax, a new coat should be applied as discussed in Chapter 12 of this manual.
4. Lubricate exposed exterior metal fittings, hinges, push rods, etc. Use plugs or moisture resistant tape to seal all openings except fuel vent holes and drain holes.
5. Remove the battery and store in a cool, dry location. The battery may need periodic servicing and recharging depending on the storage period.
6. Prominently tag areas where tape and plugs are installed.

10-4. INSPECTIONS DURING TEMPORARY STORAGE

a. The following inspections should be performed while the airplane is in temporary storage.

1. Check the cleanliness of the airframe as frequently as possible and remove any dust that has collected.
2. Check the condition and durability of the protective wax coat and renew as required.
3. Every 30 days, check the interior of at least one cylinder for evidence of corrosion.

10-5. INSPECTIONS DURING INDEFINITE STORAGE

a. The following inspections should be performed while the airplane is in indefinite storage.
   1. Check the condition of the dehydrator plugs ever 30 days to verify that the color has not changed. Bad dehydrator plugs should be replaced. If more than half of the plugs change color, the bad plugs and all the desiccant bags on the engine should be replaced.
   2. Every six months the dehydrator plugs should be replaced and the cylinders re-sprayed with preservation oil. When removing the plugs, check the cylinder interior. If rust stains are noted, spray the cylinder with preservation oil, turn the prop through six revolutions, and then re-spray all cylinders.

NOTE

When an airplane has been in storage for a long period, the date of the required annual inspection may have passed. There is no requirement to perform this inspection during the temporary or indefinite storage period. However, the inspection must be completed before the airplane is returned to service.

10-6. RETURN TO SERVICE FROM TEMPORARY STORAGE

a. To return an airplane that has been in temporary storage to active service, perform the following steps.

1. Remove seal plugs, tape, and all methods of tagging the airplane, including items tagged on the airframe.
2. Remove the bottom spark plug from each of the six cylinders and rotate the propeller several times to remove the preservation oil.
3. Reinstall the spark plugs according to manufacturer's recommendations.
4. Conduct a normal engine start, and idle the engine for several minutes until oil temperature is within normal limits. Monitor engine instruments to ensure they are within normal operating ranges.
5. Stop the engine and inspect the entire airplane before test flying.
6. Test fly the airplane.

10-7. RETURN TO SERVICE FROM INDEFINITE STORAGE

a. To return an airplane that has been in indefinite storage to active service, perform the following steps.

1. Remove all dehydrator plugs, seal plugs, tape, and all methods of tagging the airplane including items tagged on the airframe.
2. Drain the preservation oil, and service the airplane engine with the recommended lubricating oil.
3. Remove the bottom spark plugs from each of the six cylinders, and rotate the propeller several times to remove the preservation oil.
4. Re-install the spark plugs, and carefully rotate the propeller by hand several times to check for possible liquid lock.
5. Conduct a normal engine start, and idle the airplane for several minutes until the oil temperature is within normal limits. Monitor all engine instruments to ensure they are within normal operating ranges.
6. Stop the engine and inspect the entire airplane before test flying.
7. Test fly the airplane.

10-8. RETURN TO SERVICE AIRFRAME ITEMS

a. To return the airframe portion of an airplane that has been in temporary or indefinite storage to active service, perform the following steps, as applicable.

1. Remove all methods of tagging and sealing the airplane including any items on or in the engine area.
2. Remove tire covers or other protective devices. Check the condition of the tires, and service to proper pressures. Cracked, deformed, and desiccated tires should be replaced.
3. Thoroughly clean the exterior of the airplane, including the transparencies. If necessary, renew the protective wax coat. See Chapter 12 for instructions on care of the airframe.
4. Check the condition and charge of the battery. If the battery is still serviceable, reinstall it in the airplane; otherwise, install a new battery.

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CHAPTER 11

PLACARDS & MARKINGS

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Chapter 11

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11-1. GENERAL

a. The various airplane placards on the interior and exterior of the airplane are required items for the continued airworthiness of the airplane. The placards must be inspected at the required inspection intervals for cleanliness, legibility, and security. The following pages contain the text and location of all required interior and exterior placards. In addition, procedures for removal, replacement, and care are included.

11-2. REPLACING WORN, DAMAGED, OR MISSING PLACARDS

a. If necessary, refer to Chapter 11 in the airplane parts catalog to determine the part number for the affected placard.

b. Use the following steps to remove and replace a placard.

1. Apply the placard in a relatively warm area. It is best if the ambient air temperature and the temperature of the surface to which the placard is applied is above 70ºF.
2. Make a few reference marks around the old placard to aid in positioning the new one. If the placard is missing, refer to paragraphs 11-4 and 11-5 for the location, and then apply reference marks.
3. Carefully remove the existing placard by raising one corner and slowly peeling it back. **Do not pull <u>out</u> on the placard**! Pull in a manner that keeps the peeled portion near or parallel to the surface. It helps to apply primer remover between the glue on the back of the placard and the attachment surface.
4. Do not remove the placard too quickly. A slow steady pull works best. Discard the placard once it is removed.
5. Clean excess adhesive off of the attachment surface with PPG D837 spirit wipe.
6. When the attachment surface is clean and dry, remove the paper backing on the new placard, and apply it using the reference marks described in paragraph 2 above.
7. Attach one end of the placard to the attachment surface, and slowly roll it on in a manner that avoids air bubbles. Because the airplane's surface has no rivets, air bubbles are infrequent.
8. If the placard has a protective covering remove it at this point.
9. If an air bubble does occur, prick the bubble with a small diameter needle. The number of holes depends on the size of the bubble. Five to ten needle holes are usually sufficient. Slowly work the placard until all air is removed.

11-3. PLACARD CARE

a. Wash the placard with mild soap and water. Do not use petroleum based, caustic, or abrasive products. Avoid applying wax to placards because it may contain certain amounts of petroleum-based substances.

b. If there is any question about the composition of the cleaner being used, apply a small amount to a test corner of the placard, and wait a few minutes to see if it produces adverse results.

c. If aviation fuel accidentally comes in contact with a placard, immediately rinse with water.

d. Never apply paint to or paint over a placard.

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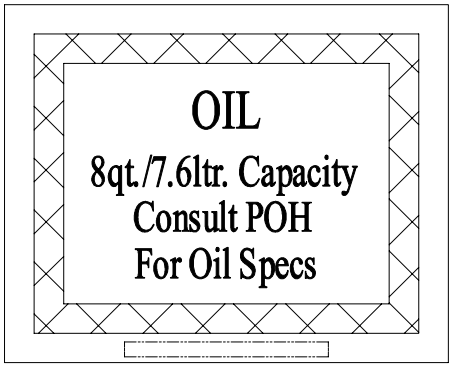
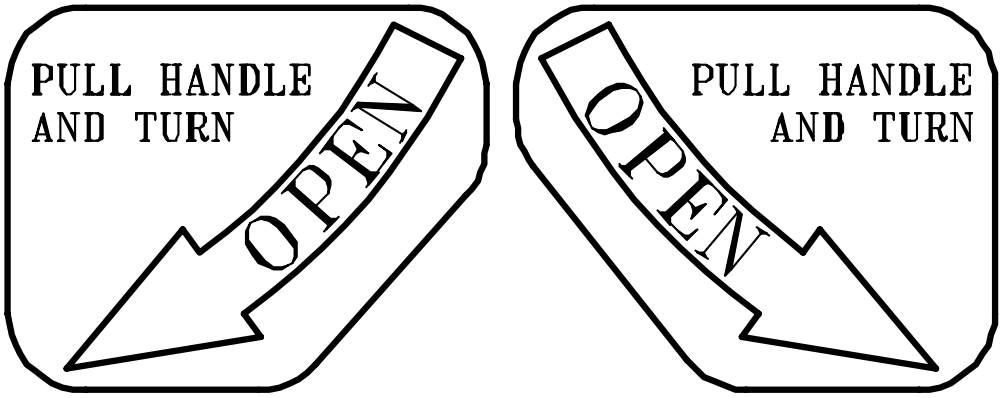
11-4. EXTERIOR PLACARDS

On Oil Filler Access Door



Near Pilot and Passenger Exterior Door Handles



On Main Wheel Fairing

MAIN 55 psi

On Nose Wheel Fairings

NOSE 88 PSI

On Flaps Near Wing Root (Both Sides)

NO STEP

Near Fill Cap Of Fuel Tank

S/N 42001 to 42567

AVGAS ONLY
MIN FUEL GRADE 100 / 100LL
TOTAL USEABLE 49 GAL US / 185 LTR
TOTAL CAPACITY 53 GAL US / 201 LTR

S/N 42568 and on

AVGAS ONLY
MIN FUEL GRADE 100 / 100 LL
TOTAL STANDARD USEABLE 43 GAL US / 163 L
TOTAL LONG RANGE USEABLE 51 GAL US / 193 L
TOTAL CAPACITY 53 GAL US / 201 L

Under Each Wing Near Fuel Drains

FOR DRAINING OF WING FUEL SUMP:
TO OPEN: PRESS CUP GENTLY INTO BOTTOM OF VALVE TO DRAIN REQUIRED AMOUNT OF FUEL.
TO CLOSE: REMOVE CUP AND VALVE WILL CLOSE.

TO DRAIN WING TANKS: REFER TO MAINTENANCE MANUAL.

Under Left Side Wing Fillet Near Ground Power Supply Plug (S/N 42001 to 42500)

12 VDC POWER
CONNECTION

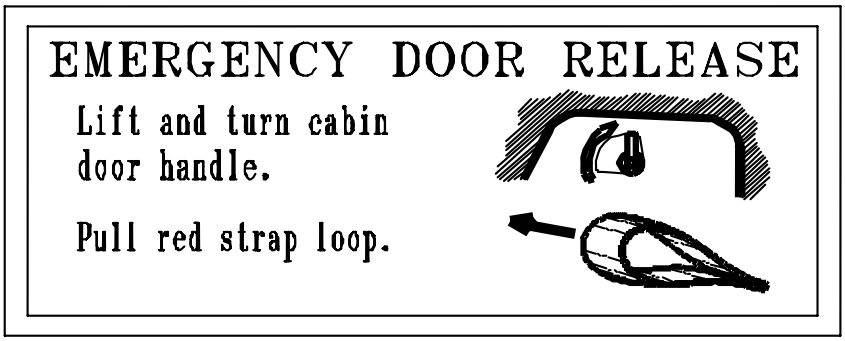
Left Underside of Fuselage on Ground Power Supply Plug Cover (S/N 42501 and on)

24 VDC POWER
CONNECTION

On Exterior of Gascolator Door (Underside of Fuselage)

EMERGENCY
DOOR RELEASE
PRESS BUTTONS TO OPEN

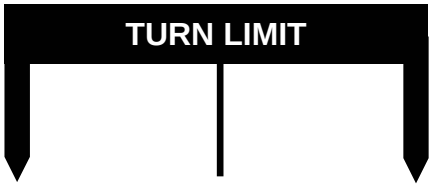
On Interior of Gascolator Door



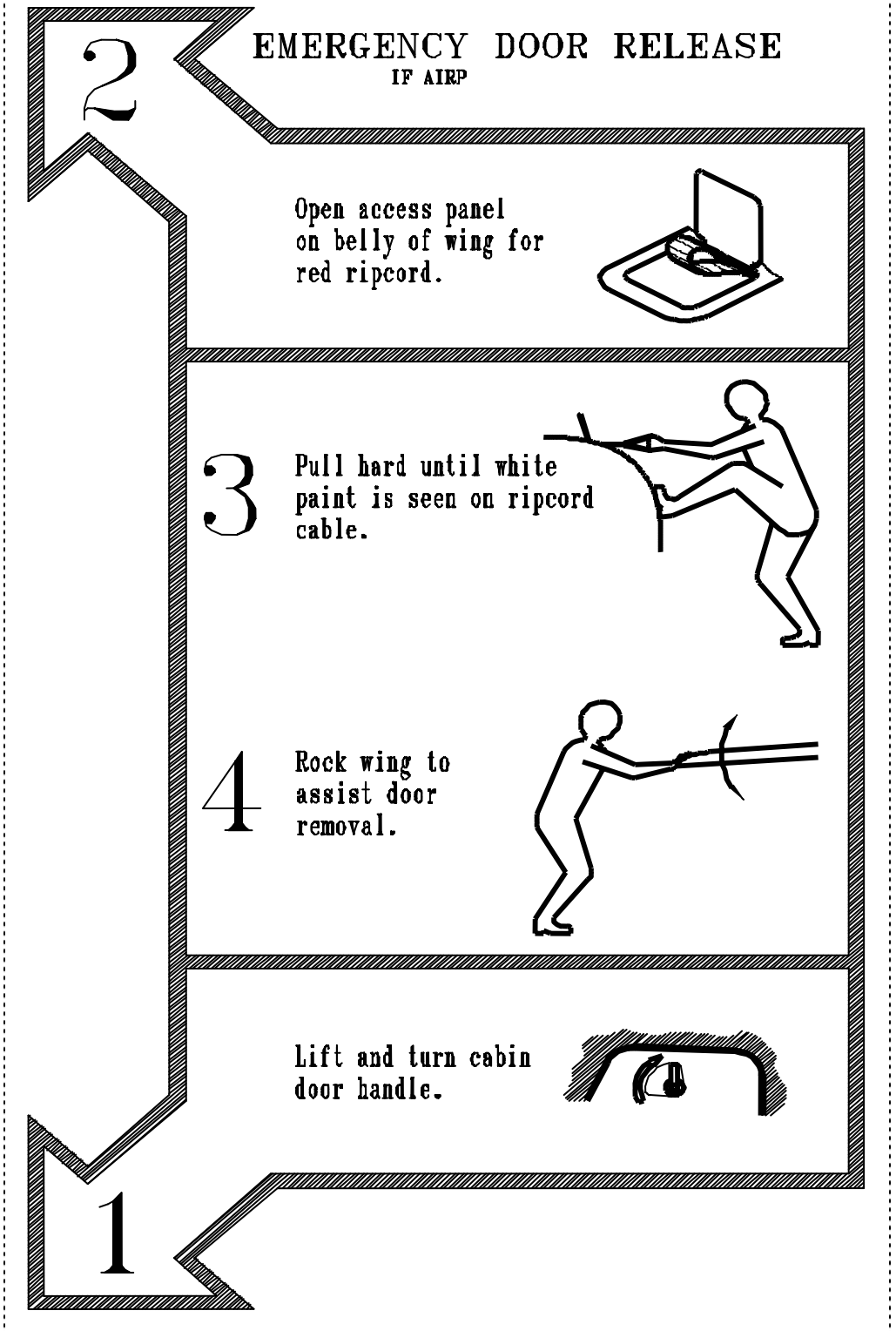
On Exterior of Fuselage – Forward of Wing on Copilot's Side

On Top of Nose Wheel Fairing – (Pointing Aft)

On Forward Portion of Nose Gear Fairing



On Exterior of Fuselage – Forward of Wing on Pilot's Side



11-5. INTERIOR PLACARDS

On Center Console Below Radios

FIRE EXTINGUISHER LOCATED UNDER CO-PILOT'S SEAT

The markings and placards installed in this airplane contain operating limitations that must be complied with when operating this airplane in the Utility category. Other operating limitations that must be complied with when operating this airplane in this category are contained in the Airplane Flight Manual.

Utility Category – No acrobatic maneuvers approved, except those listed in the Pilot's Operating Handbook.

FLIGHT INTO KNOWN ICING PROHIBITED.

SPINS PROHIBITED.

APPROVED FOR DAY/NIGHT – VFR/IFR.

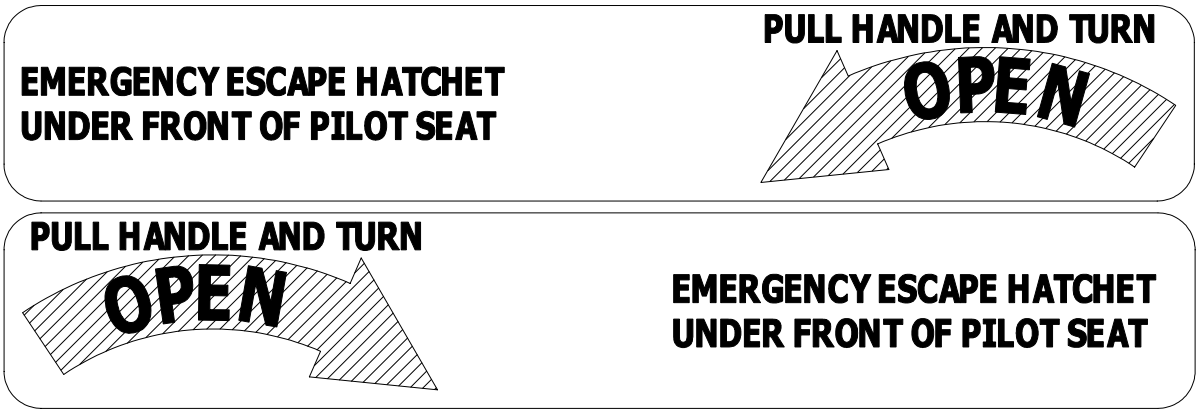
NO SMOKING

On Left Panel Behind Pilot's Control Stick (Garmin G1000 Option)

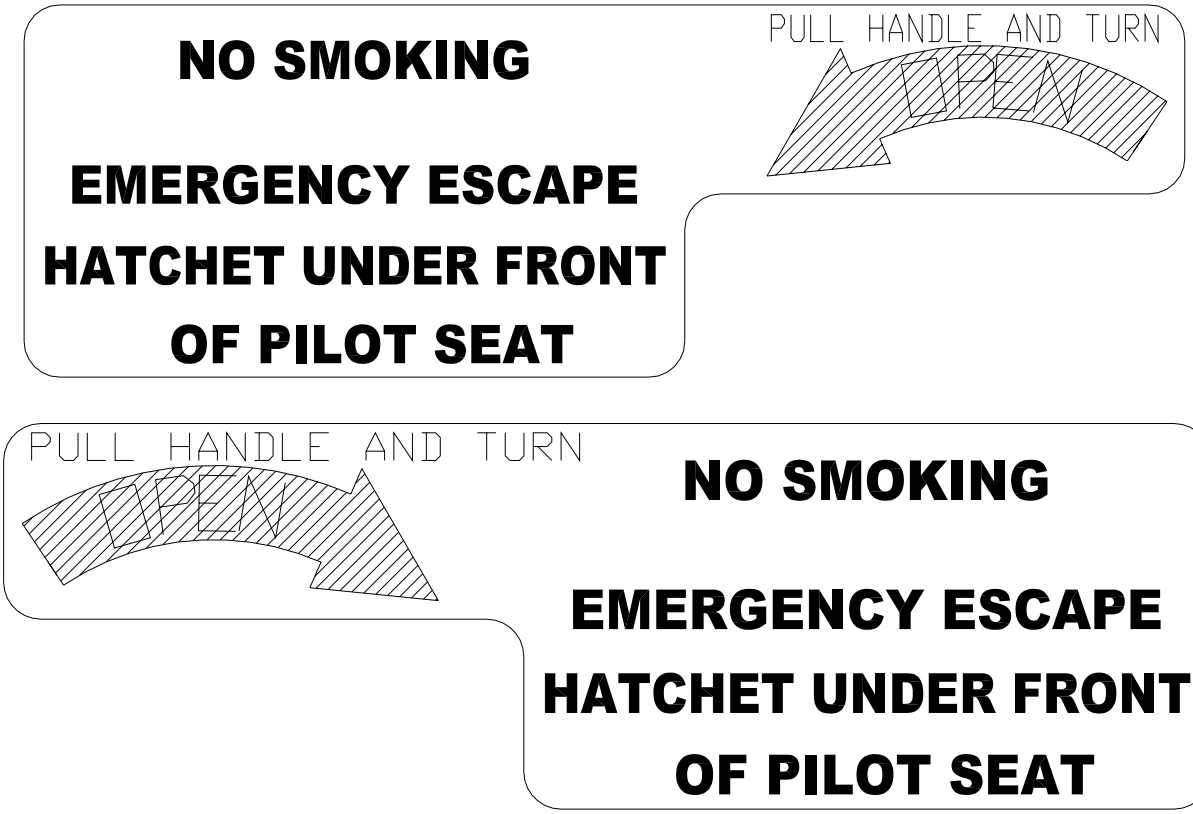
On Instrument Panel to Left of Backup Attitude Indicator (Garmin G1000 Option)

MANEUVERING SPEED
148 KIAS (3400 LBS)
127 KIAS (2500 LBS)

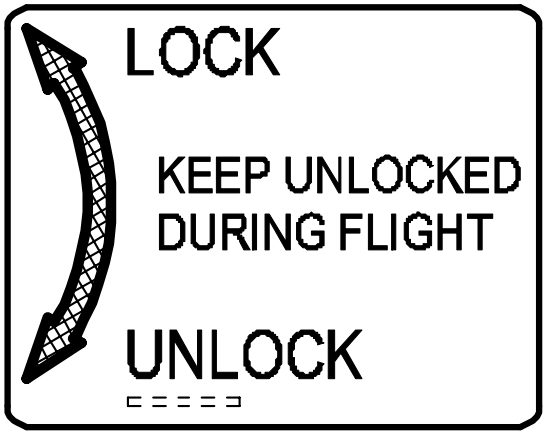
Near Pilot and Copilot Interior Door Handles (S/N42001 to 42005)



Near Pilot and Copilot Interior Door Handles (S/N 42006 and on)



Near Interior Door Handle on Passenger Side



On Bottom of Baggage Compartment Door Opening

TOTAL BAGGAGE WEIGHT 120 LBS., SHELF WEIGHT 20 LBS. MAX.
SEE WEIGHT AND BALANCE DATA FOR ADDITIONAL LOADING INSTRUCTIONS

or

TOTAL BAGGAGE WEIGHT 120 LBS. (54 kg.), SHELF WEIGHT 20 LBS. (9 kg.), MAX.
SEE WEIGHT AND BALANCE DATA FOR ADDITIONAL LOADING INSTRUCTIONS

On The Front of the Pilot's Seat Base On Escape Hatchet

IN EMERGENCY
BREAK DOOR WINDOW

In Aft Cabin on Aft Baggage Bulkhead

EMERGENCY LOCATION TRANSMITTER
LOCATED AFT OF THIS POINT. IT MUST
BE MAINTAINED IN ACCORDANCE WITH
THE FEDERAL AVIATION REGULATIONS

Under Left Rear Seat Next to Leveling Washer

AIRCRAFT LEVELING WASHER.
DO NOT REMOVE WASHER.

Under All Seats

SEAT BOTTOM CAVITY
MUST REMAIN CLEAR

On Parking Brake Handle

Near Airspeed Indicator (Basic or Avidyne Option)

MANEUVERING SPEED: **148 KIAS (3400 LBS)**
127 KIAS (2500 LBS)

Near Airspeed Indicator (Basic or Avidyne Option)

ABOVE 12,000 FT (PA) REDUCE: V_{NE} **5 KIAS/1000 FT**
V_{NO} **4 KIAS/1000 FT**

Near Manifold Pressure Gauge (Basic or Avidyne Option)

DO NOT EXCEED 20"
MANIFOLD PRESSURE
BELOW 2200 RPM

On Center Overhead Console (Basic or Avidyne Option)
On Compass (Garmin G1000 Option)

Without Electric A/C

For	N	30	60	E	120	150
Steer						
For	S	210	240	W	300	330
Steer						
Radios: On Strobes: On				DATE __/__/__		

With Electric A/C

Deviation > 10° with air conditioning on						
For	N	30	60	E	120	150
Steer						
For	S	210	240	W	300	330
Steer						
Radios: On Strobes: On				DATE __/__/__		

The magnetic direction indicator is calibrated for level flight with the engine, radios, and strobes operating.

Above Copilot's Fresh Air Vent (Basic or Avidyne Option)

Acrobatic Maneuvers:
Chandelles.....150 KIAS entry speed
Lazy Eights....150 KIAS entry speed
Steep Turns....150 KIAS entry speed
Stalls.....Slow Deceleration
SPINS PROHIBITED

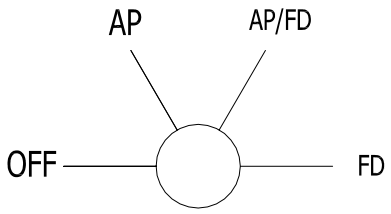
On Right Panel Behind Copilot's Control Stick (Garmin G1000 Option)

Acrobatic Maneuvers:
Chandelles.....150 KIAS entry speed
Lazy Eights....150 KIAS entry speed
Steep Turns....150 KIAS entry speed
Stalls.....Slow Deceleration
SPINS PROHIBITED

On Top Center of Engine Instrument Panel (when autopilot installed) (Basic or Avidyne Option)

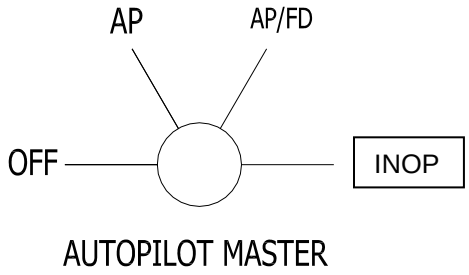
AUTOPILOT OPERATION - SPEEDS/ALTITUDE
MIN OPERATING: 90 KIAS
MAX OPERATING: 210 KIAS
MIN ALTITUDE IN APPROACH: 400 FT. AGL

On Top Center of Flight Instrument Panel (when autopilot installed and FD Only Mode enabled) (Basic or Avidyne Option)



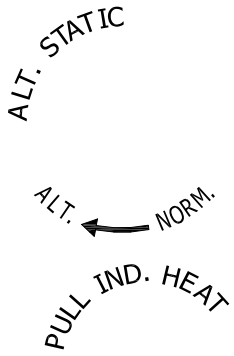
AUTOPILOT MASTER

On Top Center of Flight Instrument Panel (when autopilot installed and FD Only Mode disabled) (Basic or Avidyne Option)



Near the Left Dimmer Switch on the Pilot's Knee Bolster (Basic or Avidyne Option)

On the Upper Left Side of the Tower Assembly (Garmin G1000 Option)



On the Back Lower Portion of the Front Seat Headrests (S/N 42001 to 42005)
(Embroidered into the leather with red stitching)

NO SMOKING

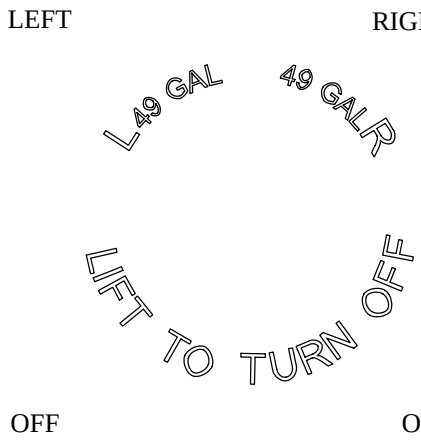
EMERGENCY ESCAPE HATCHET
UNDER FRONT OF PILOT SEAT

On Top Front of Center Console (Garmin G1000 Option)

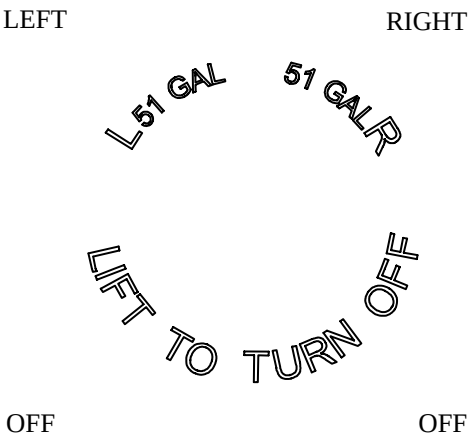
PANEL LIGHT DIMMERS
LOCATED FRONT CONSOLE

Engraved On Fuel Selector Knob and Upper Plate

S/N 42001 to 42567



S/N 42568 and on



On Engine Instrument Panel Above Fuel Gauge (Basic or Avidyne Option)

MAXIMUM FUEL IMBALANCE
NOT TO EXCEED 10 GAL

On Engine Instrument Panel Near EGT/CHT
(when optional JPI digital engine scanner is installed)

MAX
CHT
460°F

On Left Center of Engine Instrument Panel
(when optional CO detector is installed) (Basic or Avidyne Option)

CO
DETECT

TEST/RESET

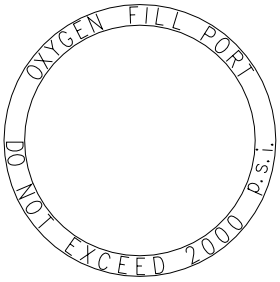
On Oxygen Distribution Manifold (if present) in Forward Overhead Panel

OFF

OXYGEN SHUTOFF

ON

On Oxygen Fill Port (if present) set into Hat Shelf



On Air Conditioning System Bay Access Cover

UPON REINSTALLATION ENSURE THIS ACCESS PANEL IS SEALED TO PREVENT CARBON MONOXIDE FROM ENTERING THE CABIN

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CHAPTER 12

SERVICING

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12-1. SERVICING

a. There are recommended and required servicing intervals specified in Chapter 5 of this maintenance manual, beginning with paragraph 05-3. It is important to note that the servicing items for the recommended and required inspection intervals are based on typical operations. If the owner or operator of the airplane uses the airplane on an ongoing basis in an atypical environment, then the servicing procedures must be modified accordingly. These environmental conditions might include extremes in ambient temperature or humidity, operation in areas of high dust or sand, or unusual operating requirements, to mention a few.

b. In general, minimal servicing of the airplane is required between required maintenance inspection other than fuel, oil, and maintenance of proper tire pressures. In some instances, the nose strut may require unscheduled service, depending on how the airplane is used.

12-2. REFUELING AND DEFUELING THE AIRPLANE PROCEDURES

a. The high-speed characteristics of the airplane make generation of static electricity more likely, so it is important for the airplane to be grounded to the fuel source during refueling and defueling operations. Place the fuel source grounding clamp on the right or left exhaust stack of the airplane before touching the filler neck of the fuel tanks with metal parts of the ground refueling/defueling equipment. Remember that refueling is often done at the conclusion of a flight, and the exhaust stacks may still be hot, so care must be used when attaching the clamp.

b. Some defueling is possible using the defueling feature on the delivery system of the Avgas fuel supplier. This procedure is usually adequate for removing fuel when gross takeoff weight is an issue. To completely defuel the airplane use the following steps.

1. Select a well-ventilated area for the defueling operation.
2. Remove all fuel possible from the tank or tanks using the defueling feature on the Avgas delivery system.
3. Disconnect the main fuel line downstream of the auxiliary fuel pump, at the firewall.
4. Connect a flexible hose to the fuel fitting, and direct the hose into a fuel receptacle.
5. Set the fuel selector to either the left or right tank, depending on which tank is to be defueled.
6. Operate the auxiliary boost in low until the tank is drained. When the pump first starts to cavitate, turn it off, and attempt no further defueling with the auxiliary boost pump.

12-3. PROCEDURES FOR DRAINING THE FUEL SYSTEM

a. Using the procedures described in paragraph 1 through 6 will leave approximately three to four gallons in the fuel tank. If further draining is required to work on a fuel tank or to establish the empty weight with unusable fuel (see paragraph 1), use the following procedure.

1. Select a well-ventilated area for draining fuel from the tank.
2. Create a small (2 in. diameter) dam around the wing tank drain with aluminum foil tape (sometime referred to as speed tape) or a similar substance.
3. Place a container with a large opening under the fuel drain.
4. Set the wing fuel drain to the locked-open position, and drain until fuel stops running.
5. If all fuel is to be drained from the airplane, perform the above procedure for both tanks, and then drain the gascolator.

6. To establish the airplane's empty weight with unusable fuel, complete steps 1 through 4 for each tank. At this point, approximately 4 to 8 ounces of undrainable fuel will remain in each tank. Ensure the fuel drains are closed, and then add three gallons of fuel to each tank.

12-4. FUEL DRAINS

a. The inboard section of each tank contains a fuel drain near the lowest point in each tank. The fuel drain can be opened intermittently for a small sample, or it can be locked open to remove a large quantity of fuel. The gascolator, or fuel strainer, is located under the fuselage, on the left side, near the wing saddle. The gascolator is a conventional drain device that operates by pushing up on the valve stem. There is an internal bypass in the strainer that routes fuel around the filter if it becomes clogged.

12-5. FUEL CONTAMINATION

a. To test for fuel contamination, fuel samples must be taken from each of the wing drains and from the gascolator before each flight and after the airplane is refueled. Wait approximately 10 minutes before taking a sample after refueling to allow water and other contaminates to settle to the bottom of the tank. There are three types of contaminates that can inadvertently be introduced to the fuel system: (1) sediment such as dirt and bacteria, (2) water, and (3) the improper grade of fuel.

b. The accumulation of sediments is an inherent issue with most aircraft and can never be completely eliminated. Refueling the airplane at the conclusion of each flight and using fuel from a supplier who routinely maintains the filtration of the refueling equipment will lessen the problem somewhat. If specks are observed in the fuel sampler, continue the sampling operation until no debris is observed. Be sure the sampling device is clean before using it.

c. The two more common sources of water contamination are condensation of water from the air within a partially filled fuel tank and water-contaminated Avgas from a fuel supplier. Again, refueling after each flight and proper filtration of the fuel delivery system will mitigate water contamination. Water, which is heavier than Avgas, will collect near the bottom of the sampling device. If water is observed in the fuel sampler, take additional fuel samples until all the water is removed.

d. Aviation fuel is dyed according to its grade, and on new aircraft, like the Cessna 350 (LC42-550FG), the filler neck is sized to only accept fuel of the proper grade. Still, the color of the fuel must be verified according to the specifications in paragraph 12-5, since the fuel truck might have been refilled improperly. If fuels of two different grades are mixed, the fuel sample will be clear. If an inferior, improper grade of fuel is noted, completely defuel and drain both tanks, and refuel with the proper grade of Avgas.

12-6. FUEL SERVICING AND RECOMMENDED FUEL GRADES

a. If possible and practicable, the airplane should be serviced at the conclusion of each flight. This procedure will significantly mitigate water accumulation in the fuel tanks. The following fuel grades are approved.

1. 100LL Grade Aviation Fuel (Blue)
2. 100 Grade Aviation Fuel (Green)

12-7. FUEL CAPACITIES

a. Total Capacity: 106 Gallons US (401 L)
b. Total Capacity Each tank: 53 Gallons US (201 L)
c. Total Usable Fuel:
   1. S/N 42001 to 42567: 49 Gallons US (186 L)/tank, 98 Gallons US 371 L) Total
   2. S/N 42568 and on:
      (a) Standard: 43 Gallons US (163 L)/tank, 86 Gallons US (326 L) Total
      (b) Long Range: 51 Gallons US (193 L)/tank, 102 Gallons US (386 L) Total

12-8. APPROVED FUEL ADDITIVES

a. Under certain ambient conditions of temperature and humidity, water can be supported in the fuel in sufficient quantities to create restrictive ice formation along various segments of the fuel system. To alleviate the possibility of this occurring, it is permissible to add Isopropyl Alcohol to the fuel supply in quantities not to exceed 3% of the total. In addition, ethylene glycol monomethyl ether (EGME) and diethylene glycol monomethyl ether (DiEGME) compounds in accordance with military specification MIL-I-27686E may be added for this purpose. The EGME and DiEGME compounds must be carefully mixed with fuel concentrations not to exceed 0.15 percent by volume.

b. It is important that the approved fuel additives are mixed in correct proportions. Consideration is required to ensure the appropriate concentration levels are achieved when the tank is filled. For example, adding 40 gallons of fuel with a 0.15 percent concentration of EGME to a tank with 10 gallons of untreated fuel will produce a mixture of something less than 0.15 percent. Consideration must also be made for the unusable fuel in the tank since it will be combined with the total mixture.

c. The additives shall be added as the fuel is introduced to the fuel tank so that the mixture is properly combined. Alternatively, the additive can be mixed with a small amount of fuel in a separate container, such as a five-gallon can, and added to the fuel tank before normal fueling. The table in paragraph 12-8 lists the number of ounces of each additive for a given fuel quantity.

12-9. FUEL ADDITIVE MIXTURE TABLE

Fuel (Gallons)	Isopropyl Alcohol (3%) Fluid Ounces	EGME & Di-EGME (0.15%) Fluid Ounces	Fuel (Gallons)	Isopropyl Alcohol (3%) Fluid Ounces	EGME & Di-EGME (0.15%) Fluid Ounces
1	3.8	0.2	27	103.7	5.2
2	7.7	0.4	28	107.5	5.4
3	11.5	0.6	29	111.4	5.6
4	15.4	0.8	30	115.2	5.8
5	19.2	1.0	31	119.0	6.0
6	23.0	1.2	32	122.9	6.1
7	26.9	1.3	33	126.7	6.3
8	30.7	1.5	34	130.6	6.5
9	34.6	1.7	35	134.4	6.7
10	38.4	1.9	36	138.2	6.9
11	42.2	2.1	37	142.1	7.1
12	46.1	2.3	38	145.9	7.3
13	49.9	2.5	39	149.8	7.5
14	53.8	2.7	40	153.6	7.7
15	57.6	2.9	41	157.4	7.9
16	61.4	3.1	42	161.3	8.1
17	65.3	3.3	43	165.1	8.3
18	69.1	3.5	44	169.0	8.4
19	73.0	3.6	45	172.8	8.6
20	76.8	3.8	46	176.6	8.8
21	80.6	4.0	47	180.5	9.0
22	84.5	4.2	48	184.3	9.2
23	88.3	4.4	49	188.2	9.4
24	92.2	4.6	50	192.0	9.6
25	96.0	4.8	51	195.8	9.8
26	99.8	5.0	52	199.7	10.0

WARNING

Proper mixing of EGME and DiEGME compounds is extremely important because concentrations more than 0.15 percent by volume can have a harmful effect on engine components.

12-10. ENGINE OIL CAPACITY

a. The dipstick and oil filler cap access door are located on the top left engine cowl about two feet from the propeller hub. The engine must not be operated with less than six quarts of oil and must not be filled above eight quarts. For extended flights, the oil should be brought up to full capacity.

b. The engine drain-refill capacity is eight (8) quarts of oil. The airplane engine holds approximately 11 quarts of oil, however, about 3 quarts are for the oil cooler, oil filter, and propeller governor and cannot be drained from the system. Before checking the oil on a hot engine, ensure sufficient time has elapsed to allow the oil to drain back into the sump.

12-11. RECOMMENDED OIL FOR ENGINE BREAK-IN PERIOD

a. During the engine break-in period, straight mineral oil must be used for the first 25 hours unless the oil consumption has not stabilized. If the oil consumption has stabilized after the first 25 hours of the airplane's time in service, the oil and oil filter must be changed and a new supply of Teledyne Continental Motors specification MHS-24 (latest revision) ashless dispersant oil must be used. At 50 hours of time in service, the oil and oil filter shall be changed and the filter and discarded oil checked for evidence of metal particles. Thereafter, the oil and oil filter must be changed at every 100 hours of time in service.

b. If after the initial 25 hour break-in period the oil consumption has not stabilized, use the following procedure.
   1. Inspect and replace the oil filter.
   2. Drain and inspect the engine oil.
   3. Install a fresh supply of approved mineral oil.
   4. Continue to monitor oil consumption.
   5. When oil consumption stabilizes, remove, inspect, and replace the oil filter. Drain and inspect the engine oil, and add a new supply of Teledyne Continental Motors (TCM) specification MHS-24 (latest revision) ashless dispersant oil.

c. In the extremely unlikely event the oil consumption has not stabilized at 50 hours of time in service, the cylinders may need deglazing or some other problem may exist. In either event the issue should be referred to TCM.

12-12. OIL SERVICING

a. The oil grades shown below in paragraph 12-13 are recommended after the initial engine break-in period. Only lubricant oils conforming to Teledyne Continental Motors Specification MHS-24D (latest revision) can be used. Note, the approval for use of MHS-25 synthetic oils has been removed. Each time the oil and the oil filter are changed, the discarded filter and oil must be checked for evidence of metal particles. See Chapter 79 for oil draining instructions.

12-13. RECOMMENDED OIL GRADES

a. The following oil grades are recommended for the two ambient temperature ranges shown below.
   1. Below 40°F: SAE 30, 10W30, 15W50, or 20W50
   2. Above 40°F: SAE50, 15W50, 20W50, or 20W60

12-14. OXYGEN SYSTEM SERVICING

General safety guidelines and servicing procedures must be followed and are as indicated in Chapter 35. Use only MIL-O-27210 aviator's breathing oxygen for system recharging.

NOTE

This fixed oxygen system has a maximum bottle pressure of 2,000 psi.

WARNING

Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM in Chapter 35 before attempting any servicing or maintenance of the oxygen system.

a. **Basic or Avidyne Option**

1. Open all aircraft doors to provide air circulation.
2. Set the Left Bus switch to ON.
3. Set the Oxygen Switch to DISP.
4. Locate the Oxygen Fill Port in the baggage bulkhead.
5. Remove the Oxygen Fill Port protective cap and connect the service line.
6. Slowly fill the oxygen supply. By filling slowly, the temperature rise due to the compression of gas into the oxygen supply cylinders will be kept to a minimum.
7. Fill the system until a pressure of 1950 ± 50 psig is achieved. This value is valid at a steady state condition after the system has cooled to 70 º F from the recharging heat buildup. System pressure may be increased 3.5 psig for each degree of increase in temperature above 70 º F and decreased 3.5 psig for each degree decrease in temperature below 70 º F.
8. Shut off the oxygen supply to the aircraft.
9. Set the Oxygen Switch to ON.
10. Ensure that the Outlet Pressure Display green indication is illuminated.
11. Set the Oxygen Switch to OFF.
12. Set the Left Bus switch to OFF.
13. Disconnect the service line from the Fill Port and install the protective cap.

b. **Garmin G1000 Option**

1. Open all aircraft doors to provide air circulation.
2. Set the Right Bus and Avionics Bus switches to ON.
3. Turn on the MFD and access the Engine Indication System (EIS) page.
4. Locate the Oxygen Fill Port in the baggage bulkhead.
5. Remove the Oxygen Fill Port protective cap and connect the service line.
6. Slowly fill the oxygen supply. By filling slowly, the temperature rise due to the compression of gas into the oxygen supply cylinders will be kept to a minimum.
7. Fill the system until a pressure of 1950 ± 50 psig is achieved. This value is valid at a steady state condition after the system has cooled to 70 º F from the recharging heat buildup. System pressure may be increased 3.5 psig for each degree of increase in temperature above 70 º F and decreased 3.5 psig for each degree decrease in temperature below 70 º F.
8. Shut off the oxygen supply to the aircraft.
9. Set the Oxygen Switch to ON.
10. Ensure that the Outlet Pressure and Oxygen Quantity gauges confirm oxygen supply.
11. Set the Oxygen Switch to OFF.
12. Turn off the MFD.

13. Set the Right Bus and Avionics Bus switches to OFF.
14. Disconnect the service line from the Fill Port and install the protective cap.

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12-15. OIL FILTER SERVICING

a. A full flow, spin on-type, 20-micron oil filter is used. As mentioned previously, the filter must be changed during the initial break-in period at 25 and 50 hours time in service. Thereafter, the filter must be replaced at every 100 hours time in service interval. If the airplane is predominately operated in conditions of high dust or other atmospheric debris, it is advisable to change the filter at 50 hour intervals. See Chapter 79 for instructions for removal and replacement of the oil filter. Use a TCM filter or approved equivalent.

12-16. INDUCTION FILTER SERVICING

a. The airplane has a disposable air filter, Brackett Aircraft Company, Inc. Part No. BA-9005. The rectangular filter element is a wettant impregnated foam. The filter must be serviced annually, every 100 hours, or more frequently if operating in areas of high atmospheric debris. To service the filter, use the following procedure.

1. Remove the air filter from the engine per instructions in Chapter 71.
2. Dispose of old filter element.
3. Install a new filter element per instruction in Chapter 71.

12-17. TIRES AND WHEELS SERVICING

a. Ensure tires are serviced to their proper pressures. See paragraph 06-8 pressure specifications. The following items must also be inspected.

1. Check tires for flat spots, splitting, dry rot, cuts, and any other evidence of damage.
2. Check valve stem slip marks for evidence of slippage.
3. Check wheels for damage.
4. Check wheel bearings for condition and lubrication.

b. **Tire Considerations** – The airplane is normally delivered with Goodyear tires. These tires have a profile that provides about $^{3}/_{8}$ in. clearance between the tire and wheel pants. Other brands of tires with similar specifications and TSO's may have slightly larger profiles. Tires with larger profiles are not recommended since damage to the tire or wheel pant is possible, particularly during landing. If other brands of tires are used, the profile of the tire must be precisely measured and compared with the Goodyear tire.

CAUTION

The profile of replacement tires that are not a recommended brand should be measured precisely to ensure they are the same height and width. The use of tires that have slightly larger profiles can cause damage to the tire and to the wheel pant, particularly during landing operations.

12-18. SERVICING THE NOSE STRUT

The nose strut is normally serviced at required maintenance inspections. If less than three to four inches of chrome are visible on the nose strut or a low oil level is indicated, then unscheduled servicing is necessary. A low oil level in the nose wheel strut may be indicated by one or both of two symptoms. First, there will a general deterioration in shimmy damping of the nose strut. Second, the nose will have a tendency to bounce noticeably while taxiing. If either or both of these conditions are experienced, a low oil level in the nose strut is the most likely cause.

a. **Cessna Nose Strut** (see Figure 12 - 1 and Figure 12 - 2)

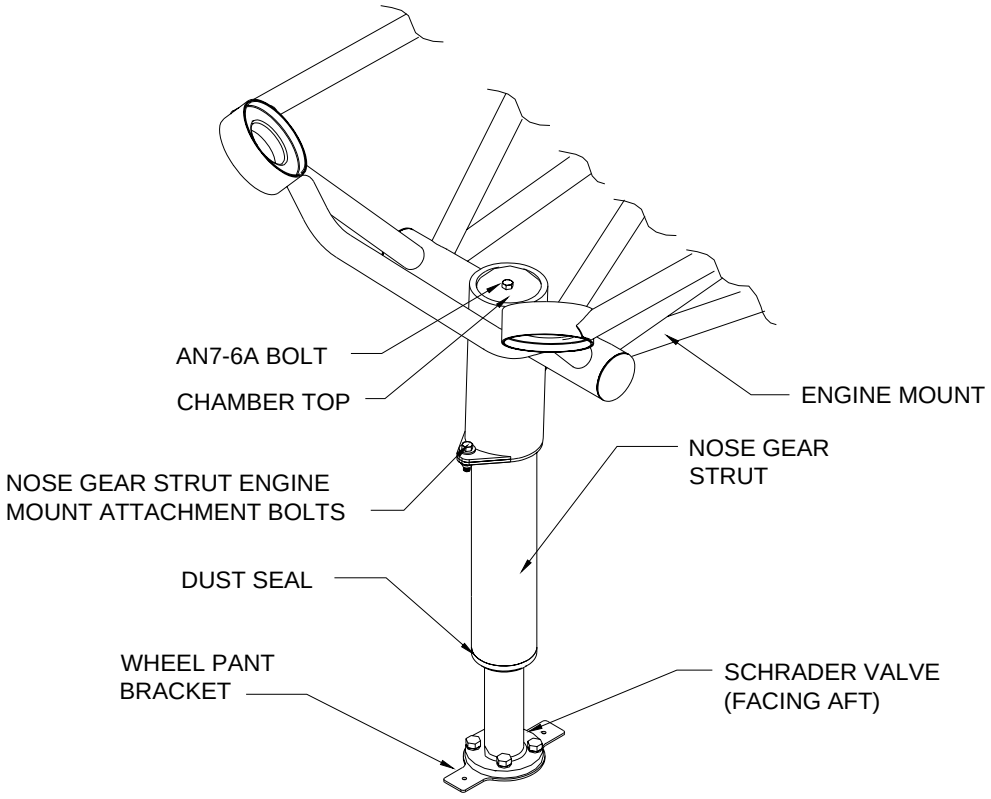

Figure 12 - 1 Cessna Nose Strut Assembly

1. Remove the nose strut from the nose gear per instructions in Chapter 32.
2. Remove the valve cap from the strut schrader valve. Using a small flat bladed screwdriver, gently press in on the schrader valve core stem to slowly bleed off nitrogen gas pressure in the strut.

CAUTION

Always wear safety goggles when releasing gas pressure.

WARNING

Removal of the AN7-6A bolt without de-pressurizing the strut could cause serious injury.

3. Remove the AN7-6A bolt at the top of the strut.
4. Pull the strut shaft to its fully extended position.
5. Inflate the strut with nitrogen gas to a pressure of 50 psi.
6. With the strut standing vertically, add MIL-H-5606F hydraulic oil through the AN7-6A bolt hole until the chamber is full.
7. Create an oil bleed device composed of a flexible hose inserted into a reservoir filled with MIL-H-5606F hydraulic oil. Ensure the hose is filled with oil and attach the free end to the AN7-6A bolt hole. Cycle the strut up, down, left, and right through its full range of motion until all air is removed.
8. With the strut fully extended, remove the valve core from the schrader valve and release the nitrogen gas pressure.
9. Pinch shut the flexible hose on the oil bleed device and push the strut shaft up 3/4" from its fully extended position to correctly position the internal piston.
10. Un-pinch the flexible hose.
11. Ensure the valve core is free from contaminates and reinstall it into the schrader valve. Tighten snug.
12. Slowly pull the strut shaft to its fully extended position.
13. Remove the bleed device and add additional hydraulic oil until the chamber is filled to the top.
14. Replace the AN7-6A bolt and tighten snug.
15. Inflate the strut with nitrogen gas until pressure reaches 250 psi ± 25 psi. Best results are obtained using the following two techniques.
    a) Apply enough physical force on the nitrogen hose chuck to ensure a tight seal on the filler valve. The volume of area being filled is small and requires a lot of physical pressure to hold the chuck on the valve.
    b) When the strut is filled, remove the air hose chuck from the filler valve as quickly as possible. A persistent low nose gear strut is frequently the result of losing 50 pounds or so of pressure when the air chuck is removed. Over-pressurization will prevent full castoring movements of the nose wheel.
16. With the strut firmly placed on the ground, attempt to compress the strut by hand. If the strut can be compressed, repeat the above instructions until all air is removed and the strut cannot be compressed.
17. Torque the AN7-6A bolt at the top of the strut 80 to 100 in.-lbs.
18. Install the schrader valve cap and torque 10 to 15 in.-lbs.
19. Install the nose strut into the nose gear per instructions in Chapter 32.

NOTE

Persistent and frequent loss of oil is an indication of more serious problems. The oil seals may need replacement or there may be issues with the integrity of the cylinder. See Chapter 32 for information.

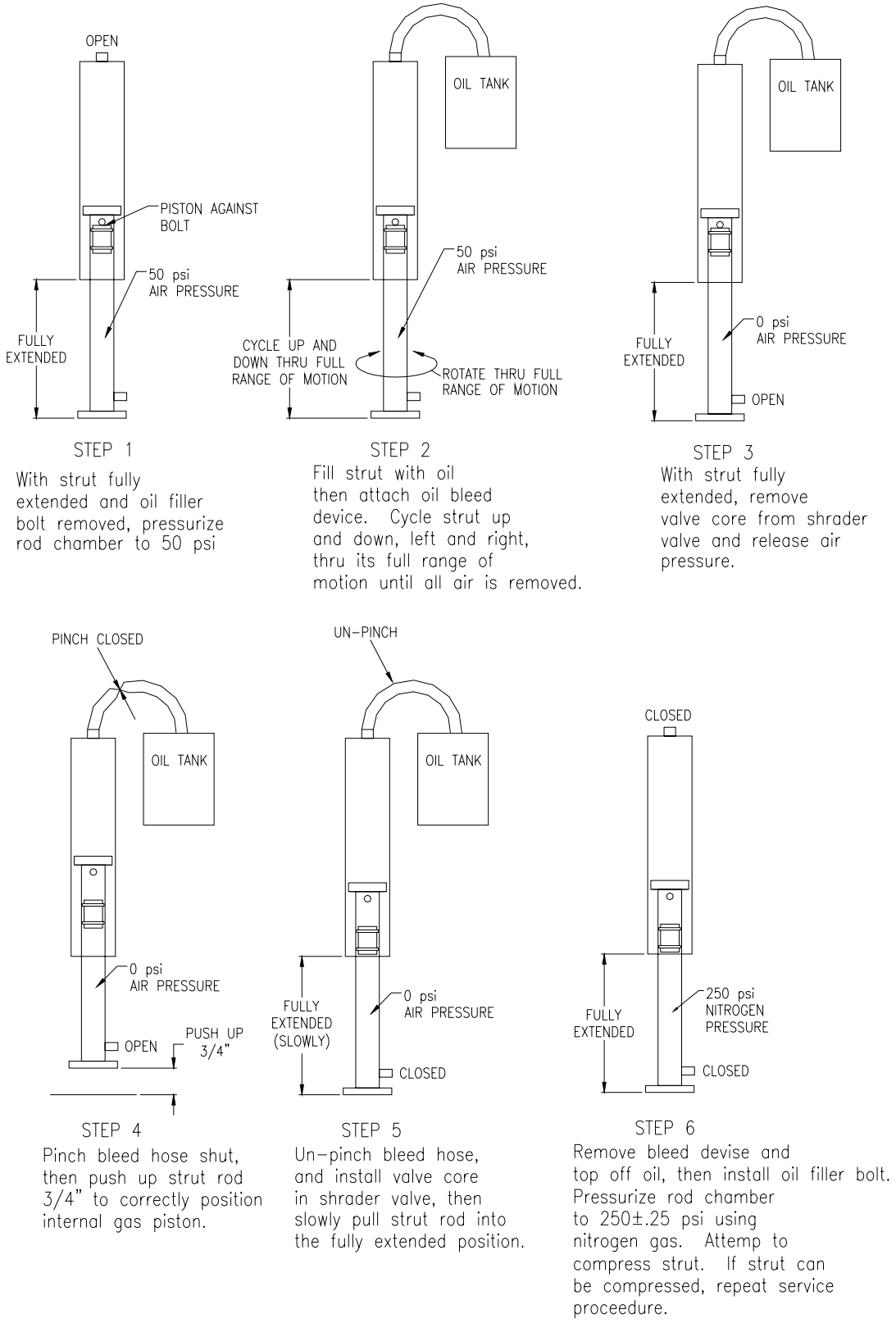

Figure 12 - 2 Cessna Nose Strut Service

b. **ESCO Nose Strut** (see Figure 12 - 3)

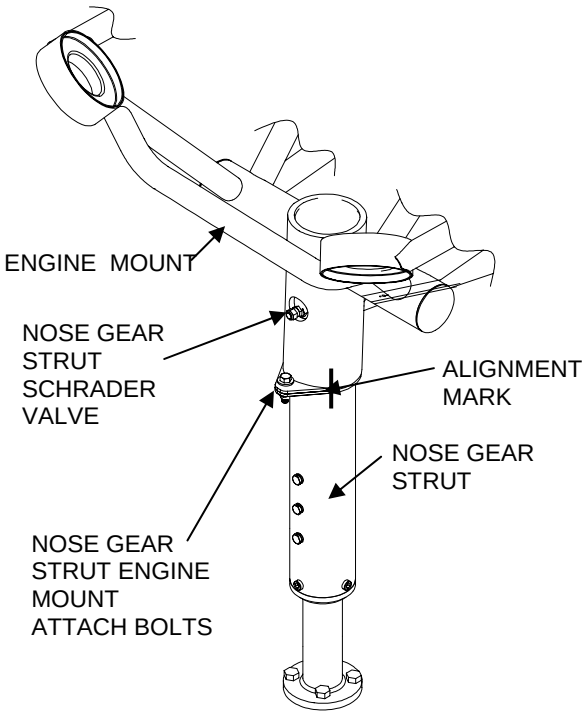

Figure 12 - 3 ESCO Nose Strut Assembly

1. Remove the lower cowling, using procedures described in Chapter 71, to expose the filler valve (Schrader valve).
2. Remove all the weight from the nose strut by placing ballast on the tail, and let the strut settle for one hour.
3. Release the nitrogen gas from the strut by pressing on the stem of the filler valve until most of the nitrogen is released. Hold a rag over the valve stem and use the hand to release the pressure to prevent a small amount of oil from being scavenged from the main cylinder as the nitrogen is released.
4. Partially remove the valve core to evacuate the remaining gas in the cylinder.
5. Remove the schrader valve.
6. Ensure that the strut is fully extended.
7. Hydraulic oil that meets MIL-H-5606F can now be added slowly with a pump action oil can. Add oil until the level in the cylinder reaches the bottom of hole from which the schrader valve was removed.
8. Rotate the strut left and right throughout its full range of rotation (60° each direction) to completely fill internal components of the strut with oil. When internal components are full of oil, resistance should be felt while rotating the strut. Keep adding oil while rotating until the oil drains back out of the filler valve hole and no more can be added.
9. Refit the schrader valve and torque 150 to 200 in.-lbs.
10. Ensure that the strut is fully extended. Insert the air hose chuck onto the valve stem. Fill until pressure reaches 250 psi. Best results are obtained using the following two techniques.

a) Apply enough physical force on the nitrogen hose chuck to ensure a tight seal on the filler valve. The volume of area being filled is small and requires a lot of physical pressure to hold the chuck on the valve.
b) When the strut is filled to 250 psi, remove the air hose chuck from the filler valve as quickly as possible. A persistent low nose gear strut is frequently the result of losing 50 pounds or so of pressure when the air chuck is removed.

11. Repeat steps 12-18.b.3 through 12-18.b.6 and finish filling the strut with oil until oil drains back out the filler valve hole and no more can be added.
12. Refit the schrader valve and torque 150 to 200 in.-lbs. Service the strut with nitrogen per the instructions in paragraph 12-18.b.10, replace the schrader valve cap and tighten snug. Over-pressurization will create a nose high attitude while taxiing and will prevent full castoring movements of the nose wheel.
13. Lower the nose by progressively removing ballast from the tail. Pull down on the propeller to exert downward pressure on the nose gear. Verify that three to four inches of chrome are visible on the nose strut.
14. Replace the lower cowling per instructions in Chapter 71.

NOTE

Persistent and frequent loss of oil is an indication of more serious problems. The oil seals may need replacement or there may be issues with the integrity of the cylinder. See Chapter 32 for information.

12-19. BRAKE FLUIDS

a. Under most conditions, the airplane will not require servicing of the hydraulic system. There is a small hydraulic reservoir mounted on the upper portion of the firewall (engine side). The reservoir contains an Allen screw, however, this area is not normally used for servicing the airplane. If brake sponginess is experienced, this is an indication that there is air in the system. More important, it may be an indication of a hydraulic leak in the brake system, and the areas proximate to the hydraulic lines, master cylinders, and brake assembly must be inspected for evidence of hydraulic oil. If no leaking is detected, then the brakes should be bled. Chapter 32 contains instructions for bleeding the brakes. If hydraulic fluid is added, use MIL-H-5606F.

12-20. BATTERY SERVICING

a. For 12V aircraft, the installed batteries are Concorde RG-1215; for 24V aircraft they are Teledyne Gill G230S, Teledyne Gill G248S, Concorde RG24-10, or Concorde RG24-16. The batteries are of valve regulated lead-acid (VRLA) design. Since the batteries have no electrolyte, there is no required servicing of the unit. The posts and cables are serviced at 100 hour inspections if applicable, otherwise, they are maintained during the annual inspection. It is recommended that the batteries be replaced every four years or sooner if poor overall performance is experienced.

12-21. CLEANING AIRCRAFT

a. **Exterior** – The exterior painted surfaces are cleaned by washing with mild soap and water and drying with a soft towel or chamois. The seal coats that are applied to the painted surface, in most instances, will provide adequate protection from moisture and the sun. Some additional protection is provided by waxing the painted surface and facilitates washing the airplane since bugs and dirt will not adhere as tightly to a waxed surface. A wax with a high concentration of carnauba is recommended. There are several commercial boat waxes available that are ideal for this use. Be sure to read the label with an eye for the percentage of carnauba in the compound.

NOTE

The exterior of the airplane should not be waxed within the first 90 days of the date of delivery. This period will ensure the painted surface has adequate time to cure. The exterior paint color on the upper fuselage area and the top of the wings has a good heat reflection index. This good index is required to ensure the continued bonding and integrity of the composite material. Only approved Cessna paint colors are permitted in these areas. Care must be taken to not lay dark, heat absorbing material on the top area of the wings and fuselage.

b. **Windshield and Windows -** The proper care of the windshield and windows (sometimes referred to as transparencies) is one of the more important exterior care items on the airplane. Never do anything that will scratch the surface of the acrylic plastic. The following points for cleaning and caring for the transparencies should be observed. Refer to the Transparency Repair section of Chapter 51 for approved cleaning materials and expanded procedures.

1. First, when cleaning the windows, it is recommended that rings and watches be removed as they can cause deep scratches. Long-sleeved shirts should be turned up a few rolls to hide exposed buttons.
2. When removing bugs and dirt, avoid touching the surface. If possible, remove most of the dirt by flushing the windows and windshield with water and a mild dish soap mixture. Allow the accumulation of dirt and/or bugs to soak for a few minutes. If rubbing is required, a bare hand is best. When all the debris on the surface of the window is loosened, apply a second water flush, and then dry with a 100% cotton cloth.
3. Use a good quality, non-abrasive cleaner/polish specifically intended for acrylic windows, and apply per the manufacturer's instructions. Use up and down or side to side movements when polishing. Never use a circular movement as this can cause glare rings.
4. The best polishing cloth is the softest cotton available. One hundred percent cotton flannel is ideal and available in yard goods stores. **Never** use any type of paper product or synthetic material. In particular, never use shop rags or shop towels. Be sure the polishing cloth is clean and dry. Reserve polishing cloths should be stored in a plastic bag to limit dirt accumulation.

5. Small scratches, the type that can be seen but cannot be felt with a fingernail, should be filled with a polishing compound that has scratch filling properties. The cleaner/polisher mentioned in paragraph 3 frequently has scratch filling properties and is satisfactory for regular use. Some scratches cannot be filled with a scratch-filling product. While the scratches cannot be felt, they are still visible, particularly when flying into the sun. In this instance, a mildly abrasive scratch removal cream can be used per manufacturer's recommendations. Scratches of greater magnitude require the use of high abrasives and removal of some of the window's surface around the greatest depth of the scratch. This procedure requires considerable expertise and frequently makes areas where the scratch was removed more objectionable than the original scratch.

CAUTION

Do not use anything containing ammonia, aromatic solvents like methyl ethyl ketone, acetone, lacquer thinner, paint stripper, gasoline, benzene, alcohol, anti-ice fluid, hydraulic fluid, fire extinguisher solutions, or window cleaner on the acrylic window surfaces. The use of these substances may cause the surface to craze.

NOTE

To remove difficult substance such as tape residue, oil, and grease, the safest solvents are 100% mineral spirits or kerosene. Some alcohols are safe, such as isopropyl alcohol.

c. **Interior Cleaning and Care** – The useful life of the airplane's interior can be extended through proper care and cleaning. One of the major elements in the aging process is the interior's exposure to sunlight. If possible, the airplane should be hangared. Routine vacuuming is another item that helps extend the life of the airplane's interior, particularly the carpets. The leather seats, seatbacks, knee bolsters, glare shield and the like, should be routinely wiped with moist, soft cotton cloth. A general rule for spills is to blot the affected area with firm pressure for a few seconds. Never rub an area to remove a spill.

1. Much of the airplane's interior is covered with high-quality leather. The leather is impregnated with polyvinyl chloride (PVC), and as treated, is impervious to most substances. The application of leather preservatives will provide little protection, and in some cases can cause problems. For difficult stains, such as an ink stain, remove with a mild soap and water. Do not use any type of chemical since these products tend to remove the impregnated PVC protection, and once removed it can never be reapplied.
2. The carpet can be cleaned with a mild foam product, but care should be used not to over saturate the carpet. Follow the manufacturer's instructions regarding use of the foam cleaner. Small spots can be cleaned with a commercial spot remover; however, this must be done with care. Again, follow the recommended procedure of the manufacturer, and try a test application in an area of limited exposure.

CHAPTER 20

STANDARD PRACTICES AIRFRAME

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20-1. GENERAL

This chapter describes the standard maintenance practices for the Cessna 350 (LC42-550FG) aircraft. The items covered are typical and apply to many areas or systems of the aircraft. Unique maintenance procedures are covered in the appropriate chapter in the maintenance manual.

NOTE

Most of this chapter is derived from FAA Advisory Circular (AC) 43.13-1B, which should be used (latest revision) as the resource for this information.

20-2. Material and Tool Cautions – General

a. **Mercury**

1. Mercury can corrode and embrittle metallic structures by the amalgamation process. Complete destruction of the load carrying capability of the structure will result.
2. No thermometers or equipment containing mercury can be used on the aircraft.

b. **Epoxy Resins**

1. Epoxy resins, particularly the hardeners, are chemically active and can cause chemical burns, allergic reactions, and strong sensitization.
2. Particular care should be taken to avoid contact with, or breathing the vapors from epoxy resins.

c. **Solvents**

1. General use solvents may include the following:
   a) Gasoline
   b) Oil
   c) Alcohol
2. Many of these solvents are flammable.
3. These solvents can cause injury if contact with the skin or eyes is made.
4. Many of these solvents are poisonous.

20-3. TORQUE DATA – MAINTENANCE PRACTICES

WARNING

The torque values outlined in this chapter and other chapters must be used during installation and repair of components. Improper torque values may result in over-stressing of components. Bolted joints in composite structures are sensitive to overload if improper over-torque is used.

a. **Torque Values –** Torque tables are listed in this section. If a component requires specific torque values, those values will be given in the applicable chapter and section of the maintenance manual. See the following figures for specific types of torque values: Figure 20 - 2 Values for Tapped Holes & Clearance Holes; Figure 20 - 3 Values for Nutplates; Figure 20 - 4 Values for Hose and Tube Fittings; Figure 20 - 5 Values for Rivnuts; Figure 20 - 6 Recommended Torque and Clamp Load Specifications (Applicable to Fasteners with a Minimum Tensile Strength Greater Than 74 ksi); Figure 20 - 7 Recommended Jam Nut Torque and Clamp Load

Specifications (Applicable to Fasteners With a Minimum Tensile Strength Greater Than 74 ksi).

b. **Free Running Torque** – The torque values listed do not include free running torque. The free running torque is the torque value to rotate the nut on the threaded shaft before contact, pull-up, and tightening. Self-locking nuts are used in applications where they will not be removed often. Repeated removal and installation will cause the self-locking nut to lose its locking feature. They should be replaced when they are no longer capable of maintaining the minimum prevailing torque.

c. **General Torquing Notes**

1. The values given do not apply to threaded adjustment devices such as rod ends or turnbuckles.
2. Appropriate torque wrenches should be used. Adjustments should be made for the readings if adapters are required. Refer to specific instructions supplied with the torque wrench.

d. **Torque Adapters –** Using one or more extensions with a torque wrench can put large side loads on torqued fasteners or anything they are attached to (i.e., hoses or tubes); therefore, the operator must support the fastener assembly appropriately when needed.

When using an extended length adapter on a torque wrench (i.e. crow's foot), apply the following equation:

$$T_E = (T \times L) / (L + E)$$ where,

T = Specified torque value to achieve

L = Wrench length from grip center to turn center (see Figure 20 - 1)

E = Length, in axial direction of wrench, from wrench turn center to extension turn center (see Figure 20 - 1)

T_E = Torque wrench setting when using adapter

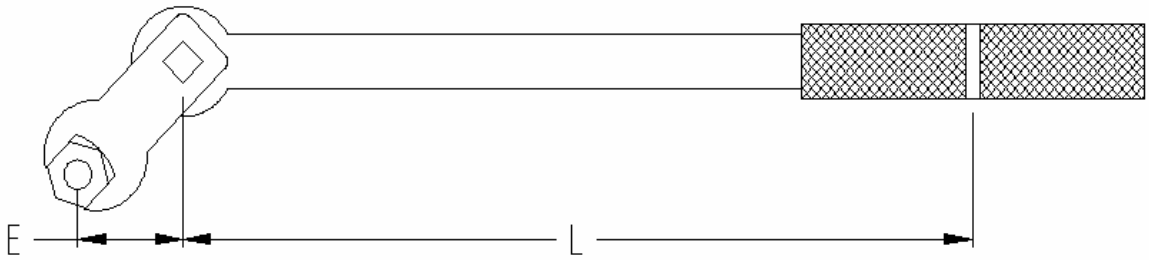
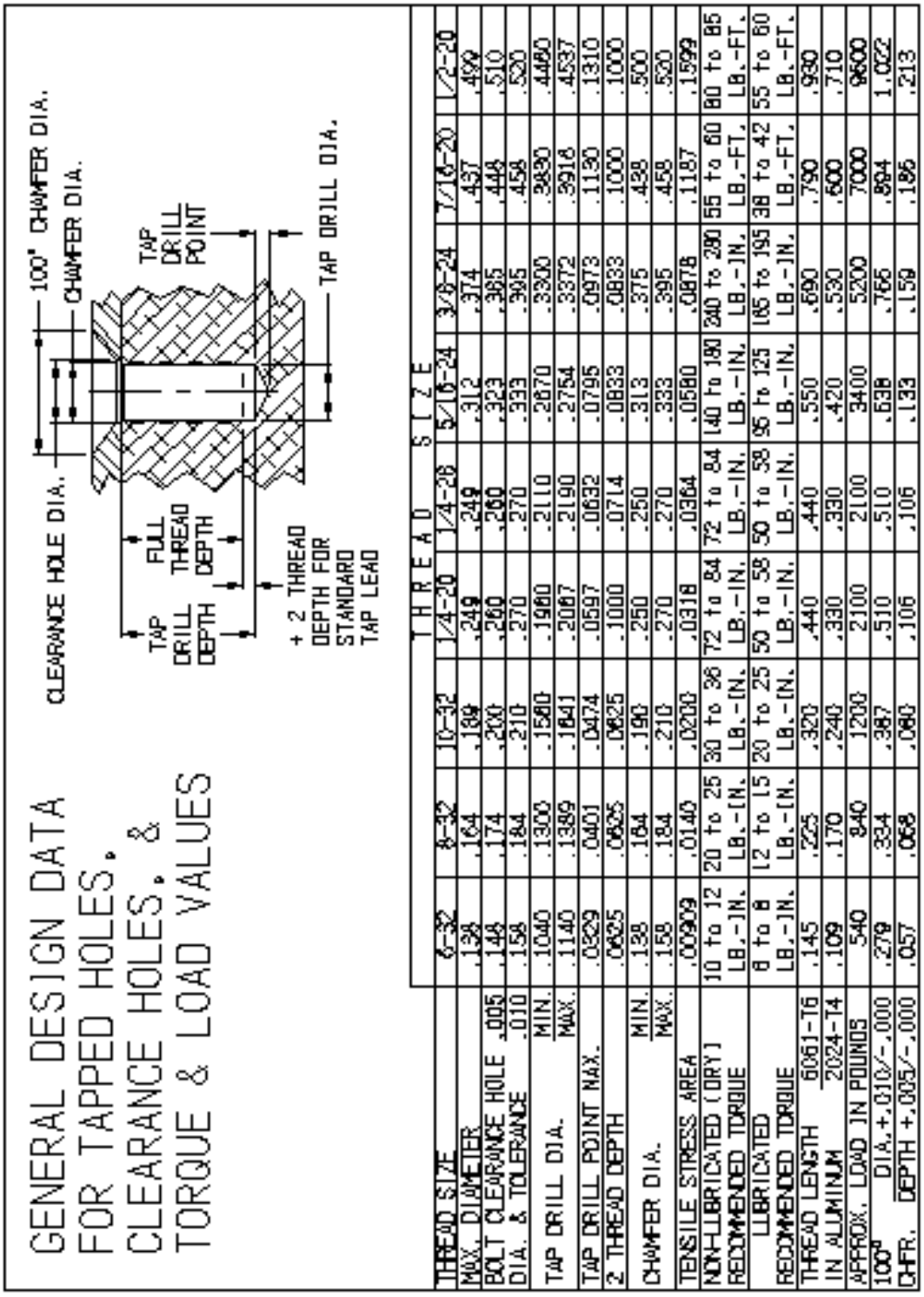
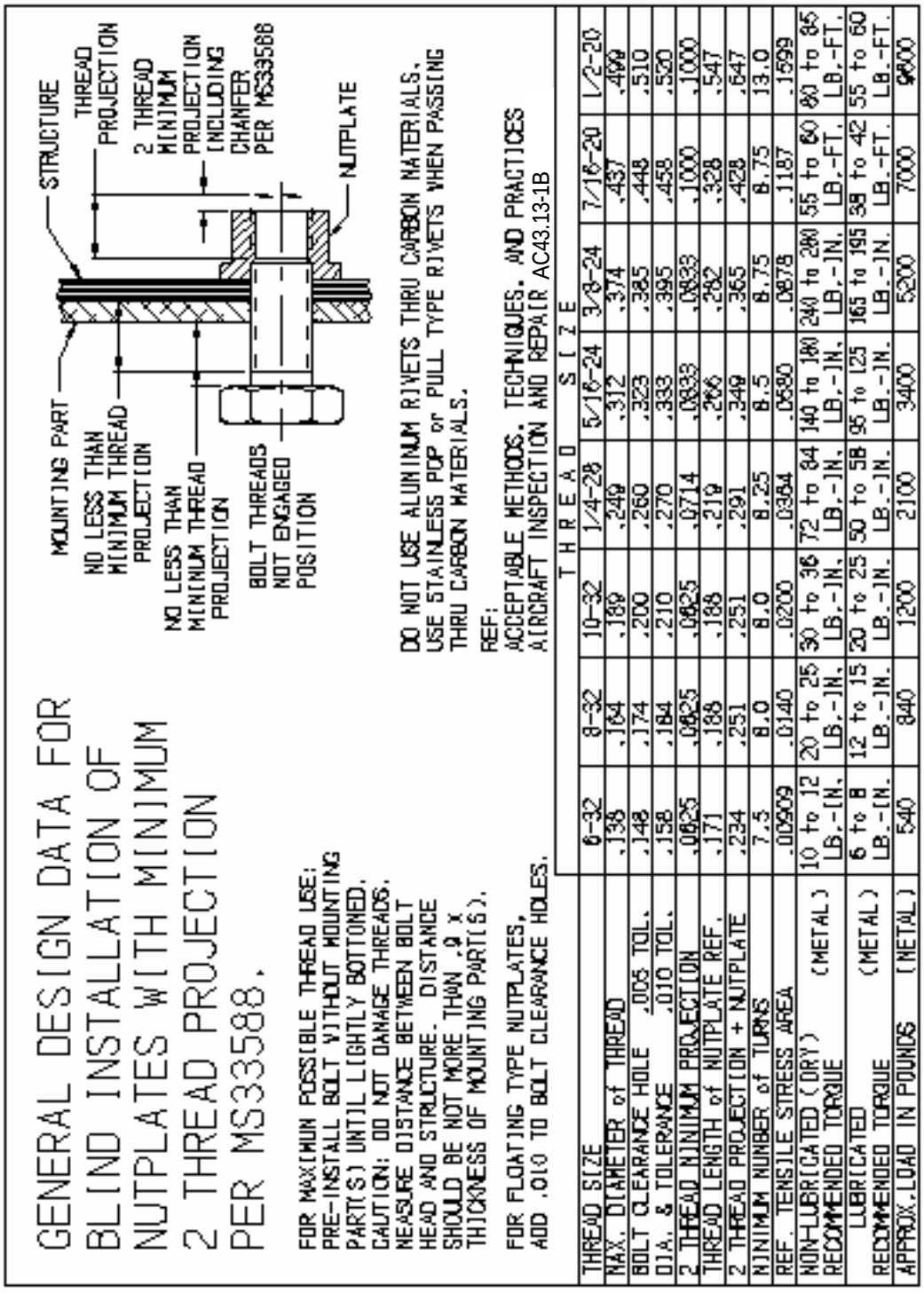


Figure 20 - 1 Length Variable for Torque Adapter Equation



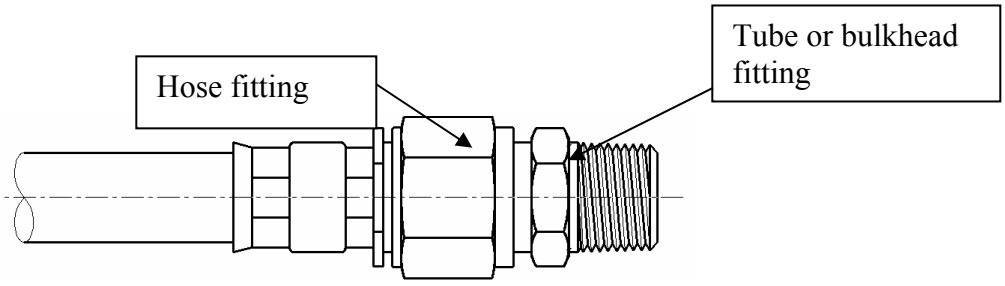
	THREAD SIZE											
THREAD SIZE		6-32	8-32	10-32	1/4-20	1/4-28	5/16-24	3/8-24	7/16-20	1/2-20		
MAX. DIAMETER		.138	.164	.189	.249	.249	.312	.374	.437	.499		
BOLT CLEARANCE HOLE DIA. & TOLERANCE	.005	.144	.174	.200	.260	.260	.323	.385	.448	.510		
	.010	.150	.184	.210	.270	.270	.333	.395	.458	.520		
TAP DRILL DIA.	MIN.	.1040	.1300	.1560	.1960	.2110	.2670	.3320	.3839	.4460		
	MAX.	.1140	.1389	.1641	.2067	.2190	.2754	.3372	.3918	.4537		
TAP DRILL POINT MAX.		.0329	.0401	.0474	.0597	.0632	.0795	.0973	.1130	.1310		
2 THREAD DEPTH		.0625	.0625	.0625	.1000	.0714	.0833	.0833	.1000	.1000		
CHAMFER DIA.	MIN.	.138	.164	.190	.250	.250	.313	.375	.438	.500		
	MAX.	.158	.184	.210	.270	.270	.333	.395	.458	.520		
TENSILE STRESS AREA		.00909	.0140	.0200	.0318	.0364	.0580	.0878	.1187	.1599		
NON-LUBRICATED (DRY) RECOMMENDED TORQUE		10 to 12 LB.-IN.	20 to 25 LB.-IN.	30 to 36 LB.-IN.	72 to 84 LB.-IN.	72 to 84 LB.-IN.	140 to 180 LB.-IN.	240 to 290 LB.-IN.	55 to 60 LB.-FT.	80 to 85 LB.-FT.		
LUBRICATED RECOMMENDED TORQUE		8 to 8 LB.-IN.	12 to 15 LB.-IN.	20 to 25 LB.-IN.	50 to 58 LB.-IN.	50 to 58 LB.-IN.	95 to 125 LB.-IN.	185 to 195 LB.-IN.	38 to 42 LB.-FT.	55 to 60 LB.-FT.		
THREAD LENGTH IN ALUMINUM	6061-T6	.145	.225	.320	.440	.440	.550	.690	.790	.930		
	2024-T4	.109	.170	.240	.330	.330	.420	.530	.600	.710		
APPROX. LOAD IN POUNDS		540	840	1200	2100	2100	3400	5200	7000	9600		
100° CHFR.	DIA.+.010/-.000	.279	.334	.387	.510	.510	.638	.765	.894	1.022		
	DEPTH +.005/-.000	.057	.068	.080	.106	.106	.133	.159	.186	.213		

Figure 20 - 2 Values for Tapped Holes & Clearance Holes



THREAD SIZE	6-32	8-32	10-32	1/4-28	5/16-24	3/8-24	7/16-20	1/2-20
MAX. DIAMETER of THREAD	.138	.164	.189	.249	.312	.374	.437	.499
BOLT CLEARANCE HOLE .005 TOL.	.148	.174	.200	.260	.323	.385	.448	.510
DIA. & TOLERANCE .010 TOL.	.158	.184	.210	.270	.333	.395	.458	.520
2 THREAD MINIMUM PROJECTION	.0625	.0625	.0625	.0714	.0833	.0833	.1000	.1000
THREAD LENGTH of NUTPLATE REF.	.171	.188	.188	.219	.266	.282	.328	.547
2 THREAD PROJECTION + NUTPLATE	.234	.251	.251	.291	.349	.365	.428	.647
MINIMUM NUMBER of TURNS	7.5	8.0	8.0	8.25	8.5	8.75	8.75	13.0
REF. TENSILE STRESS AREA	.00909	.0140	.0200	.0364	.0580	.0878	.1187	.1599
NON-LUBRICATED (DRY) (METAL) RECOMMENDED TORQUE	10 to 12 LB.-IN.	20 to 25 LB.-IN.	30 to 36 LB.-IN.	72 to 84 LB.-IN.	140 to 180 LB.-IN.	240 to 280 LB.-IN.	55 to 60 LB.-FT.	80 to 85 LB.-FT.
LUBRICATED (METAL) RECOMMENDED TORQUE	6 to 8 LB.-IN.	12 to 15 LB.-IN.	20 to 25 LB.-IN.	50 to 58 LB.-IN.	95 to 125 LB.-IN.	165 to 195 LB.-IN.	38 to 42 LB.-FT.	55 to 60 LB.-FT.
APPROX. LOAD IN POUNDS (METAL)	540	840	1200	2100	3400	5200	7000	9000

Figure 20 - 3 Values for Nutplates



Size/Thread		Material and Torque Specification		
		**Hose and/or Fitting Combination	*Inch Pounds	*Foot Pounds
- 2	5/16-24	Steel to Steel	75 - 120	6.3 - 10
		Brass/Aluminum/Steel	50 - 80	4.2 – 6.7
- 3	3/8-24	Steel to Steel	95 - 140	7.9 – 11.7
		Brass/Aluminum/Steel	70 - 105	5.8 – 8.8
- 4	7/16-20	Steel to Steel	135 - 190	11.3 – 15.8
		Brass/Aluminum/Steel	100 - 140	8.3 – 11.7
- 5	1/2-20	Steel to Steel	170 - 240	14.2 - 20
		Brass/Aluminum/Steel	130 - 180	10.8 - 15
- 6	9/16-18	Steel to Steel	215 - 280	17.9 – 23.3
		Brass/Aluminum/Steel	150 - 195	12.5 – 16.3
- 8	3/4-16	Steel to Steel	470 - 550	39.2 – 45.8
		Brass/Aluminum/Steel	270 - 350	22.5 – 29.2
-10	7/8-14	Steel to Steel	620 - 745	51.7 – 62.1
		Brass/Aluminum/Steel	360 - 430	30 – 35.8
-12	1 1/16-12	Steel to Steel	855 - 1055	71.3 – 87.9
		Brass/Aluminum/Steel	460 - 550	38.3 – 45.8
*Dry torque values shown, except as noted in General Notes below.				
** Brass/Aluminum/Steel means any combination of brass to aluminum, brass to steel, or aluminum to steel.				

General Notes:

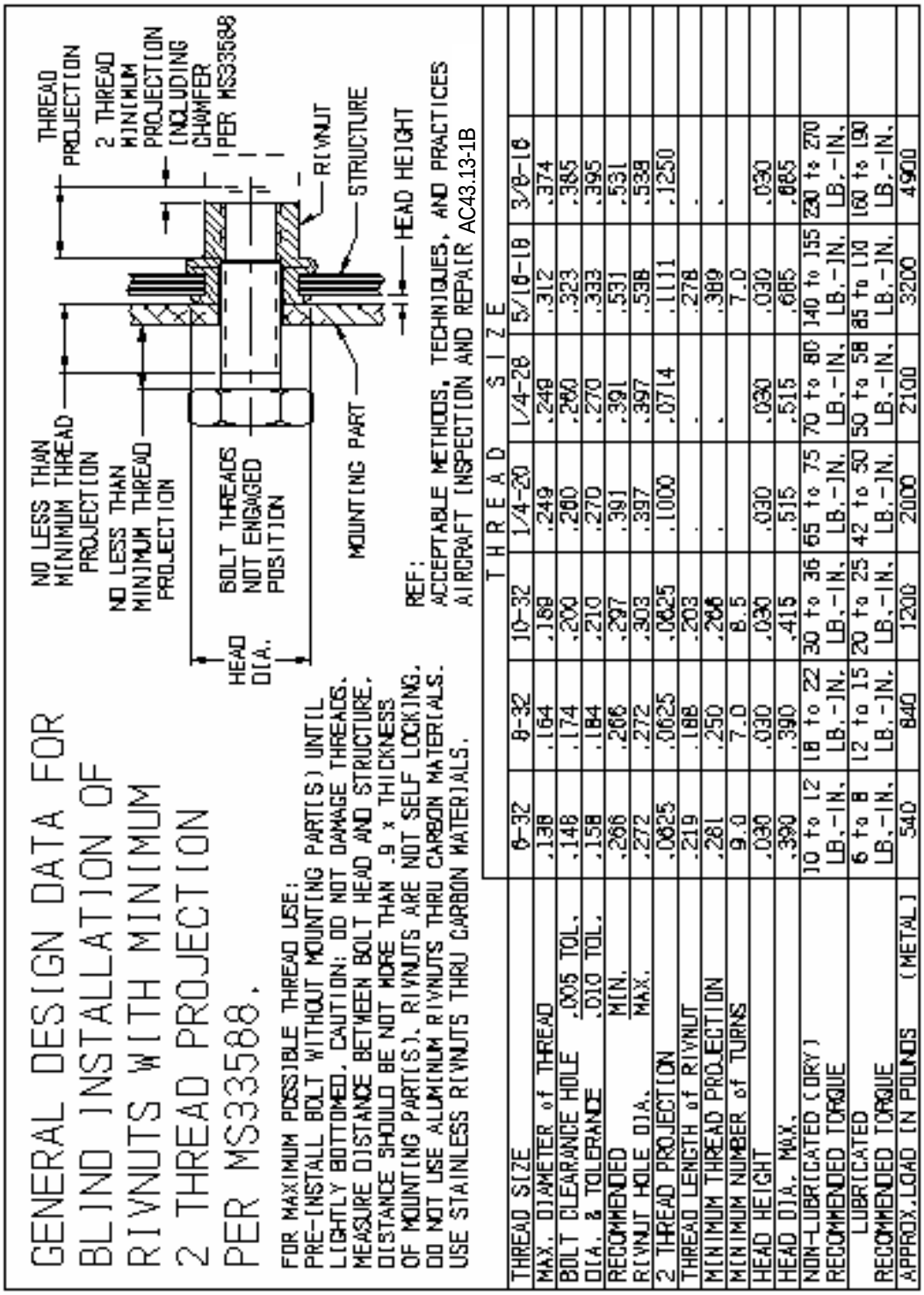
Pre-Install hose onto fitting finger tight. If parts do not assemble finger tight, check for foreign material on threads and/or damaged threads. Do not force assemblies together.

Maintain cleanliness of hose and fittings. Protect all fitting threads from damage and dirt contamination.

When attaching stainless steel to stainless steel or stainless steel to aluminum, use MIL-T-5544 Anti-seize/lubricant on fitting threads. To prevent contamination, do not apply Anti-seize to tapered seat. Use lower torque value when Anti-seize is used.

For Swagelok tube fittings use finger tight + 1-1/4 turns.

Figure 20 - 4 Values for Hose and Tube Fittings



THREAD SIZE	6-32	8-32	10-32	1/4-20	1/4-28	5/16-18	3/8-16
MAX. DIAMETER of THREAD	.138	.164	.189	.249	.249	.312	.374
BOLT CLEARANCE HOLE .005 TOL.	.148	.174	.200	.260	.260	.323	.385
DIA. & TOLERANCE .010 TOL.	.158	.184	.210	.270	.270	.333	.395
RECOMMENDED MIN.	.266	.266	.297	.391	.391	.531	.531
RIVNUT HOLE DIA. MAX.	.272	.272	.303	.397	.397	.538	.538
2 THREAD PROJECTION	.0625	.0625	.0625	.1000	.0714	.1111	.1250
THREAD LENGTH of RIVNUT	.219	.188	.203	-	-	.278	-
MINIMUM THREAD PROJECTION	.281	.250	.266	-	-	.389	-
MINIMUM NUMBER of TURNS	9.0	7.0	8.5			7.0	
HEAD HEIGHT	.030	.030	.030	.030	.030	.030	.030
HEAD DIA. MAX.	.390	.390	.415	.515	.515	.685	.685
NON-LUBRICATED (DRY)	10 to 12	18 to 22	30 to 36	65 to 75	70 to 80	140 to 155	230 to 270
RECOMMENDED TORQUE	LB.-IN.	LB.-IN.	LB.-IN.	LB.-IN.	LB.-IN.	LB.-IN.	LB.-IN.
LUBRICATED	6 to 8	12 to 15	20 to 25	42 to 50	50 to 58	85 to 110	160 to 190
RECOMMENDED TORQUE	LB.-IN.	LB.-IN.	LB.-IN.	LB.-IN.	LB.-IN.	LB.-IN.	LB.-IN.
APPROX. LOAD (N POUNDS (METAL)	540	840	1200	2000	2100	3200	4900

Figure 20 - 5 Values for Rivnuts

Fastener Size	Nominal Diameter	Clamp Load	Torque Dry*	Torque Lubed*
	[in]	[lbs]	[in-lbs,**ft-lbs**]	[in-lbs,**ft-lbs**]
2-56	.0860	150	2.5	2.0
4-40	.1120	250	5.5	4.0
6-32	.1380	370	10	8.0
8-32	.1640	580	19	14
10-32	.1900	830	31	24
1/4-20	.2500	1310	66	49
1/4-28	.2500	1500	75	56
5/16-18	.3125	2160	140	100
5/16-24	.3125	2390	150	110
3/8-16	.3750	3200	**20**	**15**
3/8-24	.3750	3620	**23**	**17**
7/16-20	.4375	4900	**36**	**27**
1/2-20	.5000	6600	**55**	**41**
*Maximum Error of ±10%				
**Applicable for standard or self-locking types of nuts				

Figure 20 - 6 Recommended Torque and Clamp Load Specifications (Applicable to Fasteners with a Minimum Tensile Strength Greater Than 74 ksi)

Fastener Size	Nominal Diameter	Clamp Load	Torque Dry*	Torque Lubed*
	[in]	[lbs]	[in-lbs,**ft-lbs**]	[in-lbs,**ft-lbs**]
2-56	.0860	62	1.0	0.8
4-40	.1120	100	2.5	1.5
6-32	.1380	150	4.0	3.0
8-32	.1640	250	8.0	6.0
10-32	.1900	360	14	10
1/4-20	.2500	550	27	20
1/4-28	.2500	730	36	27
5/16-18	.3125	800	50	37
5/16-24	.3125	1240	77	58
3/8-16	.3750	1330	100	75
3/8-24	.3750	1330	100	75
7/16-20	.4375	3120	**23**	**17**
1/2-20	.5000	3730	**31**	**23**
*Maximum Error of ±10%				
**AN316 or equivalent				

Figure 20 - 7 Recommended Jam Nut Torque and Clamp Load Specifications (Applicable to Fasteners With a Minimum Tensile Strength Greater Than 74 ksi)

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20-4. SAFETYING – MAINTENANCE PRACTICES

a. **General** – The word safetying is a term universally used in the aircraft industry. Briefly, safetying is defined as: "Securing by various means any nut, bolt, turnbuckle etc., on the aircraft so that vibration will not cause it to loosen during operation." These practices are not a means of obtaining or maintaining torque, rather a safety device to prevent the disengagement of screws, nuts, bolts, snap rings, oil caps, drain cocks, valves, and parts. Three basic methods are used in safetying: safety-wire, cotter pins, and self-locking nuts. Retainer washers and pal nuts are also sometimes used.

1. Wire, either soft brass or steel, is used on cylinder studs, control cable turnbuckles, and engine accessory attaching bolts.
2. Cotter pins are used on aircraft and engine controls, landing gear, or any other point where a turning or actuating movement takes place.

b. **Safety Wire**

WARNING

Do not use stainless steel, monel, carbon steel, or aluminum alloy safety wire to secure emergency mechanisms such as switch handles, guard covering handles used on exits, fire extinguishers, emergency gear releases, or other emergency equipment. Some existing structural equipment or safety-of-flight emergency devices require copper or brass safety wire (0.020 inch diameter only). Where successful emergency operation of this equipment is dependent on shearing or breaking of the safety wire, particular care should be used to ensure that safetying does not prevent emergency operation.

1. There are two methods of safety wiring; the double-twist method that is most commonly used, and the single-wire method used on screws, bolts, and/or nuts in a closely-spaced or closed-geometrical pattern such as a triangle, square, rectangle, or circle. The single-wire method may also be used on parts in electrical systems and in places that are difficult to reach (See Figure 20 - 8).

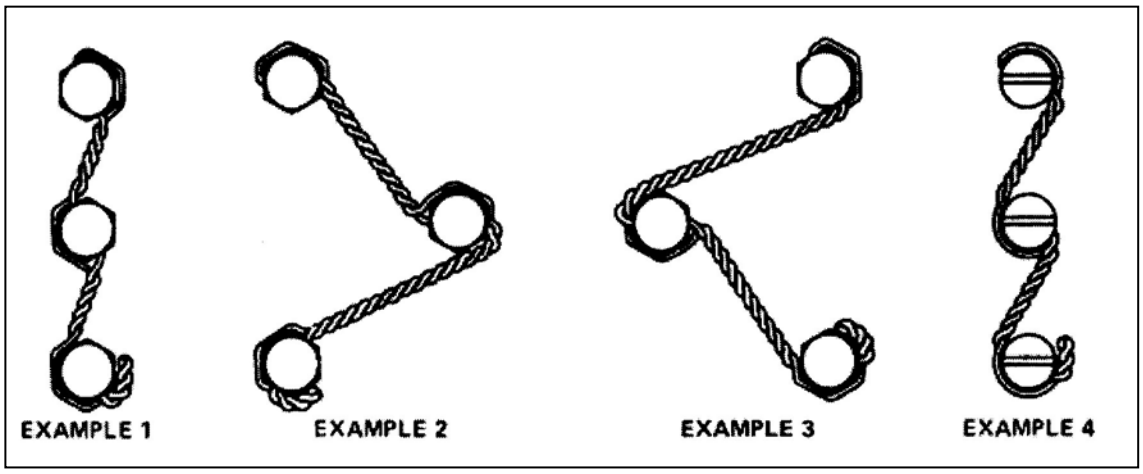

Figure 20 - 8 Examples of Safety Wiring "Safetying"

2. When using the double-twist method of safety wiring, 0.032 in. minimum diameter wire should be used on parts that have a hole diameter larger than 0.045 in. Safety

wire of 0.020 in. diameter (double strand) may be used on parts having a nominal hole diameter between 0.045 and 0.062 in. with a spacing between parts of less than 2 in. When using the single-wire method, the largest size wire that the hole will accommodate should be used. Copper wire (0.020 in. diameter), aluminum wire (0.031 in diameter), or other similar wire called for in specific technical orders, should be used as seals on equipment such as first-aid kits, portable fire extinguishers, emergency valves, or oxygen regulators. See Figure 20 - 9.

CAUTION

Care should be taken not to confuse steel with aluminum wire.

3. A secure seal indicates that the component has not been opened. Some emergency devices require installation of brass or soft copper shear safety wire. Particular care should be exercised to ensure that the use of safety wire will not prevent emergency operation of the devices.

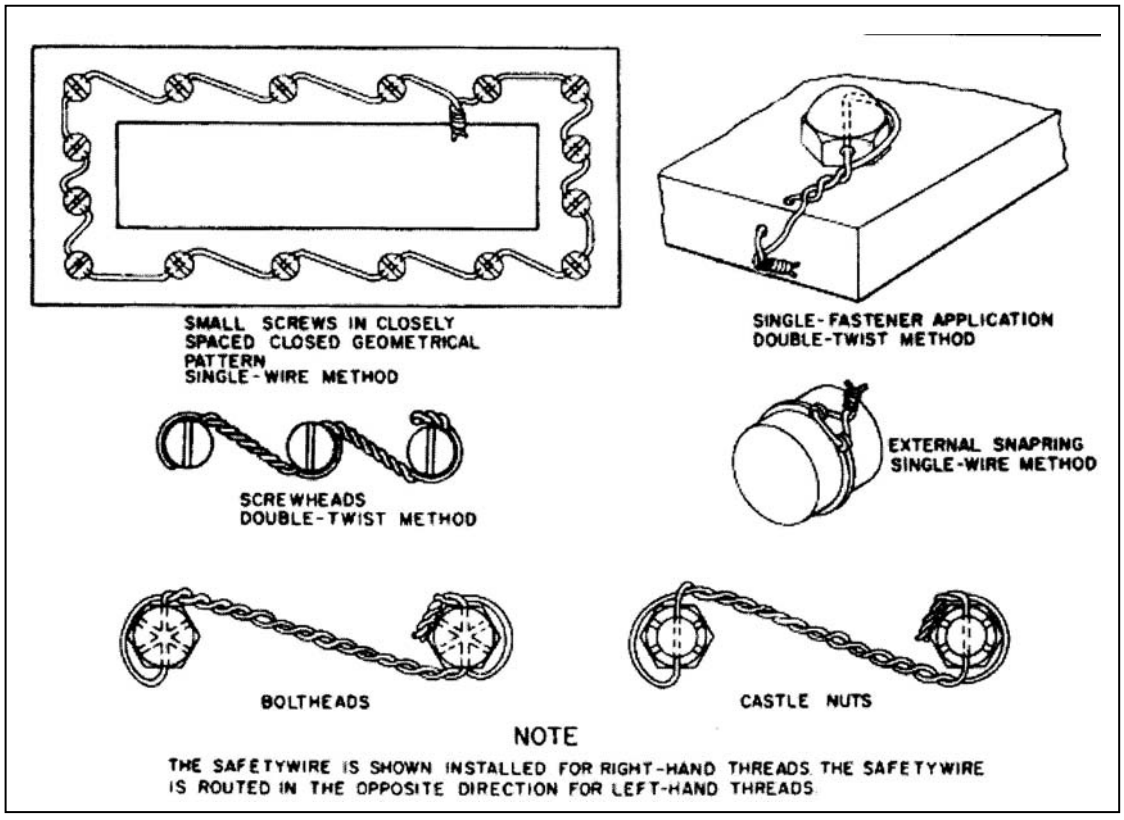

Figure 20 - 9 Securing Screws, Nuts, Bolts, and Snap Rings

20-5. SAFETY WIRING PROCEDURES

a. There are many combinations of safety wiring with certain basic rules common to all applications. These rules are as follows:

1. When bolts, screws, or other parts are closely grouped, it is more convenient to safety wire them in series. The number of bolts, nuts, screws, etc., that may be wired together depends on the application.
2. Drilled bolt heads and screws need not be safety wired if installed with self-locking nuts.
3. To prevent failure due to rubbing or vibration, safety wire must be tight after installation.
4. Safety wire must be installed in a manner that will prevent the tendency of the part to loosen.
5. Safety wire must never be over-stressed. Safety wire will break under vibrations if twisted too tightly. Safety wire must be pulled taut when being twisted and maintain a light tension when secured.
6. Safety-wire ends must be bent under and inward toward the part to avoid sharp or projecting ends, which might present a safety hazard.
7. Safety wire inside a duct or tube must not cross over or obstruct a flow passage when an alternate routing can be used.

b. Check the units to be safety wired to make sure that they have been correctly torqued and that the wiring holes are properly aligned to each other. When there are two or more units, it is desirable that the holes in the units be aligned to each other. Never over-torque or loosen to obtain proper alignment of the holes. It should be possible to align the wiring holes when the bolts are torqued within the specified limits. Washers may be used to establish proper alignment. However, if it is impossible to obtain a proper alignment of the holes without under-torquing or over-torquing, try another bolt that will permit proper alignment within the specified torque limits.

c. To prevent mutilation of the twisted section of wire when using pliers, grasp the wires at the ends. Safety wire must not be nicked, kinked, or mutilated. Never twist the wire ends off with pliers. When cutting off ends, leave at least four to six complete turns ($^1/_2$ to $^5/_8$ inch long) after the loop. When removing safety wire, never twist the wire off with pliers. Cut the safety wire close to the hole, exercising caution.

d. Install safety wire where practicable with the wire positioned around the head of the bolt, screw, or nut, and twisted in such a manner that the loop of the wire fits closely to the contour of the unit being safety wired.

20-6. TWISTING WITH SPECIAL TOOLS

Twist the wire with a wire twister as follows.

CAUTION

When using wire twisters and the wire extends 3 in. beyond the jaws of the twisters, loosely wrap the wire around the pliers to prevent whipping and possible personal injury. Excessive twisting of the wire will weaken the wire.

a. Grip the wire in the jaws of the wire twister, and slide the outer sleeve down with your thumb to lock the handles of the spring-loaded pin.
b. Pull the knob to make the spiral rod spin and twist the wire.
c. Squeeze handles together to release the wire.

20-7. SECURING OIL CAPS, DRAIN COCKS, AND VALVES

When securing oil caps and drain cocks, the safety wire should be anchored to an adjacent fillister-head screw. See Figure 20 - 10 Securing Oil Caps, Drain Cocks, and Valves. This method of safety wiring is applied to wing nuts, filler plugs, single-drilled head bolts, fillister-head screws, etc., that are safety wired individually. When securing valve handles in the vertical position, the wire is looped around the threads of the pipe leading into one side of the valve, double-twisted around the valve handle, and anchored around the threads of the pipe leading into the opposite side of the valve. When castle nuts are to be secured with safety wire, tighten the nut to the low side of the selected torque range, unless otherwise specified. If necessary, continue tightening until a slot lines with the hole. In blind tapped hole applications of bolts or castle nuts on studs, the safety wiring should be in accordance with the general instructions of this chapter. Hollow-head bolts are safetyed in the manner prescribed for regular bolts.

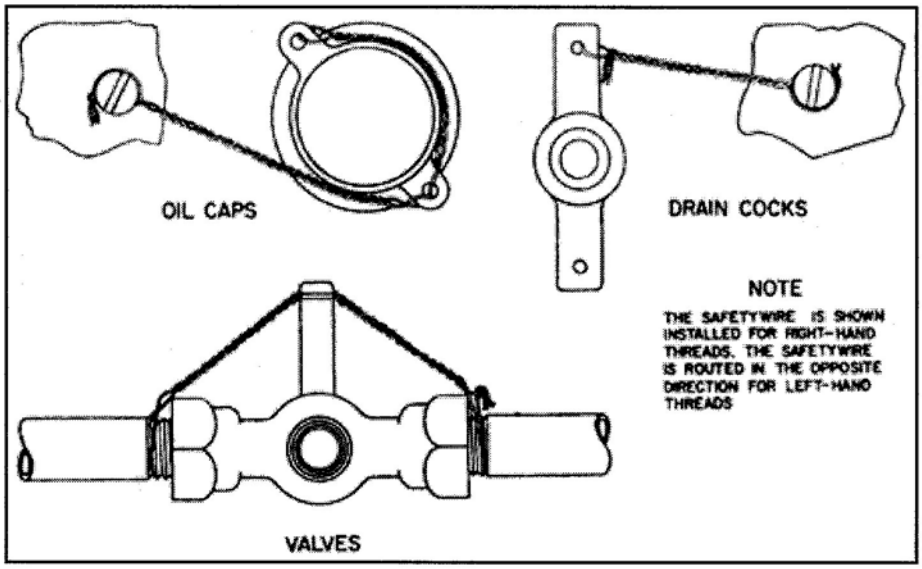

Figure 20 - 10 Securing Oil Caps, Drain Cocks, and Valves

NOTE

Do not loosen or tighten properly tightened nuts to align safety-wire holes.

NOTE

Although there are numerous safety wiring techniques used to secure aircraft hardware, practically all are derived from the basic examples shown in this text.

20-8. PROPER SECURING OF FITTINGS

a. There are several examples on how to secure fittings listed in this section.

1. Examples 1, 2, 3, and 4 apply to all types of bolts, fillister-head screws, square-head plugs, and other similar parts, which are wired so that tightening of the other part

counteracts the loosening tendency of either part (see Figure 20 - 11). The direction of twist from the second to the third unit is counterclockwise in examples 1, 3, and 4 to keep the loop in position against the head of the bolt. The direction of twist from the second to the third unit in example 2 is clockwise to keep the wire in position around the second unit. The wire entering the hole in the third unit will be the lower wire, except as in example 2. By making a counterclockwise twist after it leaves the hole, the loop will be secured in place around the head of that bolt.

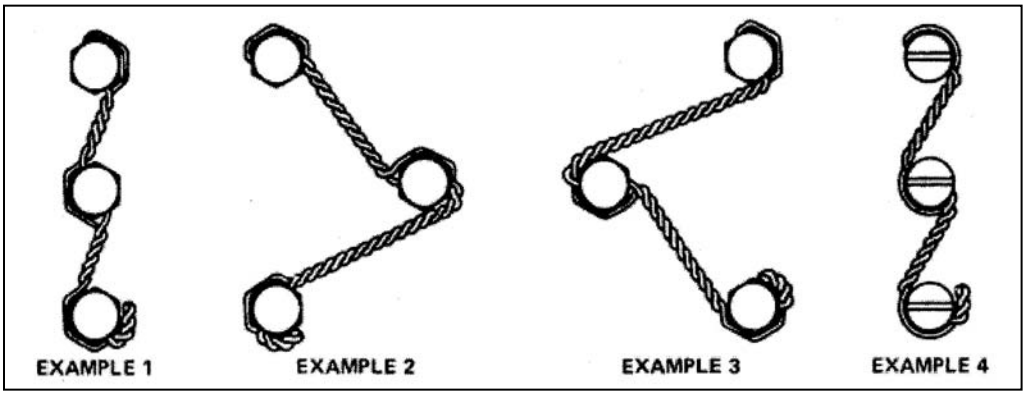

Figure 20 - 11 Securing Bolts, Screws, Plugs, etc.

2. Examples 5, 6, 7, & 8 show methods for wiring various standard items. See Figure 20 - 12 Securing Various Standard Items. Note that wire may be wrapped over the unit rather than around it when wiring castle nuts or on other items when there is a clearance problem.

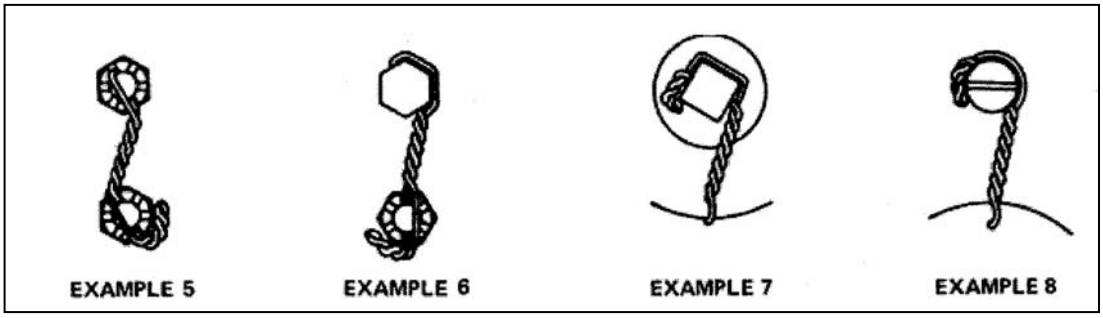

Figure 20 - 12 Securing Various Standard Items

3. Example 9 (see Figure 20 - 13) shows the method for wiring bolts in different planes.. Note that wire should always be applied so that tension is in the tightening direction. Hollow-head plugs shall be wired as shown with the tab bent inside the hole to avoid snags and possible injury to personnel working on the engine.

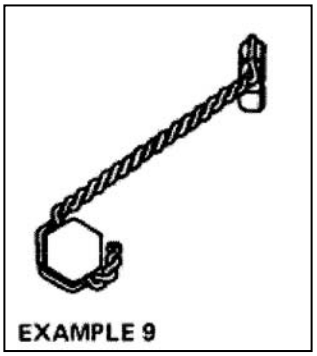

Figure 20 - 13 Wiring Bolts in Different Planes

4. Examples 12 and 13 show methods for attaching lead seal to protect critical adjustments. See Figure 20 - 14.

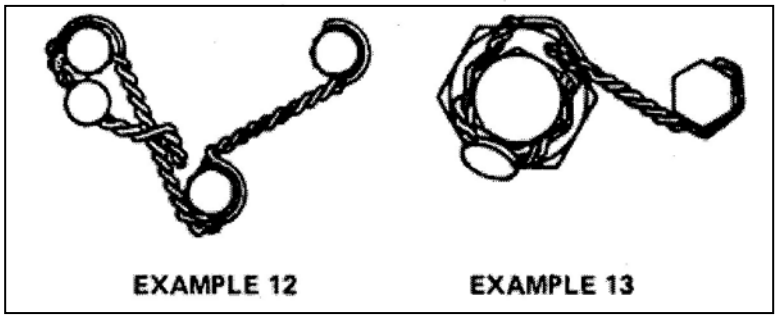

Figure 20 - 14 Attaching Lead Seal

5. Example 14 shows bolt wired to a right-angle bracket with the wire wrapped around the bracket. See Figure 20 - 15 Wiring Brackets, Connecting Rods, Rigid Tubes.

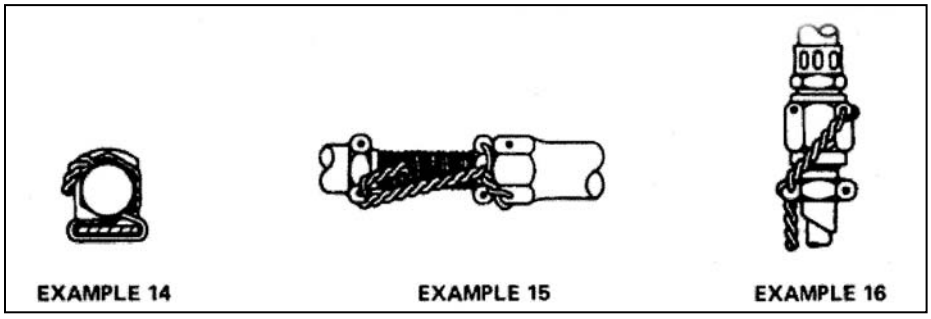

Figure 20 - 15 Wiring Brackets, Connecting Rods, Rigid Tubes

6. Example 15 shows the correct method for wiring adjustable connecting rods. See Figure 20 - 15.
7. Example 16 shows the correct method for wiring the coupling nut on flexible line to the straight connector brazed on rigid tube. See Figure 20 - 15.

b. Fittings incorporating wire lugs shall be wired as shown in Examples 17 and 18. Where no lock-wire lug is provided, wire should be applied as shown in examples 19 and 20 with caution being exerted to ensure that the wire is wrapped tightly around the fitting. See Figure 20 - 16.

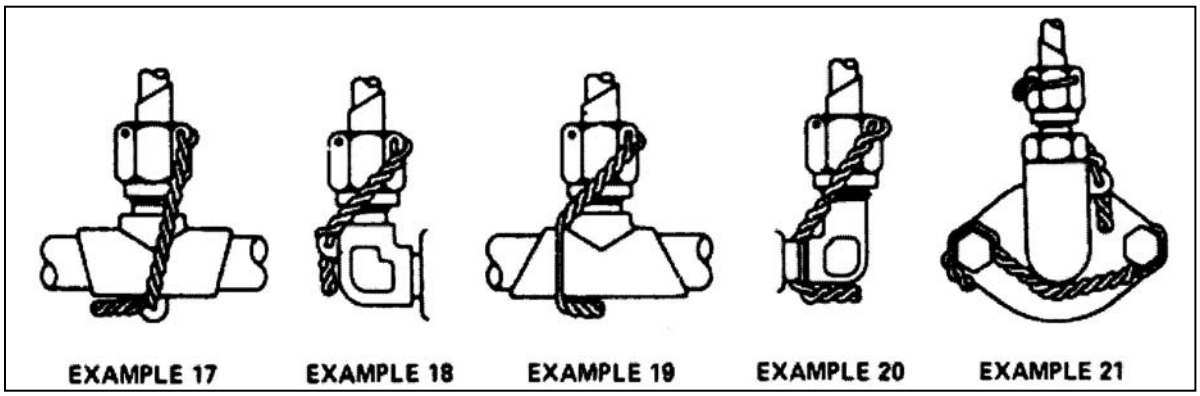

Figure 20 - 16 Fittings Incorporating Wire Lugs

c. Small size coupling nuts shall be wired by wrapping the wire around the nut and inserting it through the holes as shown.

d. Coupling nuts attached to straight connectors shall be wired as shown when the hex is an integral part of the connector. Coupling nuts on a tee shall be wired as shown so that tension is always in the tightening direction.

20-9. SECURING WITH COTTER PINS

a. Cotter pins are used to secure such items as bolts, screws, pins, and shafts. The use of cotter pins is favored because they can be removed and installed quickly. The diameter of the cotter pins selected for any application should be the largest size that will fit consistent with the diameter of the cotter pin hole and/or the slots in the nut. Cotter pins should not be re-used on aircraft.

b. To prevent injury during and after pin installation, the end of the cotter pin can be rolled and tucked as shown in Figure 20 - 17 and Figure 20 - 18.

c. When tightening castellated nuts on bolts, begin by applying the minimum torque specified. If the cotter pin hole does not align with the slots in the nut, continue to tighten the nut until the next set of slots is aligned with the hole. In special cases where there is risk of the threads stripping or the bolt shearing, the maximum torque value specified may not be exceeded. For most installations, the nut may be over tightened to permit the slots to align with the hole.

NOTE

In using the method of cotter pin safetying ensure the prong, bent over the bolt, is seated firmly against the bolt shank and does not exceed bolt diameter. Also, when the prong is bent over the nut, ensure the bent prong is down and firmly flat against the nut and does not contact the surface of the washer.

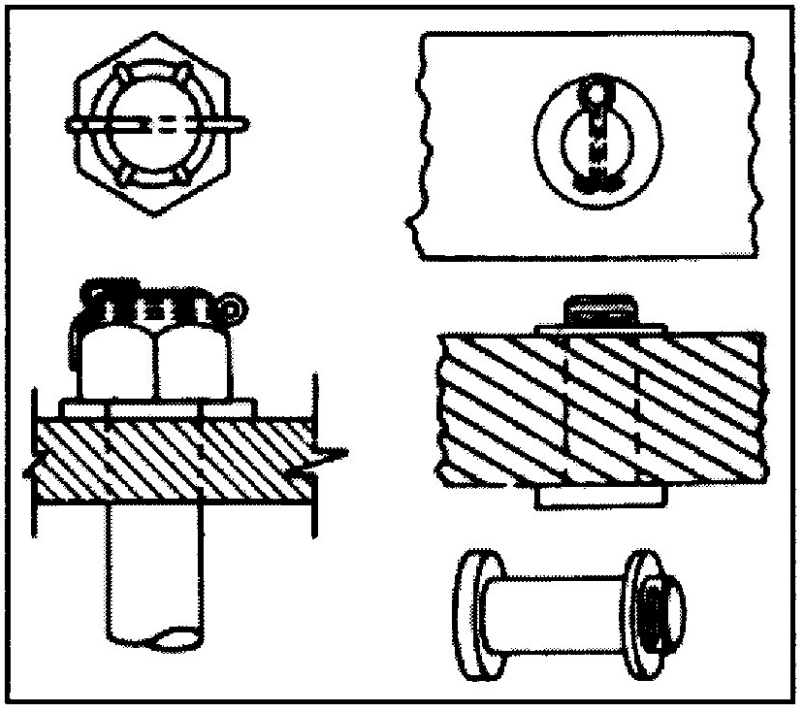

Figure 20 - 17 Securing with Cotter Pins

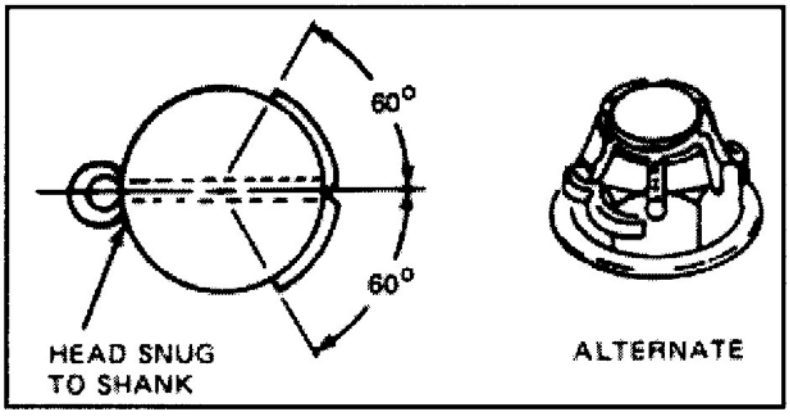

Figure 20 - 18 Alternate Method for Securing with Cotter Pins

20-10. CONTROL CABLE WIRE BREAKAGE AND CORROSION LIMITATIONS

Aircraft control cables in the Cessna 350 (LC42-550FG) are fabricated from corrosion-resistant steel wire of flexible-type construction. For example, $^{1}/_{16}$ 7 X 7 MIL –W-83420/2 and $^{1}/_{8}$ in. 7 X 19 MIL-W-83420 cables are used in the rudder control circuit.

a. **Cable Definitions –** The following cable components are defined in accordance with Military Specifications MIL-W-83420, MIL-C-18375, and MIL-W-87161.

1. Wire Center – The center of all strands shall be an individual wire and shall be designated as a wire center.
2. Strand Center or Core – A strand center is a single, straight strand made of preformed wires, similar to the other strands comprising the cable in arrangement and number of wires.
3. Independent Wire Rope Center (IWRC) 7 by 7 – A cable or wire rope of six strands of seven wires each, twisted or laid around a strand center or core consisting of seven wires.

b. **Flexible Cables –** Flexible, preformed, corrosion-resistant, Type I, composition B cables, MIL-W-87161, MIL-W-83420, and MIL-C-18375 are manufactured from steel made by the electric-furnace process. These cables are of the 3 by 7, 7 by 7, 7 by 19, or 6 by 19 IWRC construction, according to the diameter. The 3 by 7 cable consists of three strands of seven wires each. There is no core in this construction. The 7 by 7 cable consists of six strands, of seven wires each, laid around a center strand of seven wires. The wires are laid so as to develop a cable that has the greatest bending and wearing properties. The 7 by 7 cable has a length of lay of not more than eight times or less than six times the cable diameter. The 7 by 19 cable consists of six strands laid around a center strand in a clockwise direction. The wires composing the seven individual strands are laid around a center wire in two layers. The center core strand consists of a lay of six wires laid around the central wire in a clockwise direction and a layer of 12 wires laid around this in a clockwise direction. The six outer strands of the cable consist of a layer of six wires laid around the center wire in a counterclockwise direction and a layer of 12 wires laid around this in a counterclockwise direction.

c. **Cable Replacement –** Replace control cables when they become worn, distorted, corroded, or otherwise damaged. If spare cables are not available, prepare exact duplicates of the damaged cable. Use materials of the same size and quality as the original. Standard swaged cable terminals develop the full cable strength and may be substituted for the original terminals wherever practical.

d. **Mechanically Fabricated Cable Assemblies**

1. **Swage-Type Terminals –** Swage-type terminals, manufactured in accordance with AN, are suitable for use in civil aircraft up to, and including, maximum cable loads. When swaging tools are used, it is important that all the manufacturers' instructions, including "go and no-go" dimensions, be followed in detail to avoid defective and inferior swaging. Observance of all instructions should result in a terminal developing the full-rated strength of the cable. Critical dimensions, both before and after swaging, must be checked.
2. When swaging terminals onto cable ends, observe the following procedures:

a) Cut the cable to the proper length allowing for growth during swaging. Apply a preservative compound to the cable ends before insertion into the terminal barrel.

NOTE

Never solder cable ends to prevent fraying because the presence of the solder will greatly increase the tendency of the cable to pull out of the terminal.

b) Insert the cable into the terminal approximately 1 in., and bend toward the terminal. Push the cable end entirely into the terminal barrel. The bending actions puts a kink or bend in the cable end and provides enough friction to hold the terminal in place until the swaging operation can be performed. Bending also tends to separate the strands inside the barrel, thereby reducing the strain on them.

(Cross reference AN to MS: AN-666 to MS 21259, AN-667 to MS 20667, AN-668 to MS 20668, AN-669 to MS 21260.)

Key:
A = Cable size (inches)
B = Wire strands
C = Outside diameter before swaging
D = Bore diameter before swaging
E = Bore length before swaging
F = Swaging length before swaging
G = Minimum breaking strength (pounds) after swaging
H = Shank diameter after swaging - Note: Use gauges in kit for checking diameters.

A	B	C	D	E	F	G	H
1/16	7X7	0.160	0.078	1.042	0.969	480	0.138
3/32	7X7	0.218	0.109	1.261	1.188	920	0.190
1/8	7X19	0.250	0.141	1.511	1.438	2,000	0.219
5/32	7X19	0.297	0.172	1.761	1.688	2,800	0.250
3/16	7X19	0.359	0.203	2.011	1.938	4,200	0.313
7/32	7X19	0.427	0.234	2.261	2.188	5,600	0.375
1/4	7X19	0.494	0.265	2.511	2.438	7,000	0.438
9/32	7X19	0.563	0.297	2.761	2.688	8,000	0.500
5/16	7X19	0.635	0.328	3.011	2.938	9,800	0.563
3/8	7X19	0.703	0.390	3.510	3.438	14,400	0.625

Figure 20 - 19 Straight-Shank Terminal Dimensions

c) Accomplish the swaging operation in accordance with the instructions furnished by the manufacturer of the swaging equipment.

d) Inspect the terminal after swaging to determine that it is free from die marks and splits and is not out-of-round. Check for cable slippage in the terminal and for cut or broken wire strands.
e) Using a "go no-go" gauge or a micrometer, check the terminal shank diameter.

3. **Nicopress Process –** A patented process using copper sleeves may be used up to the full rated strength of the cable when the cable is looped around a thimble.
   a) Before undertaking a Nicopress splice, determine the proper tool and sleeve for the cable to be used. Refer to proper references for details on sleeves, tools, and the number of presses required for the various sizes of aircraft cable. The tool must be in good working condition and properly adjusted to ensure a satisfactory splice.
   b) To compress a sleeve, have it well-centered in the tool groove with the major axis of the sleeve at right angles to the tool. If the sleeve appears to be out of line after the press is started, open the tool, re-center the sleeve, and complete the press.
4. **Lap Splice** – Lap or running splices may also be made with copper oval sleeves. When making such splices, it is usually necessary to use two sleeves to develop the full strength of the cable. The sleeves should be positioned as shown in Figure 20 - 20, and the compressions made in the order shown. As in the case of eye splices (see Figure 20 - 21), it is desirable to have the cable ends extend beyond the sleeves sufficiently to allow for the increased length of the compressed sleeves.

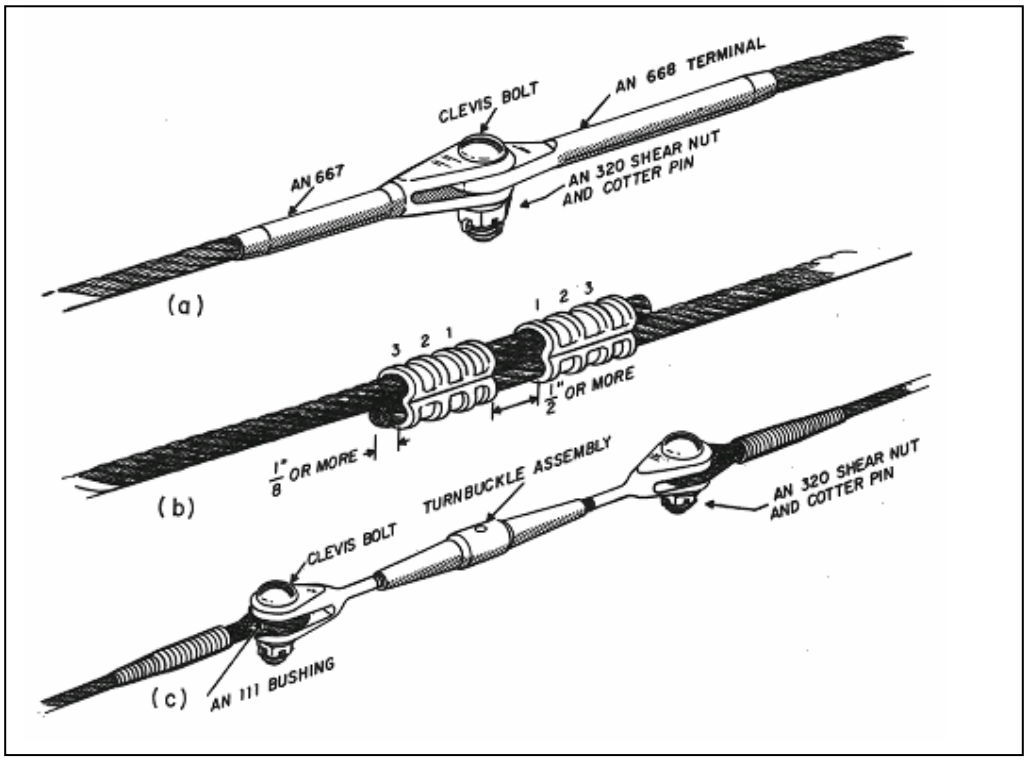

Figure 20 - 20 Lap Splices

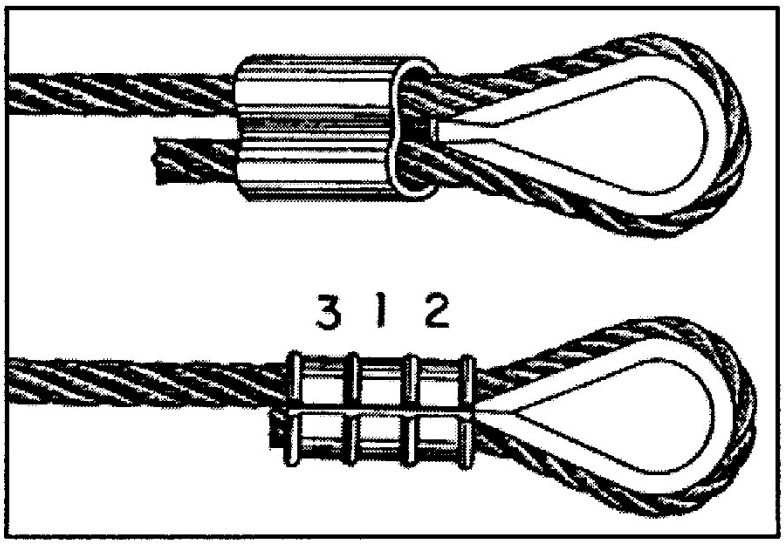

Figure 20 - 21 Typical Thimble-Eye Splice

e. **Terminal Gauge** – To make a satisfactory copper sleeve installation, it is important that the amount of sleeve pressure be kept uniform. The completed sleeves should be checked periodically with the proper gauge. Hold the gauge so that it contacts the major axis of the sleeve. The compressed portion at the center of the sleeve should enter the gauge opening with very little clearance, as shown in Figure 20 - 22. If it does not, the tool must be adjusted accordingly.

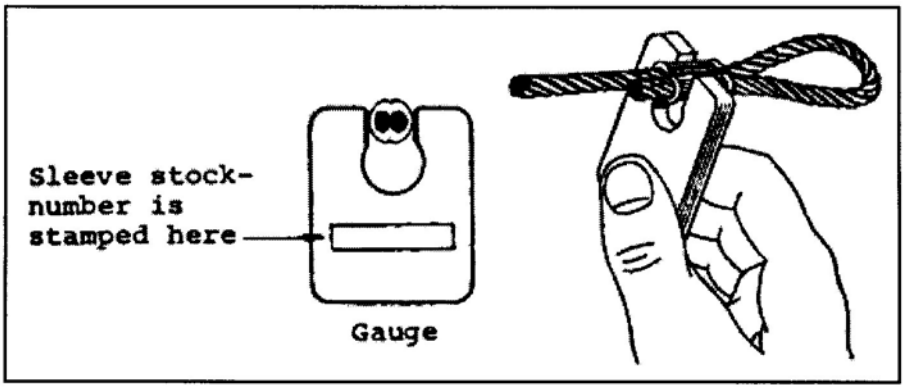

Figure 20 - 22 Typical Terminal Gauge

20-11. CABLE SYSTEM INSPECTION

Aircraft cable systems are subject to a variety of environmental conditions and deterioration. Wire or strand breakage is easy to visually recognize. Other kinds of deterioration such as wear, corrosion, and/or distortion are not easily seen, therefore, control cables should be removed periodically for a more detailed inspection.

a. At each annual or 100 hour inspection, all control cables must be inspected for broken wire strands. Any cable assembly that has one broken wire strand located in a critical fatigue area must be replaced.

b. A critical fatigue area is defined as the working length of a cable where the cable runs over, under, or around a pulley, sleeve, or through a fair-lead; or any section where the cable is flexed, rubbed, or worked in any manner; or any point within 1 ft. of a swaged-on fitting.

c. A swaged-on fitting can be an eye, fork, ball, ball and shank, ball and double shank, threaded stud, threaded stud and turnbuckle, compression sleeve, or any hardware used as a termination or end fitting on the cable. These fittings may be attached by various swaging methods such as rotary swaging, roll swaging, hydraulic pressing, and hand swaging tools. (See MIL-T-781.) The pressures exerted on the fittings during the swaging process sometimes pinch the small wires in the cable. This can cause premature failure of the pinched wires, resulting in broken wires.

d. Close inspection in these critical fatigue areas must be made by passing a cloth over the area that will snag on broken wires. This will clean the cable for a visual inspection, and detect broken wires if the cloth snags on the cable. Also, a very careful visual inspection must be made since a broken wire will not always protrude or stick out, but may lie in the strand and remain in the position of the helix as it was manufactured. Broken wires of this type may show up as a hairline crack in the wire. If a broken wire of this type is suspected, further inspection with a magnifying glass of 7 power or greater, is recommended. Figure 20 - 23 shows a cable with broken wires that was not detected by wiping, but was found during a visual inspection. The damage became readily apparent when the cable was removed and bent as shown in the figure below.

Figure 20 - 23 Cable Inspection Technique

e. Kinking of wire cable can be avoided if properly handled and installed. A cable taking a spiral shape as the result of unnatural twist causes kinking. One of the most common causes for this twist is improper unreeling and uncoiling. In a kinked cable, strands and wires are out of position, which creates unequal tension and brings excessive wear at this part of the cable. Even though the kink may be straightened so that the damage appears to be slight, the relative adjustment between the strands has been disturbed so that the cable cannot give maximum service and should be replaced. Inspect cables for a popped core or loose strands. Replace any cable that has a popped core or loose strands regardless of wear or broken wires.

f. External wear patterns will extend along the cable equal to the distance the cable moves at that location and may occur on one side of the cable or on its entire circumference. Replace flexible and non-flexible cables when the individual wires in each strand appear to blend together (outer wires worn 40 to 50 percent).

g. As wear is taking place on the exterior surface of a cable, the same condition is taking place internally, particularly in the sections of the cable which pass over pulleys and quadrants. This condition is not easily detected unless the strands of the cable are separated. This type of wear is a result of the relative motion between inner wire surfaces. Under certain conditions, the rate of this type of wear can be greater than that occurring on the surface.

h. Carefully examine any cable for corrosion, when it has a broken wire in a section that is not in contact with a wear-producing airframe component, such as a pulley, fair-lead, etc. If the surface of the cable is corroded, relieve cable tension and carefully force the cable open by reverse twisting, and visually inspect the interior. Corrosion on the interior strands of the cable constitutes failure, and the cable must be replaced. If no internal corrosion is detected, remove loose external rust and corrosion with a clean, dry, coarse-weave rag, or fiber brush. Do not use metallic wool or solvents to clean installed cables. Use of metallic wool will embed dissimilar metal particles in the cables and create further corrosion problems. Solvents will remove internal cable lubricant allowing cable strands to abrade and further corrode. After thorough cleaning, sparingly apply specification MIL-C-16173, grade 4, corrosion-preventive compound to the cable. Do not apply the material so thick that it will interfere with the operation of cables at fair-leads, pulleys, or grooved bellcrank areas.

i. Examine cable runs for incorrect routing, fraying, twisting, or wear at fair-leads, pulleys, anti-abrasion strips, and guards. Look for interference with adjacent structure, equipment, wiring, plumbing, and other controls. Inspect cable systems for binding, full travel, and security of attaching hardware. Check for slack in the cable system. Actuate the controls, and check for friction or hard movement that are indications that excessive cable tension exists.

NOTE

If the control movement is stiff after maintenance has been performed on the control surfaces, check for parallel cables twisted around each other or cables connected in reverse.

j. Check swaged terminal reference marks for an indication of cable slippage within the fitting. Inspect the fitting assembly for distortion and/or broken strands at the terminal. Ensure that all bearings and swivel fittings (bolted or pinned) pivot freely to prevent binding and subsequent failure. Check turnbuckles for proper thread exposure and broken or missing safety wires/clips.

k. Inspect pulleys for roughness, sharp edges, and presence of foreign material embedded in the grooves. Examine pulley bearings to ensure proper lubrication, smooth rotation, and freedom from flat spots, dirt, and paint spray. During the inspection, rotate the pulleys, which only turn through a small arc, to provide a new bearing surface for the cable. Maintain pulley alignment to prevent the cable from riding on the flanges and chafing

against guards, covers, or adjacent structure. Check all pulley brackets and guards for damage, alignment, and security.

l. Various cable system malfunctions may be detected by analyzing pulley conditions. These include such discrepancies as too much tension, misalignment, pulley bearing problems, and size mismatches between cables and pulleys. Examples of these conditions are shown in Figure 20 - 24.

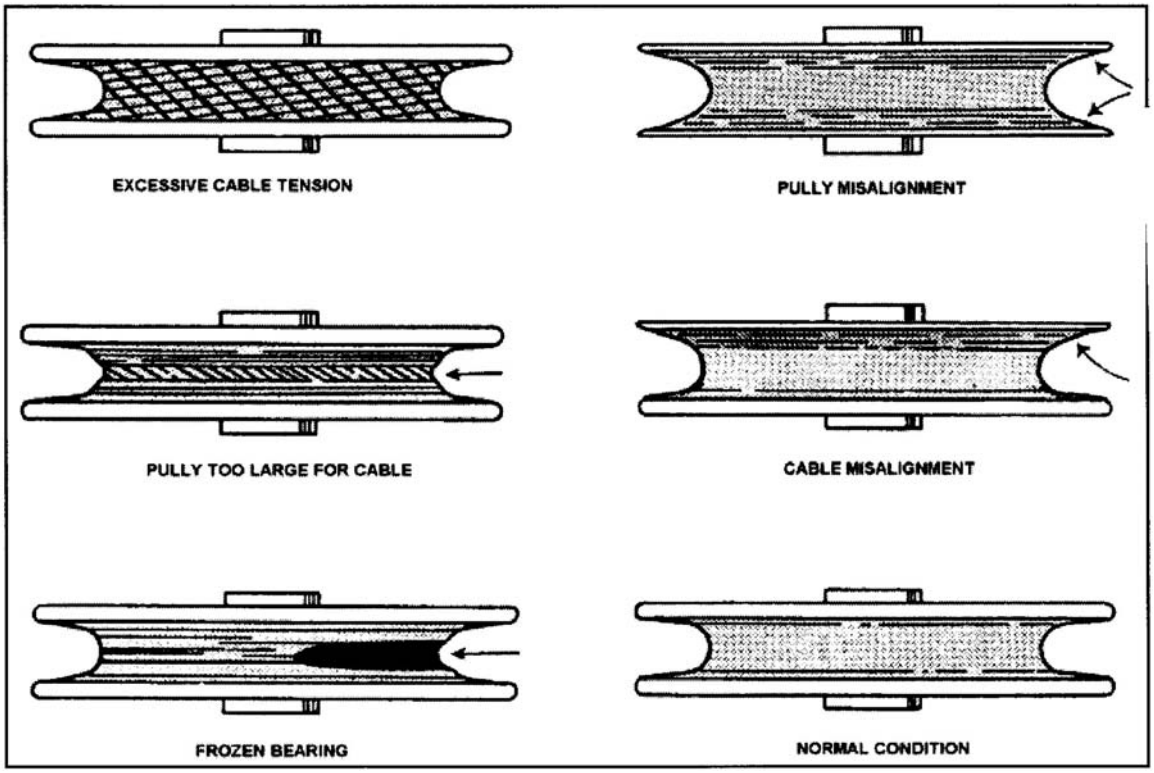

Figure 20 - 24 Pulley Wear Patterns

m. Inspect fair-leads for wear, breakage, alignment, cleanliness, and security. Examine cable routing at fair-leads to ensure that deflection angles are no greater than 3° maximum. Determine that all guides and anti-abrasion strips are secure and in good condition.

n. Examine pressure seals for wear and material deterioration. Seal guards should be positioned to prevent jamming of a pulley in case the pressure seal fails and pieces slide along the cable.

20-12. CORROSION AND RUST PREVENTION

a. To ensure a satisfactory service life for aircraft control cables, use a cable lubricant to reduce internal friction and prevent corrosion.

b. Coat the cable with rust-preventive oil, and wipe off any excess. Lubrication and corrosion preventive treatment of cables may be effected simultaneously by application of compound MIL-C-16173, grade 4. MIL-C-16173 compound should be brushed, sprayed, or wiped on the cable to the extent it penetrates into the strands and adequately covers the cable surfaces. It will dry “tack free” in 24 hours at 77 °F.

20-13. CABLE MAINTENANCE

Frequent inspections and preservation measures such as rust-prevention treatments, will help to extend cable service life. Where cables pass through fair-leads or over pulleys, remove any accumulated heavy coatings of corrosion-prevention compound. Provide corrosion protection for these cable sections by lubricating with a light coat of grease or general-purpose, low-temperature oil.

20-14. CABLE TENSION ADJUSTMENT

Carefully adjust the control cable tension in accordance with the airframe requirements. If necessary, compensate for extreme surface temperature variations that may be encountered if the aircraft is operated primarily in unusual geographic or climatic conditions such as arctic, arid, or tropical locations. Use rigging pins and gust locks, as necessary, to ensure satisfactory results. At the completion of rigging operations, check turnbuckle adjustment and safetying in accordance with standard practices and information in this chapter.

20-15. TURNBUCKLES

A turnbuckle is a device used in cable systems to provide a means of adjusting tension. Turnbuckles have barrel-shaped sleeves with internal left- and right-hand threads at opposite ends. The cables, with terminals attached, are made to such a length that, when the turnbuckle is adjusted to give the specified cable tension, a sufficient number of threads on the terminal ends are screwed into the barrel to hold the load. The clip-locking turnbuckle and its associated parts are identical to standard AN and MS parts except for a slot grooved on the interior of the barrel and the shanks of the forks, eyes, etc. The clip-locking turnbuckle parts have the following drawing numbers: MS21251, turnbuckle body; MS21252, turnbuckle clevis end; MS21253, turnbuckle clevis end (for bearing); NAS649 and NAS651, turnbuckle clip; MS21254 and NAS648, turnbuckle eye (for pin); MS21255 and NAS647, turnbuckle eye end (for wire rope); NAS645 and NAS646, turnbuckle fork; MS21256, turnbuckle barrel locking clip; AN130-170, turnbuckle assemblies; and MS21259 and MS21260, terminal, wire rope, stud.

NOTE

Turnbuckles showing signs of thread distortion or bending should be replaced.

Turnbuckle ends are designed to provide the specified cable tension on a cable system. A bent turnbuckle would place undesirable stress on the cable, impairing the function of the turnbuckle.

20-16. TURNBUCKLE INSTALLATION

When installing cable system turnbuckles, it is necessary to screw both threaded terminals into the turnbuckle barrel an equal amount. It is essential that turnbuckle terminals be screwed into the barrel so that not more than three threads on the terminal are exposed. On initial installation, the turnbuckle terminals should not be screwed inside the turnbuckle barrel more than four threads.

20-17. WITNESS HOLE

Some manufacturers of turnbuckles incorporate a “witness hole” in the turnbuckle barrel to ensure that the threaded cable terminals are screwed far enough into the barrel. The “witness hole” can be inspected visually or by using a piece of safety wire as a probe.

20-18. SAFETY METHODS FOR TURNBUCKLES

Safety all turnbuckles with safety wire using either the double or single-wrap method or with any appropriately approved special safetying device complying with the requirements of FAA Technical Standard Order TSO-C21. The swaged and unswaged turnbuckle assemblies are covered by AN standard drawings. Do not reuse safety wire. Adjust the turnbuckle to the correct cable tension so that no more than three cable threads are exposed on either side of the turnbuckle barrel.

a. **Double-Wrap Method** – Of the methods using safety wire for safetying turnbuckles, the method described here is preferred, although either of the other methods described is satisfactory. The method of double-wrap safetying is shown in Figure 20 - 25.

1. Use two separate lengths of wire. Run one end of the wire through the hole in the barrel of the turnbuckle, and bend the ends of the wire toward opposite ends of the turnbuckle.
2. Pass the second length of the wire into the hole in the barrel, and bend the ends along the barrel on the side opposite the first. Spiral the two wires in opposite directions around the barrel to cross each other twice between the center hole and the ends.
3. Pass the wires at the end of the turnbuckle in opposite directions through the hole in the turnbuckle eyes or between the jaws of the turnbuckle fork, as applicable, laying one wire along the barrel and wrapping the other at least four times around the shank of the turnbuckle and binding the laid wires in place before cutting the wrapped wire off.
4. Wrap the remaining length of safety wire at least four turns around the shank, and cut it off. Repeat the procedure at the opposite end of the turnbuckle.
5. When a swaged terminal is being safety wired, pass the ends of both wires through the hole provided in the terminal for this purpose, and wrap both ends around the shank as previously described. If the hole is not large enough to allow passage of both wires, pass the wire through the hole, and loop it over the free end of the other wire, and then wrap both ends around the shank as previously described. Another satisfactory double-wrap method is similar to the previous method, except that the spiraling of the wires is omitted as shown in Figure 20 - 25.

b. **Single-Wrap Method** – The single-wrap methods described in the following paragraphs and as illustrated in Figure 20 - 25 are acceptable but are not the equal of the double-wrap methods.

1. **Single-Wrap (Spiral) –** Pass a single length of wire through the cable eye or fork or through the hole in the swaged terminal at either end of the turnbuckle assembly. Spiral each of the wire ends in opposite directions around the first half of the turnbuckle barrel, so as to cross each other twice. Thread both wire ends through the hole in the middle of the barrel so that the third crossing of wire ends is in the hole. Spiral the two wire ends in opposite directions around the remaining half of the turnbuckle, crossing them twice. Pass one wire end through the cable eye or fork or through the hole in the swaged terminals, in the manner previously described. Wrap both wire ends around the shank at least four turns each, cutting off excess wire. This method is shown in view (C), Figure 20 - 25.
2. **Single-Wrap –** For the method shown in view (D), Figure 20 - 25, pass one length of wire through the center hole of the turnbuckle, and bend the wire ends toward opposite ends of the turnbuckle. Pass each wire end through the cable eye or fork or through the hole in the swaged terminal, and wrap each wire around the shank for at least four turns, cutting off excess wire. After safetying, no more than three threads of the turnbuckle threaded terminal should be exposed.

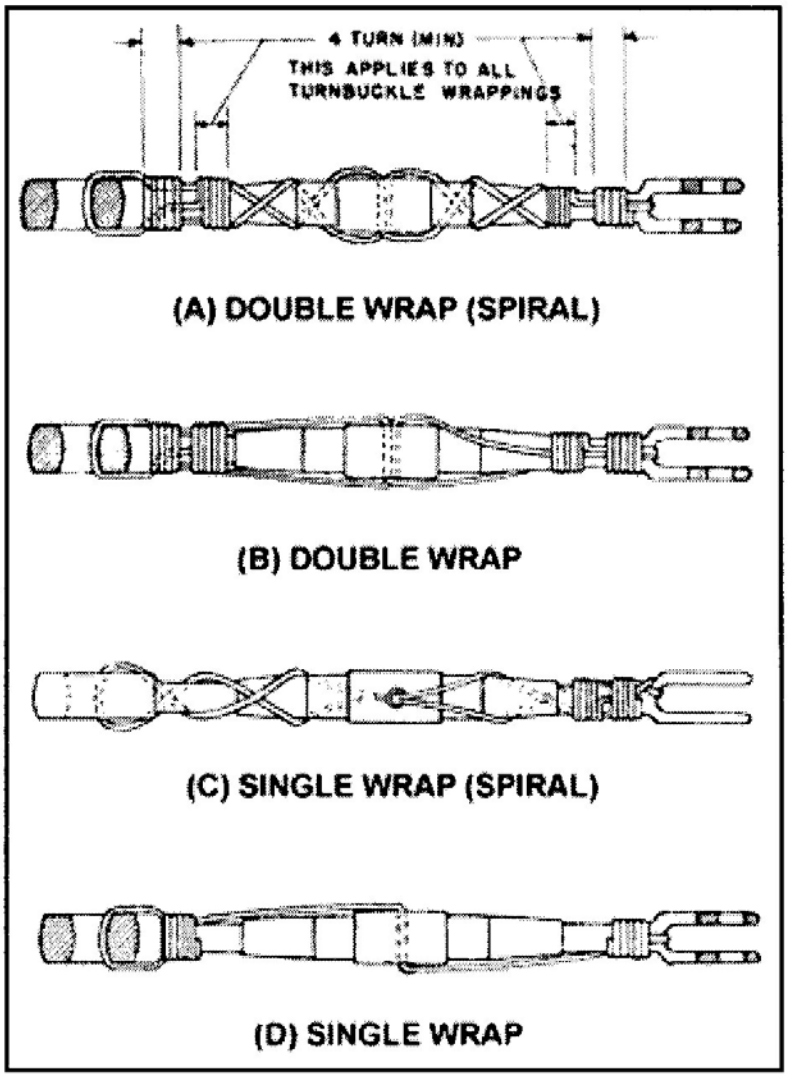

Figure 20 - 25 Safetying Turnbuckles

20-19. SECURED SAFETY-WIRED TURNBUCKLES

a. Before securing turnbuckles, threaded terminals should be screwed into the turnbuckle barrel until no more than three threads of either terminal are outside the barrel. After the turnbuckle has been adjusted for proper cable tension, two pieces of safety wire are inserted with half the wire length into the hole in the center of the turnbuckle barrel. The safety wires are bent so that each wire extends half the length of the turnbuckle on top and half on bottom. The ends of the wires are passed through the hole in the turnbuckle eyes or between the jaws of the turnbuckle fork, as applicable. The wires are then bent toward the center of the turnbuckle, and each wire is wrapped around the shank four times, binding the wrapping wires in place as shown in Figure 20 - 26.

b. When a swaged terminal is being secured, one wire is passed through the hole in the terminal and is looped over the free end of the other wire and both ends wrapped around the shank. All lock wire used in the safetying of turnbuckles should be carbon steel, corrosion-resistant steel, nickel-chromium iron alloy (Inconel), nickel-copper alloy

(Monel), or aluminum alloy. For safety cable diameter of safety wire size and material, refer to Figure 20 - 27.

c. Care should be exercised when safety wiring, particularly where corrosion will present a problem, because smaller wire sizes tend to crack when twisted.

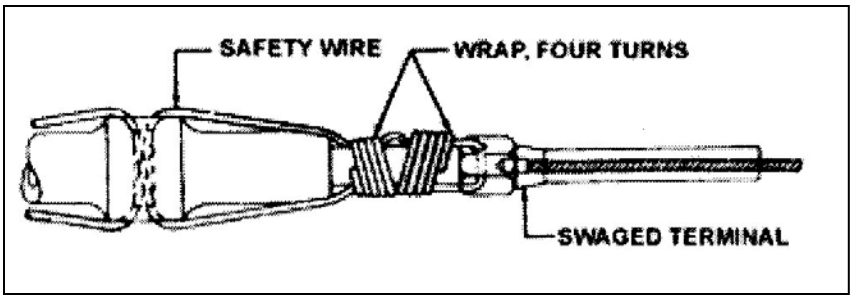

Figure 20 - 26 Securing Turnbuckles

Cable Size	Type of Wrap	Diameter of Safety Wire	Material (Annealed Condition)
1/16	Single	0.040	Copper, brass.*
3/32	Single	0.040	Copper, brass.*
1/8	Single	0.040	Stainless steel, Monel and “K” Monel.
1/8	Double	0.040	Copper, brass.*
1/8	Single	0.057 min.	Copper, brass.*
5/32 and greater	Double	0.040	Stainless steel, Monel and “K” Monel.
5/32 and greater	Single	0.057 min.	Stainless steel, Monel and “K” Monel.
5/32 and greater	Double	0.0512	Copper, brass
*** Galvanized or tinned steel, or soft iron wires are also acceptable.**			

Figure 20 - 27 Turnbuckle Safetying Guide

20-20. SPECIAL LOCKING DEVICES

Several turnbuckle-locking devices are available for securing turnbuckle barrels such as wire-locking clips. Persons intending to use a special device must ensure that the turnbuckle assembly has been designed to accommodate such devices. A typical unit is shown in Figure 20 - 28. When special locking devices are not readily available, the use of safety wire is acceptable.

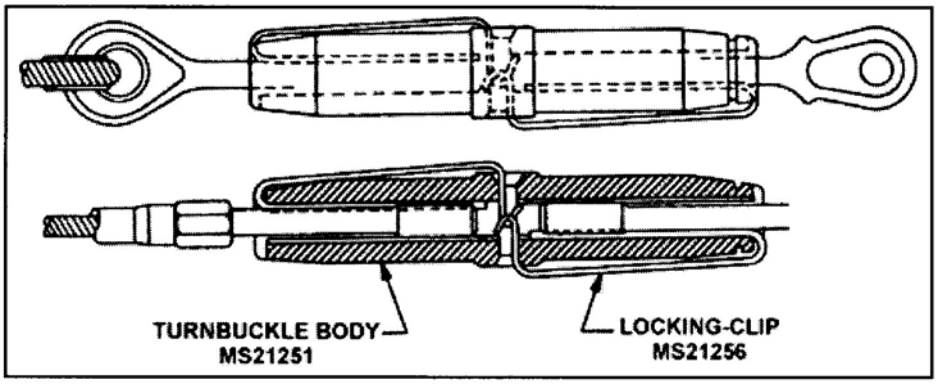

Figure 20 - 28 Clip-Type Locking Device

a. **Assembling and Securing Clip-Lock Turnbuckles –** Wire clip-locking turnbuckles are assembled and secured in the following ways (See Figure 20 - 29).

1. Engage the threads of the turnbuckle barrel with the threads of the cable terminal, and turn the barrel until proper cable tension is reached.
2. Align the slot in the barrel with the slot in the cable terminal.
3. Hold the lock clip between the thumb and forefinger at the loop end, and insert the straight end of the clip into the opening formed by the aligned slots. See Figure 20 - 30.
4. Bring the hook end of the lock clip over the hole in the center of the turnbuckle barrel, and seat the hook loop into the hole.
5. Apply pressure to the hook shoulder to engage the hook lip in the turnbuckle barrel and to complete safety locking of one end of the turnbuckle.

NOTE

Repeat the above steps to safety lock the opposite end of the turnbuckle. Either lock clips may be inserted in the same turnbuckle barrel hole or they may be inserted in opposite holes. However, do not reverse-wire locking clips.

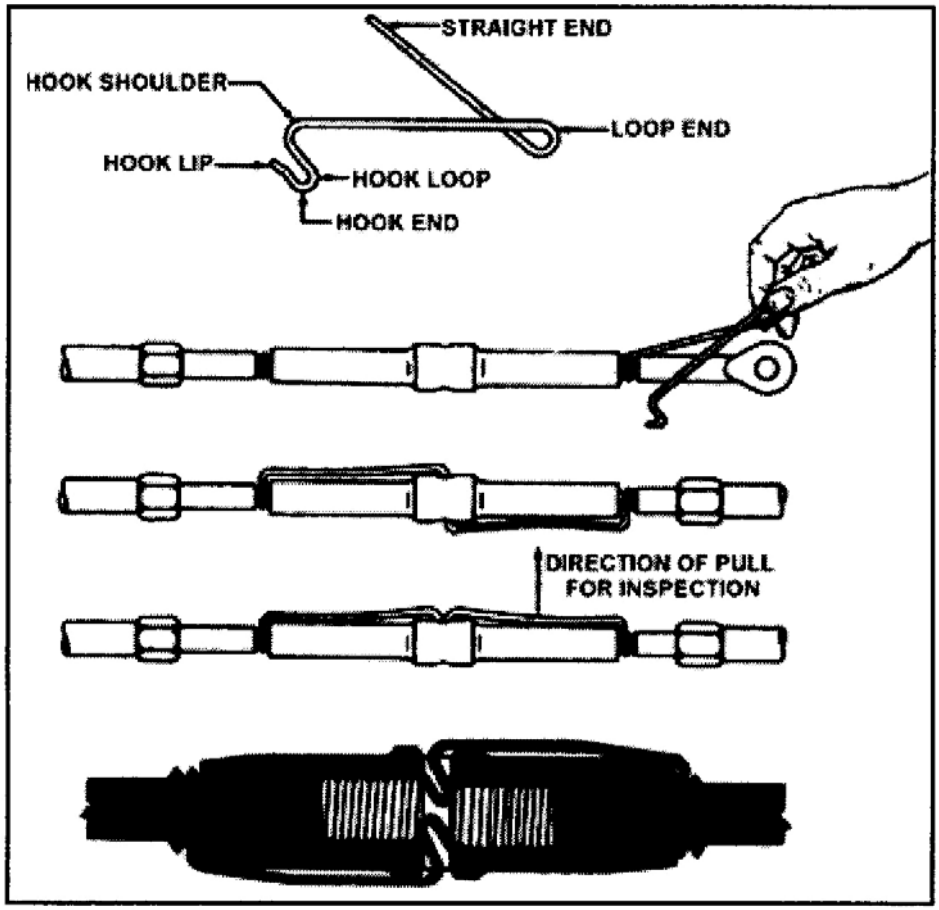

Figure 20 - 29 Assembling and Securing Clip-Locking Turnbuckles

Nominal Cable Diameter	Thread Unf-3	Locking Clip MS21256	Turnbuckle Body MS21251
1/16	No. 6-40	-1	-2S
1/8	1/4-28	-1	-4S
1/8	1/4-28	-2	-4L

Figure 20 - 30 Locking-Clip Application

20-21. Solvents, Sealants, and Adhesives

a. Many general purpose solvents, sealants, and adhesives are used in the aircraft.

b. Solvents are used for thinning and cleaning. Use of solvents used for painting are covered in Chapter 51. MEK and alcohol are used for cleaning of composite parts before bonding and their specific use is described in Chapter 51.

c. Sealants are used for protection, sealing, and thread locking. These items will be specifically called out in specific maintenance manual chapters where their use is required.

d. Adhesives are used in repair and installation. Special purpose adhesives are used for composite repair, composite bonding, upholstery installation, static wick installation, and window installation. Refer to specific maintenance manual chapters for instructions, precautions, and use of these adhesives.

e. Many oils and lubricants are also used.

f. Special purpose cleaners are also listed.

g. Figure 20 - 31 Engine and General Purpose Materials and Figure 20 - 32 Special Adhesives, Solvents, and Other Materials list these items.

Part Number.	Product Description Of Material	CAM Specification or Vendor
LL100	AVGAS, GASOLINE	
AEROSHELL 2F	BREAK-IN OIL, ENGINE, MIL-C-6529 TYPE II	AEROSHELL
AEROSHELL 15W50	OIL, ENGINE	AEROSHELL
Z24E024	OIL, RED HYDRAULIC MIL-H-5606	ESCO
	OIL, GENERAL PURPOSE, LUBRICATING MIL-L-7870	
AEROSHELL 22 MOBIL 28	GREASE, WHEEL BEARING MIL-G-81322	AEROSHELL MOBIL
	ANTISEIZE COMPOUND MIL-T-5544	LUBTORK
DC 7	SILICONE RELEASE COMPOUND	DOW CORNING
	SAE 10 OIL, GREASE SEAL PREPARATION	
AEROSHELL 6	PROPELLER GREASE	AEROSHELL
	K&N AIR FILTER CLEANER AND SOLVENT	K&N
	WINDEX, GLASS CLEANER, INSTRUMENTS AND MOVING MAP DISPLAYS	
	P210 PLASTIC CLEANER, WINDOWS	SUMNER LABS
	PERMATEX #2	SX519040 or LOCTITE
RTV133	SILICONE RUBBER, BLACK	E.V. ROBERTS
RTV5818	SILICONE RUBBER CAULK, CLEAR 400° F	E.V. ROBERTS GENERAL ELECTRIC
440014	SEALANT, HIGH PRESSURE	ND INDUSTRIES
PR 1428	FUEL TANK ACCESS PANEL AND FITTING SEALANT	PRC COURTALDS
AC236	FUEL TANK SEALANT	AC TECH
	PAINT, see Chapter 51	SX512701
	ZINC OXIDE PRIMER	SX513840
1522	FIRE RESISTANT PAINT, see Chapter 51	SX512701
	ZINC CHROMATE PRIMER	
	ACETONE	
	ALCOHOL	
	MEK	
325 SOLVENT UN1268	CLEANING SOLVENT	

Figure 20 - 31 Engine and General Purpose Materials

Part Number	Product Description Of Material	CAM Specification or Vendor
MGS418L MGS418H	ADHESIVE (STRUCTURAL), TWO PART EPOXY	SX513120
LP-89	CONDUCTIVE ADHESIVE, TWO PART EPOXY	SX518001 LUNAR PRODUCTS
EA9321	HYSOL ADHESIVE, TWO PART EPOXY	DEXTER E.V. ROBERTS
EA9339	HYSOL ADHESIVE, TWO PART EPOXY	DEXTER E.V. ROBERTS
1805	ADHESIVE, WINDOW ADHESIVE, TWO PART	SX535601 CASCO THAN
1806	CURING AGENT, WINDOW ADHESIVE, TWO PART	SX535601 CASCO THAN
1821	CURING AGENT, WINDOW ADHESIVE, TWO PART	SX535601 CASCO THAN
ADH105P	ADHESIVE, UPHOLSTERY	KEYSTON
243	LOCTITE THREADLOCKER MEDIUM (BLUE) PRIMER N or T	LOCTITE
262	LOCTITE THREADLOCKER (RED) PRIMER N or T	LOCTITE
542	LOCTITE THREADLOCKER (BROWN) PRIMER N	LOCTITE
609	LOCTITE RETAINING COMPOUND (GREEN) PRIMER T	LOCTITE
620	LOCTITE RETAINING COMPOUND (GREEN) PRIMER N	LOCTITE
660	LOCTITE RETAINING COMPOUND (SILVER) PRIMER N	LOCTITE
675	LOCTITE RETAINING COMPOUND (GREEN) PRIMER T	LOCTITE
N	LOCTITE PRIMER	LOCTITE
T	LOCTITE PRIMER	LOCTITE

Figure 20 - 32 Special Adhesives, Solvents, and Other Materials

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20-22. Lightning Protection and Static Dissipation Systems

a. The Cessna 350 (LC42-550FG) airframe is protected during lightning strikes by a combination of electrical grounding and metallic mesh over the composite structure. The lightning strike event is very powerful (hundreds of thousands of amps), of short duration (acting like very high frequency electronics), and of a powerful inductive and magnetic nature that can deform very strong conductive elements. Deviations from the approved design may not provide crew or airframe protection.

b. The metallic mesh conducts the currents of a lightning strike from the attachment point to the arc exit point. The requirements of this mesh are dependent on aircraft location.

c. Chapter 51 defines mesh requirements and mesh repair procedures.

d. **SAFE strips**

1. The aircraft interior is specially grounded to avoid large potential differences from one point to another during a lightning strike. This protection is accomplished by grounding bars, grounding straps, and Special Aluminum Foil Edging (SAFE) strips.
2. SAFE strips are thin self adhesive aluminum foil. They are strategically placed in the cockpit under the upholstery and seats on the composite structure (See Figure 20 - 33 and Figure 20 - 34). SAFE strips run across the fuselage from one grounding bar to the other.
3. SAFE strip Material Requirements:
   a) Adhesive Backed Aluminum Foil P/N 971023 from ALUMAT INC.
   b) MS20470AD4-4 Rivets
   c) AN960-4 Washers
   d) Squeegee for rubbing down SAFE strips
   e) Scissors for cutting SAFE strips
   f) Alcohol
   g) Oil-free Rags
4. SAFE strips should be replaced or repaired when they are severed across more than one half of the existing width.
5. SAFE strip repair foil must make contact with the existing foil. This is done by folding over the ends of the repair piece and securing the overlap area that is not secured with the self-adhesive backing of a new overlapping piece. This method is shown in Figure 20 - 35. Be sure to adhere to the overlap requirements of 1.0 in. foil to foil contact around the perimeter of the repair.
6. Clean all surfaces to which SAFE strip material must be applied. Use alcohol and clean, oil-free rags. Wipe surface, turning rag to use a new surface with each wipe. Apply the self-adhesive SAFE strip material by carefully aligning material and rubbing it down with a squeegee. Work slowly and carefully to prevent wrinkles and bubbles. Small wrinkles and bubbles less than 1.0 in. across are acceptable. Large bubbles should be popped with a pin or knife prick and squeegeed down. Cutting out the offending material and replacing it with a suitable patch can repair large wrinkles or creases in the SAFE strip. Be sure to follow the repair guidelines from the previous section.

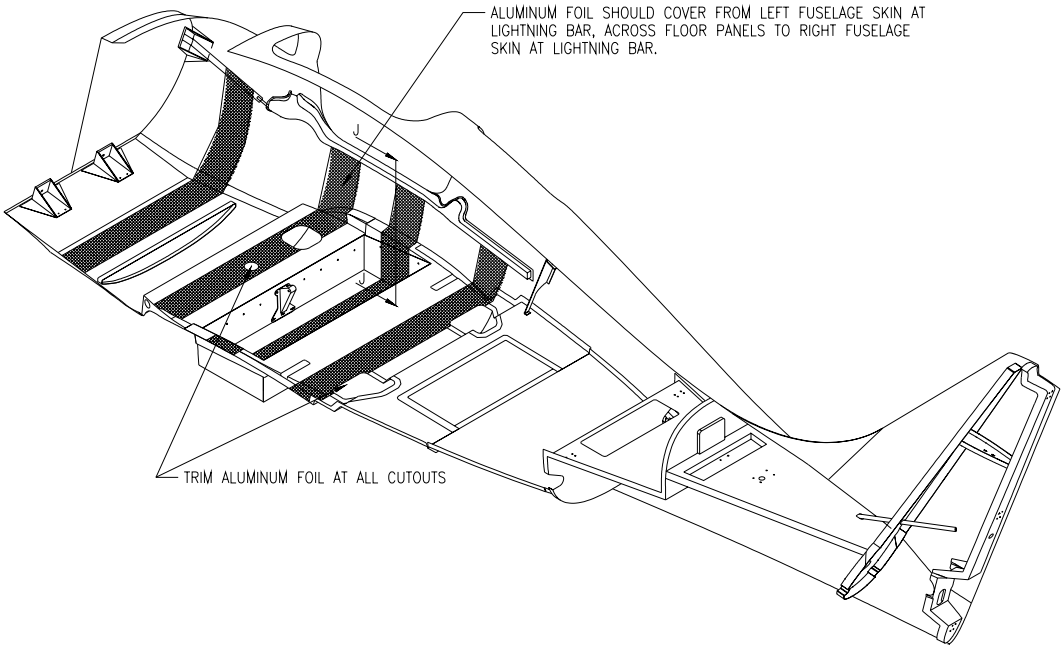

Figure 20 - 33 Placement of SAFE Strips (S/N 42001 to 42501)

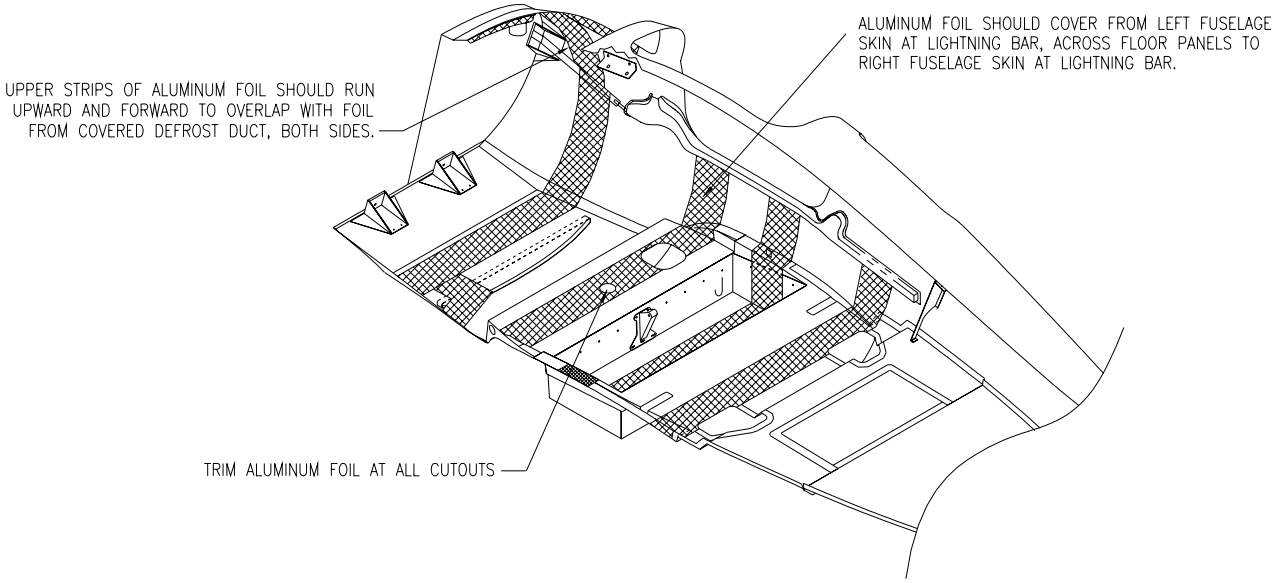

Figure 20 - 34 Placement of SAFE Strips (S/N 42502 and on)

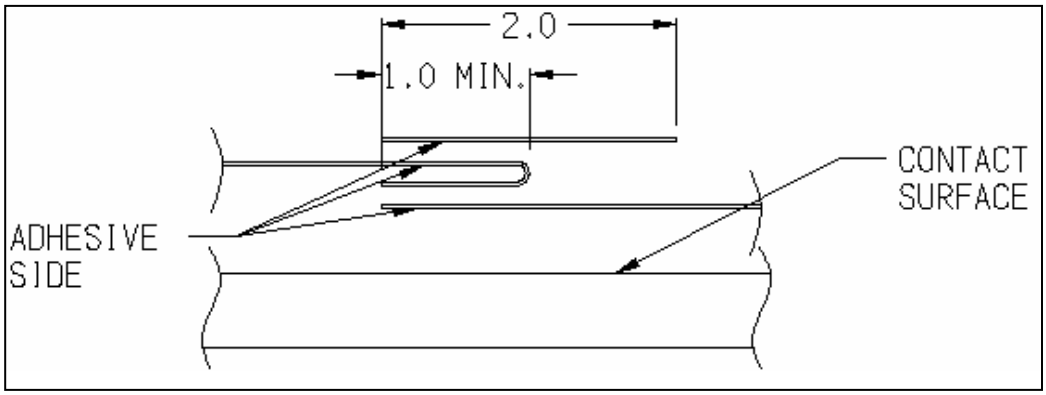

Figure 20 - 35 Folding SAFE Strip

7. SAFE strips are grounded to the grounding bars with rivets (See Figure 20 - 36). This connection should be inspected. If electrical contact is not confirmed by visual inspection of contact between the washer and SAFE strip, the connection should be removed for replacement. The use of a voltmeter to check the connection will not work. If any connection exists from the SAFE strip and the grounding bar, conductivity will be indicated. The conductivity check does not substantiate the current carrying ability of the connections.
8. Rivets are MS20470AD4-4 pop type and secure AN960-4 washers against the SAFE strips.

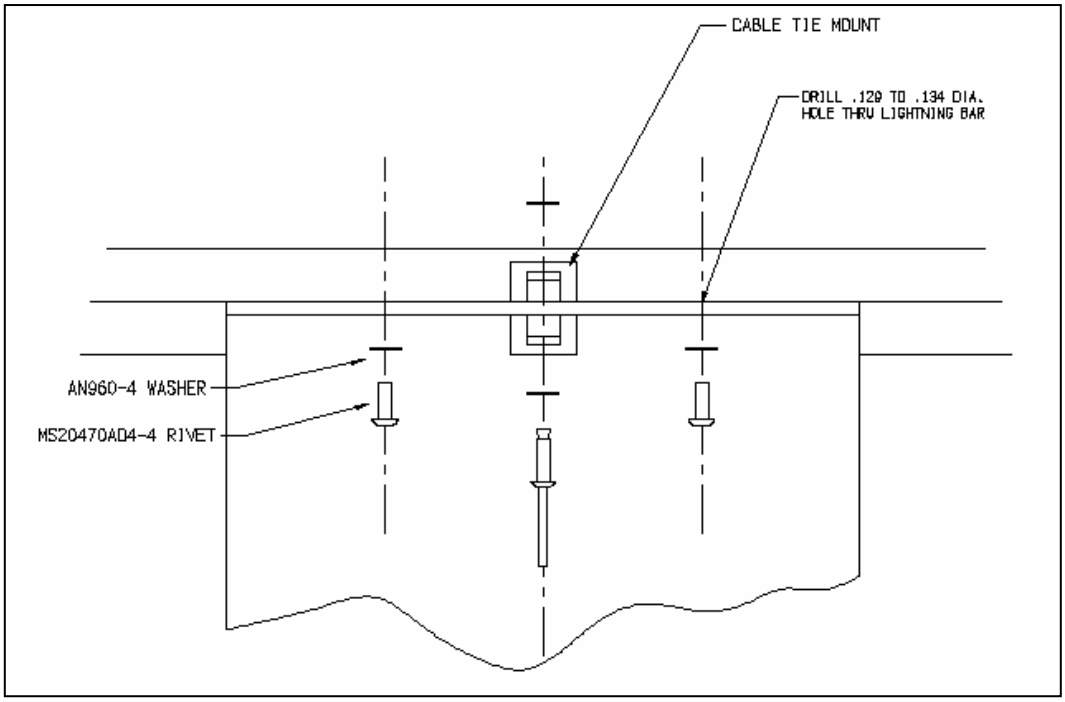

Figure 20 - 36 Grounding SAFE Strip to Grounding Bar

9. If complete replacement of a SAFE strip is required, the material should be folded over at the ends where contact with the grounding bar is required. This procedure is

shown in Figure 20 - 37. Installation of rivets and washers complete the electrical contact connection.

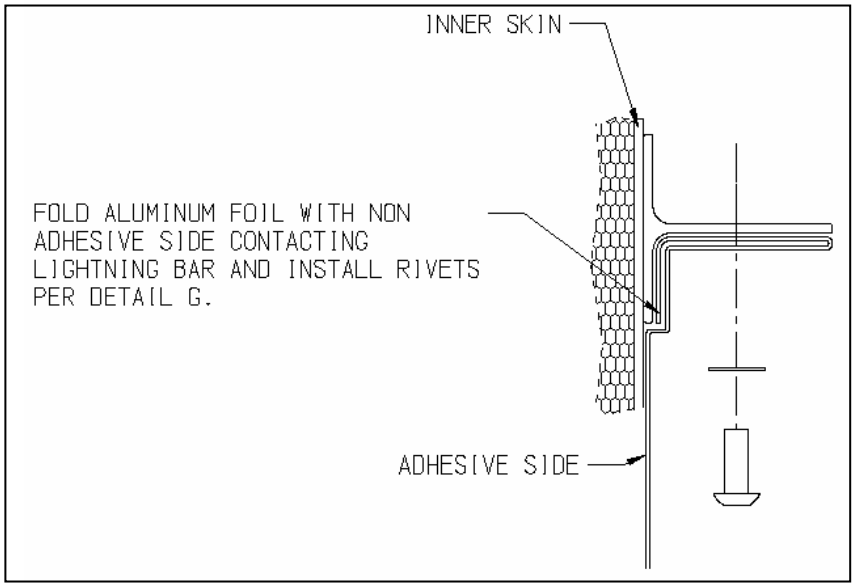

Figure 20 - 37 Complete Replacement of SAFE Strip

10. SAFE strips run into the footwell. Inspect this SAFE strip connection by visual confirmation of aluminum foil integrity in the contact area. This detail is shown in Figure 20 - 38. Note that the SAFE strip material wraps over the edges of the footwell, from the inside surface to the outside, to make contact with the SAFE strip material on the fuselage floor.

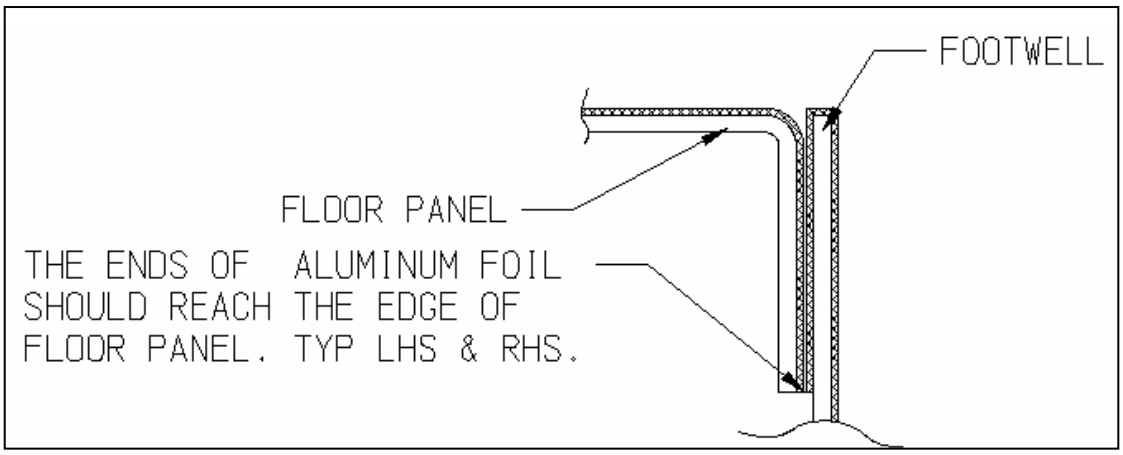

Figure 20 - 38 Installing SAFE Strip in Footwell

e. **Grounding Straps** – Grounding straps and bars are more conventional conductors. They provide a high current carrying capability between major airframe elements. Grounding straps can be found between the engine mount and the fuselage grounding bars, flight control surfaces and wing or stabilizers, gearbox and grounding bar, instrument panel and grounding bar, engine and radio rack, and various avionics components and the radio rack and instrument panel.

1. Braided Grounding Straps – These are located:
   a) Between the ailerons and wing (one each side)
   b) Between the rudder and vertical stabilizer
   c) Between the fuselage and elevator
   d) Between the gearbox and grounding bar (one each side)
   e) Between the instrument panel and grounding bar (one each side)
2. Solid Strap Conductors – These are located between the grounding bars and the upper engine mount bolts, one each side.

f. **General Aircraft Conductivity Checks**

1. The aircraft has nine static wicks. These wicks dissipate the static charge the aircraft develops as it flies through the air. The static charge the aircraft develops is strongly dependent on the atmospheric conditions at the time. Snow is most likely the atmospheric condition that will create the largest static charge. Improper static wick function may show up as a crackling sound from the radio speakers.
2. Check each static wicks (nine each) for low resistance to the nose gear or engine bonding strap. Each should measure less than 10 ohms.
3. The aircraft fuel filler caps and mounting rings should also be grounded to the rest of the airframe. This is especially important to prevent static discharge during refueling.
4. Check the fuel cap rings for low resistance to the nose gear or engine bonding strap. This value should be 0 to 500 ohms.

20-23. Inspection and Repair After Lightning Strike

a. In the event of a lightning strike, or suspected lightning strike, inspect the surface of the aircraft for evidence of damage such as bubbled or bumpy paint.

b. Remove paint and bodywork until mesh is exposed.

NOTE

If damage is noted to the composite material, evidenced by brown or darker fibers, contact Cessna for further instructions.

c. Using a 10x magnifying glass isolate the entire damaged area of mesh and a 1 inch square area of damaged mesh within that, evidenced by deformed mesh strands – flat or bulbous in appearance.

d. Count the number of damaged strands in that 1 inch square area and calculate what percentage that is of the entire affected area. If it is less than 2% of the total area no mesh repair is necessary. If greater than 2% of the total area a mesh repair is necessary.

e. For mesh repairs, overlap affected area 1 inch beyond damaged area with replacement mesh and apply using Jeffco epoxy.

f. Upon completion of repairs or aircraft exterior inspection, bodywork and paint affected areas in accordance with Chapter 51.

g. Inspect all hardware associated with damaged areas for magnetism. If no magnetism is noted, re-use hardware.

h. If hardware is found to be magnetized, inspection of respective hardware locations where hardware contacts mesh is required. Wing attachment hardware contacting mesh must have associated bolt holes inspected for damaged mesh. If mesh around the bolt holes is damaged the affected holes and hardware must be increased one size to restore contact of mesh to hardware for proper lightning protection. Enlarge holes using a 5/16 tapered drill ream.

i. Actuate flight controls to ensure proper operation. If defects are noted, inspect for evidence of arcing on control surface attachments and associated hardware and replace as needed.

j. Remove the circuit breaker panel in accordance with Chapter 25. Remove and replace the avionics bus surge suppressor attached to the front side of the aft circuit breaker panel half.

k. Reinstall the circuit breaker panel and perform one of the following electrical/avionics checkout procedures for all equipment. One procedure is specifically of Basic or Avidyne avionics, the other is for the Garmin G1000 system only. If any equipment is not operating properly, contact Cessna for further instructions for repair or replacement of the equipment.

Checkout Description for Basic or Avidyne Avionics
1. A. Connect 'Buzz Box' to aircraft MAG'S connecting to: I. Right MAG sensor wire to center post on right MAG II. Left Mag sensor wire to center post on left MAG III. Ground sensor wire to engine ground **Note:** It may require the prop to be turned clockwise until an audible click can be heard for the MAG's to work. B. Verify ignition switch positions: I. Left II. Right III. Both IV. Off
2. **WARNING:** Clear the area around the prop of personnel and equipment. To avoid the engine from starting, ensure that the spark plugs are not installed or have been disconnected. Energize the starter with only the left bus energized and verify operation and proper direction of rotation of the engine. Energize the starter with only the right bus energized and verify the operation of the starter motor. Prop should rotate clock-wise as viewed from the pilots seat. The Starter Engaged light on the annunciator panel should be lit when the Starter is energized.
3. Verify operation of the battery switches and essential bus feed. Turn on only the left bus and verify items on the left bus are energized. Also check that the essential bus is energized. Turn on only the right bus and verify items on the right bus are energized. Check again that the essential bus is energized. (Note: No ground power is applied for this test.)

Checkout Description for Basic or Avidyne Avionics
4. Connect a 14 VDC (S/N 42001 to 42500) or 28 VDC (S/N 42501 and on) power supply to the ground power plug. (Maintain voltage between 13 VDC and 14 VDC (S/N 42001 to 42500) or between 26 VDC and 28 VDC (S/N 42501 and on) for this test.) Turn on the Crosstie switch and verify that the right bus is energized when the Crosstie switch was turned to 'ON' and that both the right and left buses are energized. Verify that the green lights on the left and right voltage regulators are illuminated when the "ALT" switches are turned on. (Voltage regulators are located on the inside of the firewall, right hand side.) Verify the ammeter shows a charge on the corresponding battery when that battery switch is turned on and the ammeter is in the "BATT" mode.
5. With the Right 'ALT' switch on and no external power applied, check for approximately 10-11 volts (S/N 42001 to 42500) or 24-26 volts (S/N 42501 and on) on the F-1 terminal of the Right alternator located at the front of the engine. Terminal F-2 is grounded. With the Left 'ALT' switch on check for approximately 10-11 volts (S/N 42003 to 42500) or 24-26 volts (S/N 42501 and on) on the field terminal of the Left alternator located at the back of the engine.
6. With no ground power supplied, energize the avionics master switch and check for voltage on the avionics bus with only the left bus energized and also with only the right bus energized.
7. **CAUTION:** Use care when touching hot pitot mast (located under right wing). Ensure **no** object (e.g. pitot tube cover, adapter, etc.) is attached to the pitot tube. **Do not** operate pitot heat for more than two minutes. Turn on the pitot heat and check for heating at the pitot tube. Either: i) At the pitot line moisture drain point (center wing), disconnect the drain fitting hose that goes forward to instrument panel. Or: ii) Attach the pitot tube with a heat resistant adapter connected to flexible hose. Using Pitot Static test set, raise the pitot system to indicate > 80 kts on the Airspeed indicator. Verify that the Ammeter load current increases. Turn off the pitot heat and reconnect the hose fitting or remove the pitot tube adapter.
8. Turn on the position lights switch and check all 4 bulbs for illumination and proper color orientation (green right side, red left side).
9. Turn on strobe lights and verify 3-4 flashes per event. Ensure that the left and right wing flashes alternate between the left and right side.
10. Turn on landing light and verify the inboard light illuminates (the one without a diffuser).
11. Turn on the taxi light and verify the outboard light with the diffuser illuminates.

Checkout Description for Basic or Avidyne Avionics
12. Door Operational Checkouts A. Close and lock left, right and baggage doors. B. Door open light on annunciator should be extinguished. C. Open any door and check the flip lights and the step lights stays lit for approximately 10 minutes. D. Test the right and baggage doors as described in 'C' above. E. With all doors closed and locked turn on Door Seal Pump Switch, verify pump operation- door seals inflate and pump turns off when seals are full. F. Door seal pump should not cycle (on and off) for at least 30 seconds. G. Open left door slowly, dump value should relieve all pressure and door opens. H. Close door and repeat F & G above for the right side door. I. Turn door seal system off, slowly open either door to verify dump valve operation.
13. Turn on the Low Boost switch and verify low speed operation of the boost pump.
14. Momentarily move the back-up pump switch to the arm position. Verify that the blue arm light illuminates and the boost pump operates in high speed. The fuel pump annunciator should also light.
15. Press the primer button and verify high-speed operation of the boost pump.
16. Verify operation of the engine gages that can be checked without the engine running: a) fuel level gage, b) man press, c) oil temp, and d) dual amp gage (turn on a load, should show discharge)
17. Check ELT remote switch function (leave the ELT Transmitter in the armed mode). Tune the VHF COM to 121.5 MHz, and listen through the cabin speaker or the headsets. Turn the ELT switch, located on the right hand knee bolster, to the 'ON' position. **Note**: Per FCC regulations the ELT system should only be checked during the first five (5) minutes of the hour.
18. Test the static and pitot systems for leak rate and for proper operation of the system. **CAUTION: DO NOT EXCEED A RATE OF CLIMB >2000 FEET PER MINUTE.** A. Static system leak rate = _____fpm @ 1000' above field elevation. (Maximum leak rate is 100 fpm @ 1000' above field elevation for the static system only.) Raise static system 100' above field elevation and hold open the alternate static valve, located on the left side knee bolster (turn to **ALT**), the system should immediately drop to field elevation. B. With switch still in **ALT** position, raise Pitot System to 200 kts. Maximum leak rate is 50 kts per minute. Pitot System leak rate = _____kpm. After testing Pitot System return switch to the **NORM** position.
19. Ensure the turn coordinator and attitude gyro flags are not displayed and listen for the rotor of the gyros to spin-up.

Checkout Description for Basic or Avidyne Avionics
20. Skip this step if Avidyne Entegra avionics are installed on the aircraft. Verify OAT both F & C, and voltage readings using top red button. Using the select and control buttons, verify clock operations and functions. (UT, LT, FT, & ET)
21. Operate the flap switch and verify that the LED in the knob flashes while the flaps are in transition. Check all three positions. Verify the flashing light goes out once the flaps reach the position indicated on the handle.
22. Verify operation of trim servos (two axis) for travel, direction of travel, and run them from stop to stop. Ensure trim panel indications agree with external position of the tabs for two axis and that the display properly indicates stop to stop. Verify the operation of the on/off switch. Verify that the pilot's Trim Switch on the control stick (left side) overrides the co-pilot Trim Switch on the control stick (right side).
23. Skip this step if Avidyne Entegra avionics are installed on the aircraft. Compass System Check Out for KCS55A System and KI525A HSI A. Apply aircraft Battery and Avionics power B. HSI red HDG flag pulls after 2 minutes or less (top left) C. Go to Free mode on the KA51B Slaving Controller D. Move CW/CCW switch to CW, compass card moves Clock Wise and HDG flag appear. E. Move Slave/Free switch to Slave mode, compass card rotates back to aircraft heading, + - meter shows deflection until the HDG flag pulls. F. Go back to Free mode on Slaving Controller and drive the compass card to the CCW position, compass card drives Counter Clock Wise and HDG flag appears during movement. G. Move the Slave/Free switch to Slave mode, compass card rotates back to aircraft heading, + - meter shows deflection until the HDG flag pulls.
24. Perform Initial Set-Up Procedures for the Cessna 350 Avionics, as equipped.
25. Turn on the Avionics Master Switch and verify the operation of: A. Pilot's and co-pilot's Audio, VOR, ILS and MKR Navigation systems using the Nav 402AP Test set. GPS #1 _____ GPS #2 _____ Audio System _____ VOR/ILS 1 _____ VOR/ILS 2 _____ Marker Beacon _____ B. Perform the Transponder system check-out using the ATC-600A test set or ATC 601 if Mode S equipped. Record Transponder **S/N**_____ and **Model No**. _____ C. Check GPS #1 & GPS #2 for lock and display of position.

Checkout Description for Basic or Avidyne Avionics
26. Check wire runs and attachments of all wires and coax cables for abrasion potential, support, and wire tying. a. Along sides of fuselage _____ b. Behind the stick grips _____ c. Behind the instrument panel_____ d. All flight controls and fluid lines_____ e. Behind the radio stack_____ f. Ensure wiring bundles are free and clear of rudder cables on the right side and free and clear of the circuit breaker panel and rudder limiter assembly on the left side._____
27. Verify security and quality of the switches on the stick grip.
28.
29. Check the dual ammeter for proper operation. The ammeter should power up in the "BATT" mode, pressing the button in the lower right should change the mode to "ALT". It should show a corresponding discharge when a load is applied to the appropriate bus in the "BATT" mode.
30. Check the fuel cap rings for low resistance to the nose gear or engine bonding strap, (0 to 500 ohms). **CAUTION: CHECKING GROUNDS WITH AIRCRAFT POWER APPLIED WILL DISTORT RESISTANCE READING.**
31. Remove all power from the aircraft. Check each static wick (9 ea.), for low resistance to the nose gear or engine bonding strap, (less than 10 ohms). **CAUTION: CHECKING GROUNDS WITH AIRCRAFT POWER APPLIED WILL DISTORT RESISTANCE READING.**
32. Turn on the cabin fan and check for airflow through the selected vents, defroster and floor.
33. Verify the bus labels on the circuit breaker panel are correct. The top 2 rows are left bus, row 3 is right bus, row 4 is essential bus and the bottom row is avionics.
34. Activate the hot-cold control on the heater panel and verify the firewall shut-off servo works.
35. Verify the cooling hose is securely fastened and in place behind the instrument panel and the radio rack.
36. Verify operation of the avionics fan.
37. To simulate the engine running disconnect oil pressure switch connecter (aft left side of engine) and short pins A & B together on connector.

Checkout Description for Basic or Avidyne Avionics
38. Check the annunciators and Aural warning system for operation without ground power connected. A. The Left Alt switch turned on, no light or aural warning heard over speakers or headset. a. Turn Left Alt switch to off, L ALT OFF light is illuminated on annunciator panel. i. After 2 seconds "alternator off" is heard over headsets and speaker. b. Press ACK button and aural warning is canceled. B. The Right Alt switch turned on, no light or aural warning heard over speakers or headset. a. Turn Right Alt switch to off, R ALT OFF light is illuminated on annunciator panel. i. After 2 seconds "alternator off" is heard over headsets and speaker. b. Press ACK button and aural warning is canceled C. On center console select Left fuel tank; a. No light on annunciator panel or aural warning is heard over speakers or headset. Fuel Level gage illuminates green LED for tank selected. b. Turn Fuel selector to off, FUEL VALVE light is illuminated on annunciator panel. i. After two seconds "fuel valve" is heard over headsets and speaker. ii. No LED is lit on Fuel Level gage. c. Press ACK button and aural warning is canceled. d. Repeat step a., b. & c. for the Right fuel tank selection. D. On aircraft clock press "Select " button to "ET" mode. a. Press "Control" button to count down from 0:01:00 i. When timer reaches zero "Timer at Zero" is heard over headsets and speaker only twice and cancels without pressing the ACK button. E. Check that L LOW FUEL, R LOW FUEL, and DOOR OPEN lights are illuminated on annunciator panel. F. Check that OIL light on annunciator panel is off. G. Remove jumper on Oil Pressure switch and reconnect connector. a. OIL light on annunciator panel illuminates.
39. With the rudder pedals aligned evenly, activate the rudder limiter by pressing Trim Panel Test button and check that the RDR LMTR annunciator light illuminates and an audible click is heard.

Checkout Description for Basic or Avidyne Avionics
40. Speed Brake Operational Checks: **WARNING: CLEAR THE LEFT AND RIGHT WINGS OF PERSONNEL AND EQUIPMENT.** A. Ensure rudder pedals are aligned evenly and external power is applied >12.4VDC (S/N 42001 to 42500) or 26 VDC (S/N 42501 and on) B. Deploy S/B by pressing the **SPDBRK** switch located between the Throttle and Prop levers to the up position. C. S/B deploy, SPDBRK light on annunciator panel illuminates. D. Press the test button on the trim panel, S/B retract, release test button. E. Reposition the **SPDBRK** switch to the down position. F. Redeploy the Speed Brakes and move the stall vain lift detector to the stall position full up, the S/B retract. G. Reposition the **SPDBRK** switch to the down position.
41. Check the test switch on the trim panel to operate the following: A. Annunciator panel, all LED's light. B. All trim LED's light. C. Rudder limiter engages. D. All GPS remote annunciators light, if equipped. E. Flap switch LED blinks. F. Both Fuel level gauge LED's illuminate. G. Nav/Com Bypass Switch illuminates.
42. Check the speaker for operation.
43. Check the power point for voltage and verify polarity.
44. Turn on the position lights and verify that the following devices Dim: fuel level gauge LED's, Trim panel, Flap switch LED. Turn off the position lights and verify this action turns off the interior light dimming system.
45. Verify all four (4) channels of the dimmer, one (1) channel at a time: A. **UPPER PNL** (panel) lights; Verify Engine and Flight instruments have (2) bulbs illuminated per instrument. Clock light and compass light are illuminated and no other lights are on. B. **LOWER PNL** (panel) lights; Verify lower panel lights and circuit breaker panel lights are illuminated and no other lights are on. C. **FLOOD** lights; Verify glare shield lights are illuminated and no other lights are on. D. **SPOT** lights; Verify overhead spotlights are illuminated and on other lights are on. With **SPOT** lights 'ON', verify that each lamp turns 'OFF' when using the switch located on the light housing. **NOTE:** Interior lights only work with position lights rocker switch in the 'ON' position.
46. Check for painted or plastic coated screws in the instruments panels, and clean instrument dials.
47. Visually check integrity of all fuses (4) on the side of the power grid.
48. Operate the flip lights, front and rear, with the power off and verify correct operation.

Checkout Description for Basic or Avidyne Avionics
49. Pull all circuit breakers and ensure the labeling agrees with the circuit it supplies by pushing in the circuit breaker. Ensure positive latching of the circuit breaker.
50. Activate the Nav/Com Bypass Switch with the aircraft power off and ground power disconnected. The switch should illuminate and the #1 GPS, Com, and Nav should turn on. The pilots headset should receive and transmit on the #1 Com. Turn off the Nav/Com Bypass Switch and the GPS, Com and Nav should turn off.

Checkout Description for Garmin G1000 System
1. A. Connect 'Buzz Box' to aircraft MAG's connecting to: IV. Right MAG sensor wire to center post on right MAG V. Left MAG sensor wire to center post on left MAG VI. Ground sensor wire to engine ground **Note:** It may require the propeller to be turned clockwise until an audible click can be heard for the MAG's to work correctly. B. Verify ignition switch positions: V. Left VI. Right VII. Both VIII. Off Remove the 'Buzz Box'.
2. **WARNING:** Clear the area around the propeller of personnel and equipment. To avoid the engine from starting, ensure that the spark plugs are not installed or have been disconnected. Perform the following tests with the crosstie open: Energize the starter with only the left bus energized and verify operation and proper direction of rotation of the engine. The propeller should rotate clockwise as viewed from the pilot's seat. The Starter Engaged annunciation on the PFD should illuminate when the Starter relays are energized. Energize the starter with only the right bus energized and verify the starter does not engage. Remove the keys from the ignition switch and stow in the proper location.
3. Verify operation of the battery switches and essential bus feed. Turn on only the left bus and verify items on the left bus are energized. Also check that the essential bus is energized. Turn on only the right bus and verify items on the right bus are energized. Check again that the essential bus is energized. (Note: No ground power is applied for this test.)
4. Connect a 28 VDC power supply to the ground power plug. (Maintain voltage between 26 VDC and 28 VDC for this test.) Turn on the crosstie switch and verify that the left bus is energized when the crosstie switch was turned to 'ON' and that both the right and left buses are energized. Verify that the green lights on the left and right voltage regulators are illuminated when the "ALT" switches are turned on. (Voltage regulators are located on the inside of the firewall, right hand side.) Verify the ammeter shows a charge on the corresponding battery when that battery switch is turned on.
5. With the right 'ALT' switch on, check for approximately 24-26 volts on the F-1 terminal of the right alternator. Terminal F-2 is grounded. With the left 'ALT' switch on check for approximately 24-26 volts on the field terminal of the left alternator. Measure the resistance from the left alternator case, to the engine block. The total resistance should be less than 1.0 Ω.
6. With ground power turned off and using batteries, energize the avionics master switch and check for voltage on the avionics bus with only the left bus energized and also with only the right bus energized, crosstie open.

Checkout Description for Garmin G1000 System
7. Apply ground power and turn on the position lights switch. Check all 4 bulbs for illumination and proper color orientation (green right side, red left side).
8. Turn on the strobe lights and verify 3-4 flashes per event. Ensure that the wing flashes alternate between the left and right sides.
9. Turn on the landing light and verify the inboard light illuminates (the one without a diffuser).
10. Turn on the taxi light and verify the outboard light (with the diffuser) illuminates.
11. Turn on the Vapor Suppress switch and verify low speed operation of the boost pump.
12. Momentarily move the back-up pump switch to the arm position. Verify that the blue arm light illuminates and the boost pump operates in high speed.
13. Press the primer button and verify high-speed operation of the boost pump.
14. Verify operation of the engine gages that can be checked without the engine running: a) fuel level gage, b) man press, c) oil temp, d) dual amp gage, e) bus voltage gage, and f) engine temperatures. (Cycle the engine instrument circuit breaker to test needle movement).
15. Door Operational Checkouts; A. Close and lock left, right and baggage doors. B. Door Open annunciation on the PFD should be extinguished. C. Open any door and check the flip lights and the step lights stays lit for approximately 10 minutes. D. Test the right and baggage doors as described in 'C' above. E. With all doors closed and locked, turn on Door Seal Pump Switch, verify pump operation- door seals inflate and pump turns off when seals are full. F. Door seal pump should not cycle (on and off) for at least 30 seconds. G. Open left door slowly, dump value should relieve all pressure and door opens. H. Close door and repeat F & G above for the right side door. I. Turn door seal system off, slowly open either door to verify dump valve operation.

Checkout Description for Garmin G1000 System
16. Test the static and pitot systems for leak rate and for proper operation. **CAUTION: DO NOT EXCEED A RATE OF CLIMB >2000 FEET PER MINUTE.** Static system leak rate = _____fpm @ 1000' above field elevation. (Maximum leak rate allowed is 100 fpm @ 1000' above field elevation for the static system only.) Raise static system 100' above field elevation and hold open the alternate static valve, located on the left side knee bolster (turn to **ALT**), the system should immediately drop to field elevation. With switch still in **ALT** position, raise pitot system to 200 kts. Close the valves on the pitot static test pump, then time and observe the pitot system leak rate. Verify that the airspeed tape on the PFD and the mechanical airspeed operate correctly. Maximum leak rate allowed is 25 kts per minute. Pitot System leak rate = _____kpm. Note: Conduct the test for the next task concurrently with this test item.
17. Check the resistors on the autopilot 485 lines and verify their resistance per the manufacturer specifications.
18. Replace the TVS diodes on the PFD, GIA, AHRS and standby ADI.
19. Replace the zener diodes on the oxygen system.
20. Verify that the PFD is properly configured. While testing the pitot system, run the PFD and mechanical airspeed indicator high enough to verify that the following airspeed limitation markings are correct: (Below 12,000 ft PA) A. White Range, 59 – 119 KIAS B. Green Range, 71 – 179 KIAS C. Yellow Range, 235 KIAS D. Red Range, above 235 KIAS
21. Pitot Heat Test. **CAUTION:** Use care when touching the hot pitot mast. Do not operate pitot heat for more than one minute. A. Turn on the pitot heat switch on the instrument panel. Note that the current draw on the amp meter increases. Carefully feel with the back of your hand for actual heating at the pitot tube. B. Turn off the pitot heat. **Only leave the pitot heat on long enough to verify the operation of the heating element.**
22. Ensure the flag of the standby attitude gyro located on the instrument panel, are not displayed. Listen for the rotor of the gyros to spin-up.
23. Operate the flap switch and verify that the LED in the knob flashes while the flaps are in transition. Check all three positions. Verify the flashing light goes out once the flaps reach the position indicated on the handle.

Checkout Description for Garmin G1000 System

24. Trim Indicators Adjustment

A. Verify operation of trim servos (two axis) for travel, direction of travel, and run them from stop to stop. Verify the operation of the on/off switch.

B. Ensure the trim indications agree with the external position of the tabs for the aileron and elevator trim and that the display properly indicates stop to stop.

C. If trim panel adjustment is required contact Cessna for instructions. Repeat step a if adjustments are made.

25. Engage the Autopilot with the audio panel on at the same time. Disconnect the autopilot, while listening for the audio level of the AP DISC through the aircraft speaker.

26. Perform Initial Set-Up Procedures for the Cessna 350 Avionics, as equipped

27. Avionics Tests

Turn on the Avionics Master Switch and verify the operation of:

Comm 1/2:

☐☐ TX/RX 123.225
☐☐ T/R QUALITY
☐☐ SIDE TONE
☐☐ MAN SQUELCH
☐☐ PILOT'S PTT
☐☐ COPILOT'S PTT

GPS 1/2:

☐☐ Verify that GPS 1 and GPS 2 are receiving a signal
☐☐ Sat Lock
☐☐ Setup per procedure
☐☐ TIS traffic data – TIS will not be available unless a terminal radar site is nearby providing data. Most of the time, this will not be the case. If TCAD is installed TCAD must be failed before TIS will be displayed.
☐☐ DX to PFD, MFD
☐☐ DX from PFD
☐☐ S/W and data cards revisions match
☐☐ Verify GPS positions are valid on MFD

PFD:

☐ Proper Setup for Cessna 350.
☐ RX valid GPS position data from GPS 1,2
☐ Knob operation.
☐ Air Data operation
☐ Autopilot Annunciations – Verify no red AFCS or PFT on powerup.
☐ Valid AHRS
☐ Cross talk with MFD – Verify that the message and alert box has no messages that state a backup path is being utilized.

Checkout Description for Garmin G1000 System

Avionics Tests Continued

MFD:

- ☐ Proper setup for aircraft type
- ☐ RX valid GPS position data from GPS 1,2
- ☐ Cross talk with PFD – Verify that the message and alert box has no messages that state a backup path is being utilized.
- ☐ Receives Storm scope or WX data as installed.
- ☐ Engine instrumentation data present and valid.
- ☐ Traffic – For TCAD verify PFD message box does not display TCAD system failure message (approx. one minute for TCAD to go online and failure messages to clear). For TIS verify Mode S transponder is functioning by checking the transponder annunciation on the PFD. If there is no red "X" and indicates normally, TIS will be functional.

Audio System:

- ☐ Pilot's ICS
- ☐ Copilot's ICS
- ☐ Pass 1,2 ICS
- ☐ ISO/CREW functions
- ☐ Pilot's/Copilot's side tone quality and level
- ☐ Emer. TX on Comm 1 with audio panel off.
- ☐ Nav 1, 2 Audio
- ☐ Bose Headsets **REPEAT ALL AUDIO SYSTEM CHECKS**

Conduct the following tests using the NAV-401AP test set.

Marker beacon:

- ☐ Outer Marker – 400hz tone, Blue Light
- ☐ Middle Marker – 1300 Hz tone, Amber light
- ☐ Inner Marker – 3Khz tone, White Light
- ☐ Remote Display
- ☐ Hi/Lo sensitivity tests, Hi/ Lo Split

MFD:

- ☐ Proper setup for aircraft type
- ☐ RX valid GPS position data from GPS 1,2
- ☐ Cross talk with PFD – Verify that the message and alert box has no messages that state a backup path is being utilized
- ☐ Receives Storm scope or WX data as installed.
- ☐ Engine instrumentation data present and valid.

Checkout Description for Garmin G1000 System

Avionics Tests Continued

Navigation systems 1,2 :

NAV:

- ☐☐ Receive 112.80
- ☐☐ Flags Out of view
- ☐☐ CDI (Left / Center / Right)
- ☐☐ To/From operation
- ☐☐ 30 degree accuracy checks

ILS:

- ☐☐ Receive 108.1
- ☐☐ Localizer (Left / Center / Right)
- ☐☐ Localizer flag pulled
- ☐☐ Glide slope bars (Up / Center / Down)
- ☐☐ Glide slope valid
- ☐☐ Autopilot coupling and tracking checks in Nav Mode, testing above functions
- ☐☐ Nav Audio
- ☐☐ ILS sensitivity

Autopilot:

- ☐ Master switch operation
- ☐ HDG bug operation
- ☐ NAV (Tracks D-BAR)
- ☐ CRS (Tracks Course Datum Pointer, No signal applied)
- ☐ APR mode engaged on ILS freq.
- ☐ Localizer Left Right (Stick Tracks)
- ☐ Glide Slope soft capture
- ☐ G/S bars up/down, stick tracks to glide slope bars
- ☐ ALT hold mode engages and holds
- ☐ ALT PRESELECT engages and tracks from PFD
- ☐ VS engages and tracks from PFD controls
- ☐ AP Disconnect (Both control sticks)
- ☐ Disconnect audio
- ☐ Remote Annunciator on PFD

Flight Director:

- ☐ Command Bars engage, and track in Autopilot pitch mode.
- ☐ AP FD mode, Command bars tracks with autopilot
- ☐ FD mode; Command Bars track, Servo clutches disengage.

Auto trim:

- ☐ Hold slight aft and forward pressure and verify autotrim in the opposite direction on the MFD.

Checkout Description for Garmin G1000 System
28. Physically verify the security of the BNC connections on the NAV signal splitter, and the back of the radios.
29. Check the speaker for operation.
30. Check ELT remote switch function (leave the ELT Transmitter in the armed mode). Tune the VHF COM to 121.5 MHz, and listen through the cabin speaker or the headsets. Turn the ELT switch, located on the right hand knee bolster, to the 'ON' position **Note**: Per FCC regulations the ELT system should only be checked during the first five (5) minutes of the hour.
31. Transponder Test Perform the transponder test and checkout using the ATC 600A or ATC 601-2 test set. The ATC-600A Transponder/DME test set is to be used for testing Garmin AT model SL 70 or Garmin model GTX 327 Mode A/C transponders. The ATC 600A **is not to be used** for testing of the Garmin model GTX 33 mode S transponder. The ATC-601-2 Mode S Transponder test set is required for testing Garmin model GTX 33 Mode S transponder, and may also be used for testing Garmin model GTX 327 Mode C transponder. Note: A DI or Q inspector must witness the transponder test. Record the model and serial number of the transponder below. **Make/Model_____/_____** **Serial Number_____**
32. Turn aircraft power OFF, check the fuel cap rings for low resistance to the nose gear or engine bonding strap, (0 to 500 ohms). **CAUTION: CHECKING GROUNDS WITH AIRCRAFT POWER, or GROUND POWER APPLIED WILL DISTORT RESISTANCE READING.**
33. Remove all power from the aircraft. Check each static wick or mounting pad (9 ea.), for low resistance to the nose gear or engine bonding strap, (less than 10 ohms). **CAUTION: CHECKING GROUNDS WITH AIRCRAFT POWER APPLIED WILL DISTORT RESISTANCE READING.**
34. Turn on the cabin fan and check for airflow through the selected vents, defroster and floor.
35. Verify the bus labels on the circuit breaker panel are correct. The top 2 rows are essential bus, row 3 is left bus, row 4 is right bus and the bottom row is avionics.
36. Activate the hot-cold control on the heater panel and verify the firewall shut-off servo works.
37. Verify operation of the avionics fan.

Checkout Description for Garmin G1000 System
38. Check the annunciator and aural warning system for operation without ground power connected. Start the engine. A. The left Alt switch turned on, no light or aural warning heard over speakers or headset. a. Turn left alt switch to off, L ALT OUT annunciation on the PFD. i. After 2 seconds "left alternator out" is heard over headsets and speaker. b. Press Alert button and aural warning is canceled. i. The left alt switch turned on, no light or aural warning heard over speakers or headset. c. Turn right alt switch to off, R ALT OUT annunciation on the PFD. i. After 2 seconds "right alternator out" is heard over headsets and speaker. d. Press Alert button and aural warning is canceled e. Turn the right alt switch to on. f. Shut down the engine. B. On center console select left fuel tank; a. No annunciations or aural warning is heard over speakers or headset. Fuel level gage illuminates blue indication for tank selected. b. Turn fuel selector to off, FUEL VALVE annunciation on the PFD. i. After two seconds "fuel valve" is heard over headsets and speaker. ii. No indication next to the fuel level gage. c. Press Alert button and aural warning is canceled. d. Repeat step a., b. & c. for the right fuel tank selection. C. Check that DOOR OPEN annunciates on the PFD when one of the three doors are open.
39. Speed Brake Operational Checks: **WARNING: CLEAR THE LEFT AND RIGHT WINGS OF PERSONNEL AND EQUIPMENT.** A. Apply external power > 26 VDC. B. Deploy S/B by pressing the **SPDBRK** switch located between the Throttle and Prop levers to the up position. C. S/B deploy, SPEED BRAKES annunciates on the PFD. D. Reposition the **SPDBRK** switch to the down position. E. Redeploy the Speed Brakes and move the stall vane lift detector to the stall position full up, the S/B retract. F. Reposition the **SPDBRK** switch to the down position.
40. Check the test switch on the overhead. The test switch will illuminate the pushbutton indication on the left side of the cockpit, illuminate the flap handle, and activate the rudder limiter. The rudder limiter activation will cause a chime to play and an amber caution to be displayed on the PFD.

Checkout Description for Garmin G1000 System
41. Turn on the position lights and verify that the following devices dim: flap panel switches, 5 pack switches, and flap switch LED.
42. Operate the flip lights, front and rear, with the power off and verify correct operation.
43. Verify all four (4) channels of the dimmer, one (1) channel at a time: A. **UPPER PNL** (panel) lights; MFD, PFD, Audio Panel, Remote Keypad, standby instrument lights and compass light are illuminated and no other lights are on. B. **LOWER PNL** (panel) lights; Verify Master switch panel, Heater control panel, Trim panel, and the circuit breaker panel lights are illuminated and no other lights are on. C. **FLOOD** lights; Verify glare shield lights are illuminated and no other lights are on. D. **SPOT** lights; Verify overhead spotlights are illuminated and no other lights are on. E. With the position light switch on, the backlighting on the Flap panel should illuminate.
44. Visually check the integrity of all fuses (4) on the power grid located in the aft avionics compartment.
45. Pull all circuit breakers and ensure the labeling agrees with the circuit it supplies by pushing in the circuit breaker. Ensure positive latching of the circuit breaker.
46. Check the power point for voltage and verify polarity.
47. Check for painted or plastic coated screws in the instruments panels, and clean instrument dials.
48. Verify security and quality of the switches on the stick grip.
49. Verify the cooling hose is securely fastened and in place behind the instrument panel and the radio rack.
50. Verify AHRS calibration and perform AHRS calibration on the PFD if necessary.
51. Check wire runs and attachments of all wires and coax cables for abrasion potential, support, and wire tying. ☐ Along sides of fuselage ☐ Behind the stick grips ☐ Behind the instrument panel ☐ All flight controls and fluid lines ☐ Behind the radio stack ☐ Ensure wiring bundles are free and clear of rudder cables on the right side and free and clear of the circuit breaker panel and rudder limiter assembly on the left side

l. Verify compass card accuracy.

m. Compliance with Teledyne Continental Motors service bulletin #M88-9, for powerplant lightning strike inspection criteria, is recommended.

20-24. REPAIR OF STATIC WICKS

a. The nine static wicks on the aircraft are located at the tips of the wings, horizontal, outboard trailing edge of the ailerons, outboard trailing edge of the elevator, and the top of the rudder. The rudder static wick is mounted on a threaded stud and bonded electrically and mechanically to the aluminum counterweight at the top of the rudder. The other static wicks are mounted on static wick pads. These pads are bonded electrically and mechanically to the structure, and the static wick is attached with screws to this pad.

b. If the required conductivity is not achieved, these components must be replaced or rebonded with conductive adhesive.

1. Conductive Adhesive Material Requirements

a) Equipment:

Respirator mask – Commercial

Rubber gloves – Commercial

Sandpaper, 120 & 400 grit – Commercial

Applicator spatula – Commercial

18 oz. mixing container – Commercial

Shop towel – Commercial

b) Materials:

Conductive Adhesive – SX518001

Acetone, MEK, or Ethyl Acetates – Commercial

c. **Bonding Process** – The following process describes mounting pad and thread rod styles of the static wick mount installations. Follow the steps in the order presented.

1. Mounting Pad Static Wick Procedure

a) Mark the location of retainers on the aircraft skin per the corresponding drawing. Do not exceed $^{1}/_{8}$ in. between the aft end of the mounting pad and the trailing edge of the appropriate control surface.

b) Remove paint and primer, if any, from area to be bonded using 120 grit sandpaper. Expose the lightning protection mesh without sanding through the mesh material.

c) Degrease the aircraft surface and bottom of the retainer with a grease free solvent, such as acetone, MEK, or ethyl acetate. Do not contaminate the cleaned area.

d) Use either approved conductive adhesive per SX518001:

1) Mix one part LP-89 Part A Resin with one equal part LP-89 Part B Paste Catalyst. Mix thoroughly until a smooth, creamy consistency is reached. Pot life is 30 minutes at 70°F after mixing.

2) Take a premixed cartridge of Eccobond 64C A/B from the freezer. Allow it to thaw at room temperature. Work life is approximately 30 minutes from when the adhesive is removed from the freezer.

e) Abrade the clean, dry aircraft surface and the bonding surface of the static wick retainer with 200 to 600 grit paper. Do not abrade through the lightning protection mesh of the aircraft. Remove all oxide from the retainer. Dry thoroughly and wipe the surface with a clean towel to remove loose particles.

f) After removing the oxide, immediately apply a coat of adhesive to the retainer. The time between completion of the sanding process and the application must not exceed 5 minutes.

g) Upon completion of the previous step, immediately apply the retainer to the aircraft surface. Slightly twist the retainer while applying light pressure, to ensure thorough wetting and to squeeze out excessive adhesive. Once the retainer is in

place, do not disturb the bond further. Movement will increase the bond resistance and may weaken the mechanical strength.

h) Remove excessive adhesive, leaving a fillet of adhesive around the entire edge of the retainer.

i) Use masking tape to temporarily secure the mounting pad during curing.

j) Cure the adhesive for a minimum of 24 hours at 70°F. Cure may be accelerated by applying 150°F for one hour.

k) Ensure that all retainers are secure, with no evidence of cracks in the adhesive fillet between the retainer and the aircraft surface.

l) After the mounting pad has cured, check the electrical resistance to the lightning mesh with a digital voltmeter as close to the mount pad as possible. The resistance should be 0 to 2.0 ohms.

2. Rudder Threaded Rod Static Wick Procedure

a) Locate the static wick installation position in the vertical stabilizer counterweight.

b) Degrease the threaded rod, receptor threads in the rudder counterweight, and the receptor treads in the static wick with a grease-free solvent such as acetone, MEK, or ethyl acetate. Do not contaminate the cleaned area.

c) Use either approved conductive adhesive per SX518001:

1) Mix one part LP-89 Part A Resin with one equal part LP-89 Part B Paste Catalyst. Mix thoroughly until a smooth, creamy consistency is reached. Pot life is 30 minutes at 70ºF after mixing.

2) Take a premixed cartridge of Eccobond 64C A/B from the freezer. Allow it to thaw at room temperature. Work life is approximately 30 minutes from when the adhesive is removed from the freezer.

d) Place a small amount of adhesive on 0.5 in. of one end of the threaded rod. Install the threaded rod into the stabilizer counterweight, adhesive end first, to a depth of approximately 0.5 in. to the mechanical stop.

e) After the threaded rod has cured, check the electrical resistance to the counterweight material as close to the threaded rod as possible. Resistance should be 0 to 2.0 ohms.

f) After the aircraft has been painted, install the static wick as described below. Ensure that the threaded rod is masked for the aircraft paint process.

g) With a wooden applicator, place adhesive on the first 0.1 in. of the threaded rod above its base, to allow a removable bond to the tip of the static wick.

h) Insert the static wick onto the threaded rod, and ensure that it threads on the full length to its mechanical stop, allowing contact with the 0.1 in. of adhesive.

i) Cure the adhesive for a minimum of 24 hours at 70°F. Cure may be accelerated by applying 150°F for 1 hour.

j) Ensure that the static wick is secure.

d. **Inspection Procedures** – Visually inspect all retainer installations thoroughly. Ensure that the retainer has no evidence of cracks in the adhesive fillet between the retainer and the aircraft surface. Ensure that all static wicks are secure.

e. **Rework** – If a mounting pad has evidence of cracks in the adhesive fillet that exceed the maximum length criteria, the static wick mounting pad must be removed and re-installed. If the threaded rod-style static wick is not secure, remove, clean, and re-install the static wick threaded rod as per procedure.

f. **Acceptance Criteria**

1. ADHESIVE – The adhesive application shall be acceptable if there is a smooth surface fillet between the retainer and the aircraft surface, with no evidence of cracks or unbonded areas in the adhesive of over 0.1 in. in length.
2. IMPROPER CURE – The adhesive application shall be acceptable if there is no evidence of tackiness in the adhesive after the proper cure time.
3. SECURE INSTALLATION – The static wick mount installation shall be acceptable if there is no evidence of improper bonding of the static wick pad or threaded rod allowing any movement of the static wick or evidence of cracking and tackiness of the adhesive.
4. ELECTRICAL RESISTANCE – The static wick mount installation shall be acceptable if the electrical resistance of the mounting connection is no more than 2.0 ohms, as measured from the mount to the lightning mesh or counterweight.

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20-25. REPAIR OF CLICKBOND NUTPLATES

Clickbond nutplates are a special type of nutplate that is bonded to the surface rather than being riveted to the structure. If a Clickbond nutplate should fail, it can be replaced with another Clickbond nutplate or a riveted nutplate under certain conditions as specified in Figure 20 - 39. This repair procedure can be used in lieu of specific factory instructions.

Clickbond Part Number	Application	Assembly Drawing Number
CB6009CR4-1P	Baggage Net Installation	LB53255000
CB6014CR3-1P	Baggage Net Installation Cowling Instrument Panel Installation	LB53255000 LA53600000 LB53310000
CB6014CR08-1P	Access Panel Installation, Fuselage	LB53650000

Figure 20 - 39 Permitted Areas of Clickbond or Riveted Nutplate Repair

a. **Defect Scenarios**
   1. The nut in the Clickbond nutplate is stripped or damaged.
   2. The bonding of the Clickbond to the surface has failed.

b. **Repair Procedure No. 1** – Bond a new Clickbond of the same type as a replacement for the one that failed.
   1. Remove the old Clickbond nutplate by prying it off.
   2. Sand the substrate to prepare a smooth surface using 150 grit sandpaper.
   3. Clean the surface with a dry-wipe or clean, pressurized air.
   4. Bond on a new nutplate. Use Clickbond adhesive kit CB92 or adhesive Hysol 9321. Follow manufacturer's instructions.

c. **Repair Procedure No. 2** – Replace the Clickbond with riveted nutplate.
   1. Sand the adhesive residue down to create an even surface for the nutplate. Take care not to damage the structural material below.
   2. Use a nutplate from Figure 20 - 40 to determine the proper riveted nutplate replacement.

Clickbond Nutplate	MS Nutplate	Drill ∅ Rivet Holes	Rivet Hole Spacing From Center Hole
CB6009CR4-1P	MS21075-4	0.098 in.	0.281 in.
CB6014CR3-1P	MS21075-3	0.098 in.	0.250 in.
CB6014CR08-1P	MS21075-08	0.098 in.	0.234 in.

Figure 20 - 40 MS Nutplate Replacements for Clickbonds

NOTE

The old Clickbond nutplate can be reused if the nut still has sufficient self-locking friction. Clean and sand the surface on the Clickbond nutplate for good adhesion, and follow the procedures discussed in Repair No. 1.

20-26. SWAGELOK FITTINGS

a. The information in this section is taken from the *Swagelok Tube Fitter's Manual.* The Swagelok manual should be referenced if there is any question about the information presented in this manual. Formal training from Swagelok or Cessna is recommended before using any Swagelok fittings.

b. The Swagelok fittings used in the Cessna aircraft provide a consistent tube connection for both fuel and hydraulic systems without the need for flaring tubing. These flareless, mechanical grip fittings consist of a fitting body, front ferrule, back ferrule, and nut, as seen in Figure 20 - 41. The fittings are received from Swagelok pre-assembled and normally ready for use, with the exception being the bulkhead fittings used in some systems and in other specific applications. With the installation of the bulkhead fittings, one end of the fitting must be disassembled to install the fitting and the removed end reassembled prior to installation of the tubing Figure 20 - 41 shows the correct orientation of the body, ferrules and nut for proper installation.

c. When installing Swagelok fittings, always maintain the cleanliness of the hose, fittings, and tube interiors. Protect the threads from damage and dust.

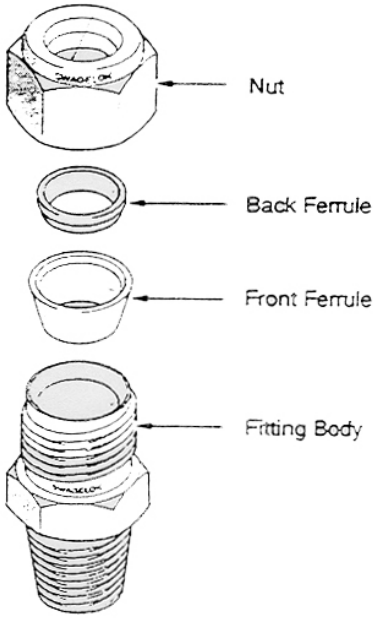

Figure 20 - 41 Swagelok Fitting (Exploded View)

d. **Swagelok Fitting Installation**

1. Visually inspect the fitting to verify that the ferrules are in place and installed correctly as shown in Figure 20 - 42 and Figure 20 - 43. When look inside the end of the fitting, three lines should be seen.

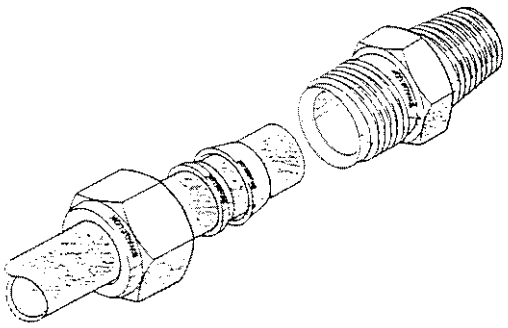

Figure 20 - 42 Proper Ferrule Position

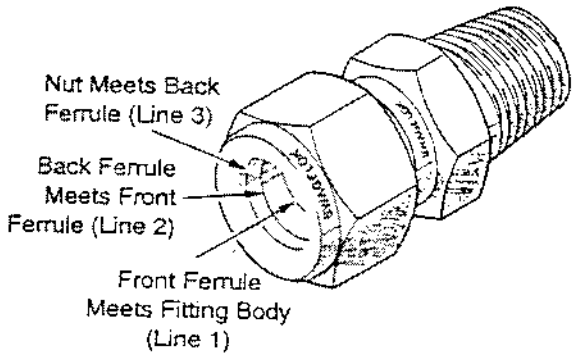

Figure 20 - 43 Swagelok Fitting Visual Inspection

2. Insert the tubing into the tube fitting such that the tubing firmly rests on the shoulder of the fitting as shown in Figure 20 - 44.

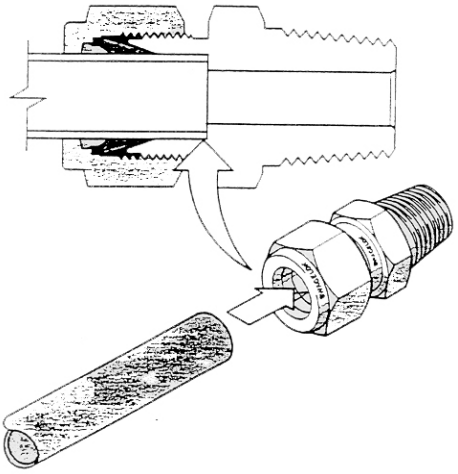

Figure 20 - 44 Fully Insert Tubing into Fitting

3. Using a permanent marker, mark the 6 o'clock position on the nut and extend onto the tube for reference as shown in Figure 20 - 45. In addition, with the tube fully inserted in the fitting and the nut finger-tight, mark on the tube the intersection between the tube and the nut.

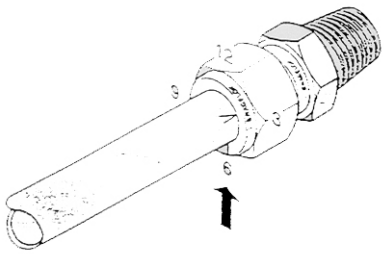

Figure 20 - 45 Mark Nut with a Permenant Reference Mark

4. While holding the fitting body with a backup wrench, tighten the nut 1-1/4 turns as shown in Figure 20 - 46. *For $^{1}/_{16}$ in., $^{1}/_{8}$ in., and $^{3}/_{16}$ in., 2 mm, 3 mm, and 4 mm fittings, only a ¾ turn from finger-tight is necessary.*

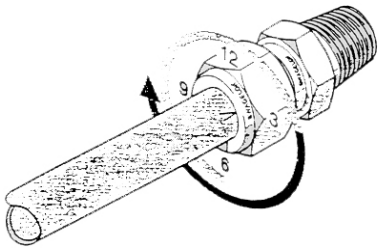

Figure 20 - 46 Tighten Nut 1-1/4 Turns for New Fittings

5. Verify installation per paragraph e of this section.

e. **Swagelok Fitting Gauging**

After the Swagelok fittings are installed, the design of the fitting allows the installation to be verified using a "go no-go" gauge. This gauge may only be used for standard fitting and unions. For bulkhead fittings and other special fittings the gauge cannot be used because the distance from the body to the nut is variable.

CAUTION

Only use Swagelok gap inspection gauges on Swagelok tube fittings. The gauge will not work properly on other tube fittings.

NOTE

Swagelok gap inspection gauges should not be used after pre-swaging or hydraulic swaging.

NOTE

Swagelok gap inspection gauges should be used only for the initial installation of the fitting and not for re-installation of fittings or installation of pre-made ends.

1. Figure 20 - 47 shows the use of the Swagelok inspection gauge and the indication of a correctly installed fitting. Note that if the gap inspection gauge does not fit between the nut and body hex, the fitting is sufficiently tightened.
2. Figure 20 - 48 shows an improperly installed fitting. Note the gap inspection tool goes into the slot between the nut and the body hex. Additional tightening is required or further investigation may be required.

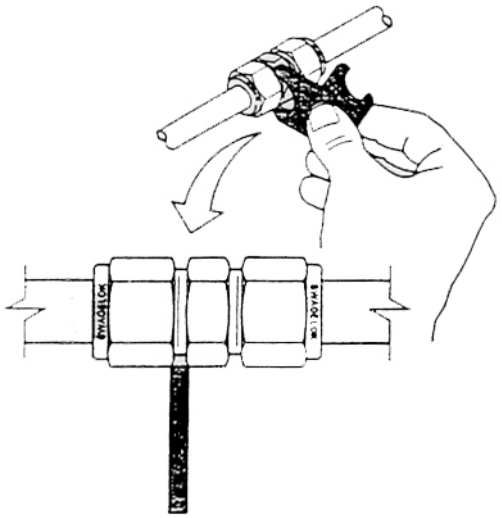

Figure 20 - 47 Insert Swagelok Gauge to Verify Correct Torque of Newly Installed Fitting

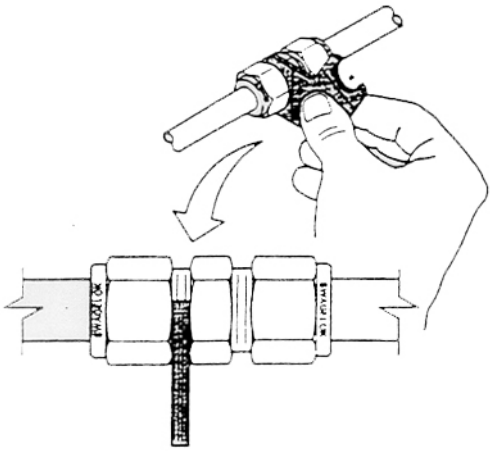

Figure 20 - 48 Gauge Shows Fitting is Not Correctly Installed and Torqued

f. **Swagelok Fittings Retightening (Installing Pre-Swaged Fittings)**

The Swagelok fittings may be disconnected and retightened many times and still provide a reliable, leak-proof seal if the following instructions are followed.

1. Clean both parts of the fitting including the tubing in the swaged area, and inspect for scratches, burrs, etc. If the tube and sealing area are damaged, replace part (See Figure 20 - 49).

Figure 20 - 49 Diconnected Pre-Swaged Swagelok Fitting

2. Fully insert the tubing with the pre-swaged ferrules into the fitting body as shown in Figure 20 - 50, and finger-tighten the nut onto the fitting body.

Figure 20 - 50 Inserted Pre-Swaged Swagelok Fitting

3. Rotate the nut to the original position with a wrench, stopping when an increase in resistance is felt. Tighten slightly past the original position using a wrench. Figure 20 - 51 shows a fully tightened fitting.

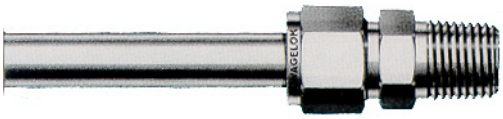

Figure 20 - 51 Tightend Pre-Swaged Swagelok Fitting

NOTE

Smaller tubing sizes require less tightening to reach the previously pulled-up position. Larger tubing sizes require more tightening. Wall thickness can also affect the amount of torque required to retighten the fittings.

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20-27. TAP TESTING

Tap testing is performed to identify areas of disbonds in the structure. The tap testing procedure consists of lightly tapping the surface of the part with a coin or small hammer (no more than 2 ounces). The acoustic response is compared with that of a known good area. A “flat” or “dead” response is considered unacceptable. Although the acoustic response can change somewhat in different areas depending on the geometry of the part, a “flat” or “dead” response can be identified when the sound changes significantly within a 1 inch area. Cessna recommends that maintenance personnel performing tap tests be trained at the Cessna facility to properly identify disbonds.

The inspections listed in Chapter 4 require that tap testing be completed in the following areas:

a. **Fuselage bonding seams** — The fuselage halves are split into left and right symmetrical sides and include the areas from the firewall back to and including the vertical stabilizer. The seam extends from centerline to 2 in. on the *right* side of centerline. The seam tapers to 1.75 in. at approximately FS 240 and tapers again to 1.5 in. at approximately FS 261.5.
b. **Wing skin leading edge and trailing edge seams** — The wing skin seams are at the leading and trailing edges of the wing. The leading edge seam is 2 in. wide and centered about the leading edge. The trailing edge should be tap tested on the top and bottom of the edge from the edge to 2 in. inboard.
c. **Wing skin to ribs and spars —** The location of each rib and spar is shown in section 57-6. The top and bottom wing skin at the areas of the ribs and spars should be tap tested.
d. **Wing flaps leading and trailing edge seams —** The flap seams are at the leading and trailing edges of the flap. The leading edge is 1 in. wide and extends 1 in. up from the leading edge chord plane of the flap. Because the bond extends onto core as shown in Figure 20 - 52, tap testing is only valid for the first 0.5 in. of the bond. The trailing edge should be tap tested on the top and bottom of the edge from the edge to 1.5 in. forward.

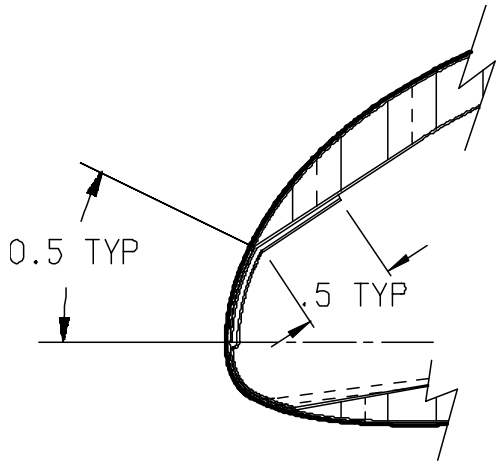

Figure 20 - 52 Wing Flap Leading Edge Tap Testing Area

e. **Aileron leading and trailing edge seams** — The aileron seams are at the leading and trailing edges of the ailerons. The leading edge seam is 1.0 in. wide starting 1 in. along the surface from the leading edge of the aileron. In the area around the drive hinge attachment pocket, tap test approximately 1.0 in. from the edge of the part to verify the bonding surface. The trailing edge should be tap tested on the top and bottom of the trailing edge from

approximately 0.90 in. from the edge to approximately 2.0 in. from the trailing edge as shown in Figure 20 - 53.

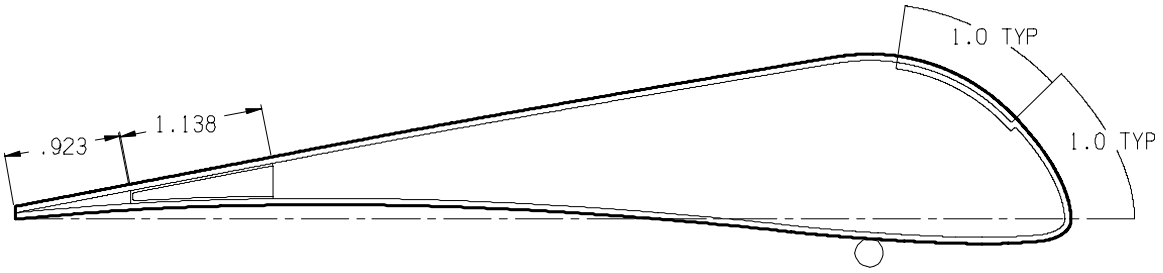

Figure 20 - 53 Aileron Tap Test Areas

f. **Rudder leading and trailing edge seams** — The rudder seams are at the leading and trailing edges of the rudder. The leading edge seam is the width of the flat leading edge of the rudder. The trailing edge seam should be tested as follows: 1) the top 14 in. should be tested by tapping on the right hand side a distance of 1.5 in. inboard from the trailing edge; 2) the remainder of the trailing edge should be tested 1.0 in. inboard from the trailing edge. See Figure 20 - 54 for areas of tap testing.

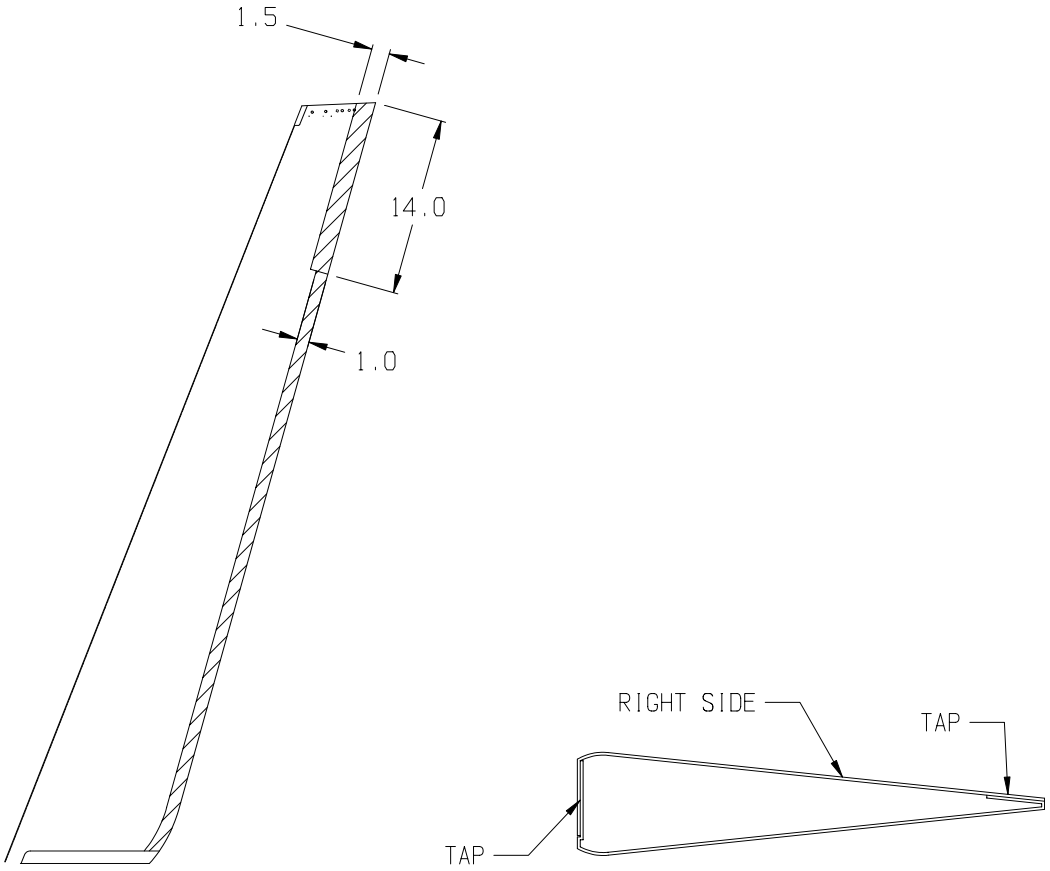

Figure 20 - 54 Rudder Tap Test Areas

CHAPTER 21

ENVIRONMENTAL CONTROL SYSTEM

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21-1. GENERAL – SCOPE AND DEFINITION

a. Environmental control in the Cessna 350 is accomplished with either an Environmental Control System (ECS) or the optional air conditioning system also known as the Automatic Climate Control System (ACCS). For information concerning the ACCS see section 21-100-00. The ECS, shown in Figure 21 - 1, controls temperature throughout the front and rear cabin area. The ECS incorporates the use of an air-to-air heat exchanger, ram intake air, and an electric fan to distribute heated and outside air to various outlets within the cabin. The ECS consists of two subsystems, heated air and fresh air. Heated air is sent to the floor vent system and defroster, and fresh air is ducted through the eyeball vents. Ram air enters through a duct on the right side of the engine cowling and flows to either the heat exchanger (located on the right exhaust manifold) or the fresh air manifold. Air to the heat exchanger, depending on the control settings, is mixed with outside air in the heater box. The air next passes through a fan unit before entering the distribution system. Operating the single speed fan will increase the airflow through the system (not including the eyeball vents).

b. An air mixing valve takes both incoming hot and fresh air and mixes them according to the needs of the passengers.

c. At the windshield area, there is a defrost plenum directly built into the fuselage. This will allow the pilot or copilot to defrost the windshield during cold weather conditions and supply fresh air to that location during periods of high ambient temperature.

d. Located in the instrument panel are adjustable eyeball vents, one each for the pilot and copilot. These vents supply only fresh air to the front seat positions and may be manually directed to a desired location.

e. Near the pilot and copilot’s feet is a floor vent tube. Like the defrost plenum, this tube can direct hot, fresh, or mixed air to the lower front cabin area.

f. Directly tied into the floor vent tube and located at the aft end of the center console is an adjustable eyeball vent. This vent supplies the rear passengers the same air temperature as the floor and defrost vents but with the ability to be manually turned off independently or directed to a desired location.

g. Just to the outside of each rear passenger and located on the interior side panel is a fresh air eyeball vent. Like the front, these vents supply only fresh air and may be manually adjusted or directed to a desired location.

h. An electrically controlled fan may be turned on while on the ground or during flight to supply cooler fresh air through the defrost plenum, floor tube, and rear passenger center console eyeball vent during the summer or aid in deicing the windshield before and after takeoff during the winter months.

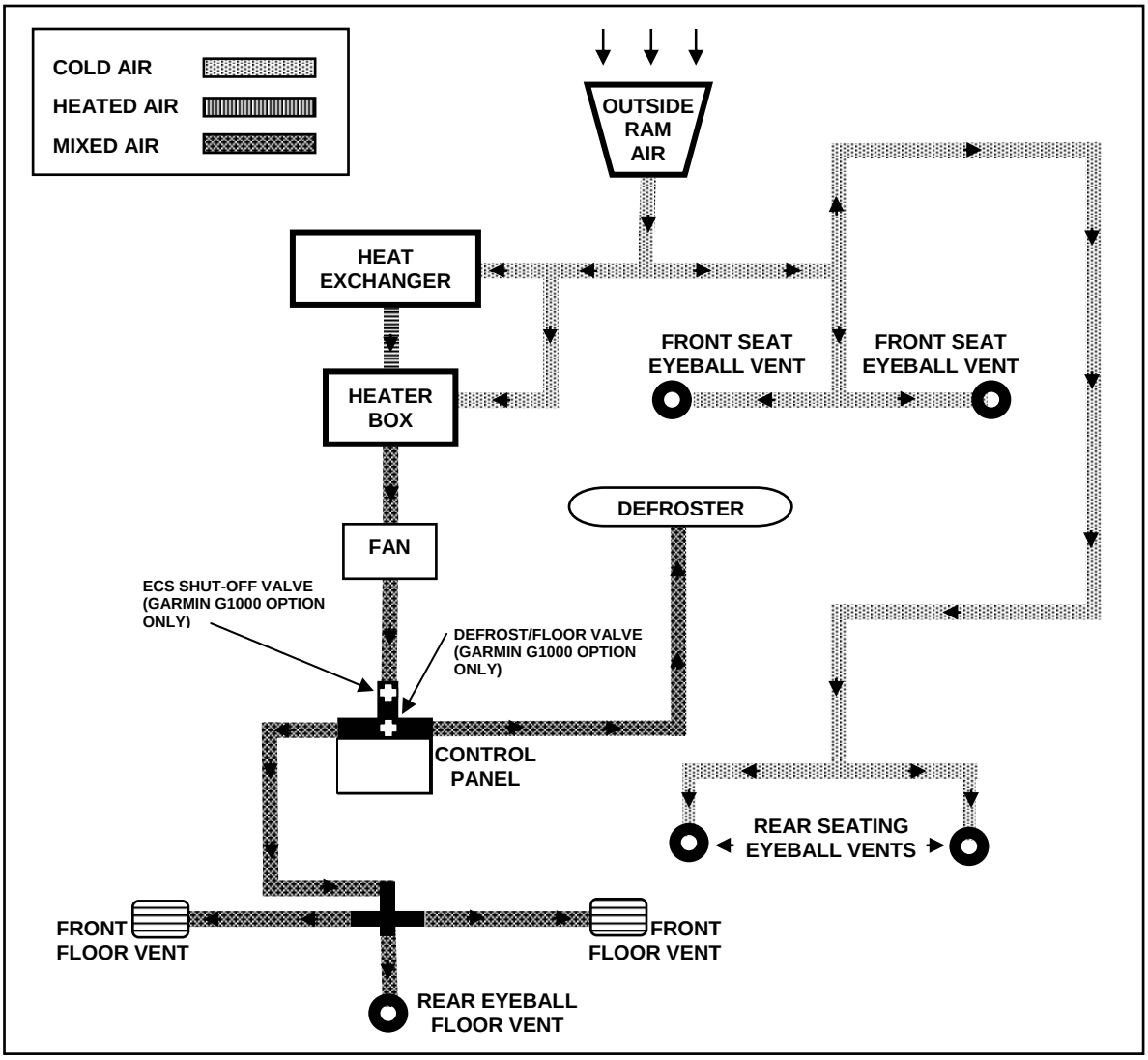


Figure 21 - 1 Environmental Control System Diagram

21-2. DISTRIBUTION SYSTEM

The components in the distribution system depend upon whether Avidyne avionics or the Garmin G1000 system is installed in the aircraft.

a. **Basic or Avidyne Option without Air Conditioning** – The distribution system includes the fan, ECS control valve, servo motor, defrost plenum, control panel, and eyeball and floor vents.
b. **Basic or Avidyne Option with Air Conditioning** – The distribution system includes the fan, ECS control valve, servo motor, ECS shut-off valve assembly, defrost/floor valve assembly, defrost plenum, control head, and eyeball and floor vents.
c. **Garmin G1000 Option with or without Air Conditioning** – The distribution system includes the fan, ECS control valve, servo motor, ECS shut-off valve assembly, defrost/floor valve assembly, defrost plenum, control head, and eyeball and floor vents.

21-3. ECS FAN

a. **Basic or Avidyne Option without Air Conditioning**
   1. **Fan Removal**
      a) Unplug the electrical connector plug, then cut and remove the plastic tie straps located at the inlet and outlet ports of the fan.
      b) Pull the flexible ducting away from the fan.
      c) Loosen and disconnect the main mounting clamp until the fan can be freed from the cradle housing.
      d) Remove the fan by pulling it out from underneath the instrument panel. See Figure 21 - 5.
   2. **Fan Installation**
      a) Position the fan in the cradle leaving enough room between the heater control box and the fan inlet port for flexible ducting.
      b) Connect and tighten the main mounting clamp around the fan motor housing.
      c) Slide the flexible ducting over the inlet and outlet ports of the fan, and secure using plastic tie straps.
      d) Reconnect the ELC connector.
      e) Turn on the power, and test the fan for proper operation.

b. **Garmin G1000 Option, Basic Option, or Avidyne Option with Air Conditioning**
   1. **Fan Removal**
      a) Unplug the electrical connector plug, then cut and remove the plastic tie straps at the inlet port, and remove the hose clamp at the outlet port of the fan.
      b) Pull the ECS shut-off valve assembly away from the outlet port of the fan. See Figure 21 - 5, and Figure 21 - 6.
      c) Pull the flexible ducting away from the fan.
      d) Loosen and disconnect the main mounting clamp until the fan can be freed from the cradle housing.
      e) Remove the fan by pulling it out from underneath the instrument panel.
   2. **Fan Installation**
      a) Position the fan in the cradle.
      b) Connect and tighten the main mounting clamp around the fan motor housing.
      c) Slide the flexible ducting over the inlet port of the fan, and secure using plastic tie straps.

d) Slide the end of the ECS shut-off valve assembly over the outlet port of the fan. Apply RTV5818 silicone all around and secure with a hose clamp. See Figure 21 - 6.
e) Reconnect the ELC connector.
f) Turn on the power, and test the fan for proper operation.

c. **Fan Maintenance** – The electrically controlled ECS fan is a self-contained unit and requires only a periodic check for proper operation.

21-4. SERVO MOTOR

a. **Servo Motor Removal**
1. With power on, adjust the temperature knob in the instrument panel to the full cold position, then turn power off.
2. Remove the ECS control valve per section 21-40-00 of this manual.
3. Remove the servo motor to ECS control valve attachment screws, washers, and nuts.
4. Pull the servo motor back and away from the control valve until the servo motor drive arm to linear drive rod clevis is exposed.
5. Remove the cotter pin, washer, and clevis pin, and remove the servo motor from the control valve.

b. **Servo Motor Installation**
1. With the servo motor arm in the fully extended position, install the servo motor drive arm into the linier actuator clevis, and reinstall the clevis pin, washer, and a new cotter pin.
2. Align the mounting holes in the servo motor with the mounting holes in the ECS control valve. Install screws, washers, and nuts only tight enough so that servo motor can be pushed forward by hand.
3. With the heater control valve secured, hand push the servo motor forward until the silicone sealing cushion inside of ECS control valve is fully seated against metal sealing ring. Tighten the servo mounting hardware.

c. **Maintenance –** The ECS control servo motor is a self-contained unit and requires only a periodic check for proper operation.

21-5. ECS SHUT-OFF VALVE ASSEMBLY

The ECS shut-off valve assembly is only present with the Garmin G1000 option or the Basic or Avidyne option with air conditioning.

a. **ECS Shut-Off Valve Removal**
1. Unplug the electrical connector plug .
2. Remove the hose clamps at the inlet and outlet ports of the ECS shut-off valve. See Figure 21 - 5 and Figure 21 - 6.
3. Pull the flexible ducting away from the outlet port.
4. Pull the ECS shut-off valve assembly away from the outlet port of the ECS fan.

b. **ECS Shut-Off Valve Installation**
1. Apply RTV5818 silicone all around the outlet port of the ECS fan and the outlet port of the ECS shut-off valve.
2. Slide the inlet port of the ECS shut-off valve over the outlet port of the ECS fan. Rotate the valve as required to clear the rudder pedal tubes and ECS servo motor actuator screw and secure with a hose clamp.

3. Slide the flexible ducting over the outlet port of the ECS shut-off valve, and secure with a hose clamp.
4. Reconnect the electrical connector.
5. Turn on the power, and test the ECS shut-off valve for proper operation.

c. **Maintenance –** The ECS shut-off valve assembly is a self-contained unit and requires only a periodic check for proper operation.

21-6. DEFROST/FLOOR VALVE ASSEMBLY

The defrost/floor valve assembly is only present with the Garmin G1000 option or the Avidyne option with air conditioning.

a. **Defrost/Floor Valve Removal**
   1. Unplug the electrical connector plug .
   2. Remove the hose clamps at the inlet and outlet ports of the defrost/floor valve. See Figure 21 - 5 and Figure 21 - 6.
      a) On Garmin G1000 aircraft, the defrost/floor valve assembly may be attached directly to the defrost channel. With this condition, remove the hose clamp and a MS51861-25C screw. There is a length of silicone sheeting between the defrost/floor valve and the defrost channel.
   3. Pull the flexible ducting away from the inlet and outlet ports.
   4. Remove the defrost/floor valve.

b. **Defrost/Floor Valve Installation**
   1. Defrost/floor valve attached directly to the defrost channel.
      a) Fill the gap between the inner diameter of the valve flange and the outer diameter of the defrost channel flange with .062” or .093” AMS3320 silicone sheet.
      b) When installing a new defrost/floor valve, match drill a 0.166” diameter (#32) hole in the valve flange.
      c) Install a MS51861-25C screw through the valve flange, the defrost channel flange, and the silicone sheet. Dry torque hand tight plus 1/4 turn.
      d) Apply RTV5818 silicone all around the inlet port, install flexible ducting, and secure with a hose clamp.
      e) Reconnect the electrical connector.
      f) Turn on the power, and test the defrost/floor valve for proper operation.
   2. Defrost/floor valve not attached directly to the defrost channel.
      a) Apply RTV5818 silicone all around the inlet and outlet ports of the defrost/floor valve.
      b) Slide the flexible ducting over the inlet and outlet ports of the defrost/floor valve. Rotate the valve as required to clear the rudder pedal tubes and ECS servo motor actuator screw, and secure with hose clamp.
      c) Reconnect the electrical connector.
      d) Turn on the power, and test the defrost/floor valve for proper operation.

c. **Maintenance –** The defrost/floor valve assembly is a self-contained unit and requires only a periodic check for proper operation.

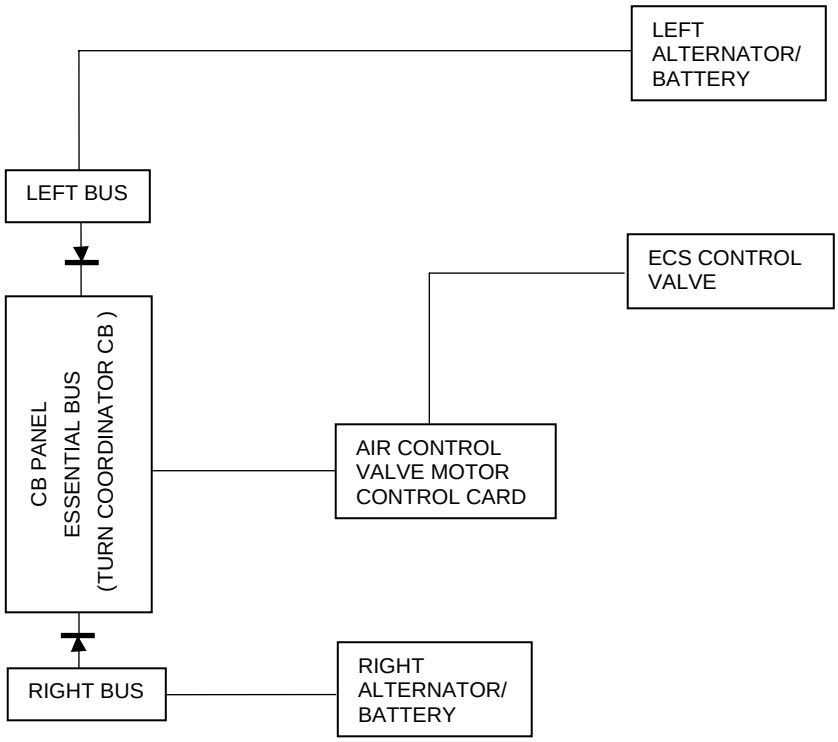


Figure 21 - 2 Schematic of ECS Electrical System Connection

21-7. DEFROST PLENUM

a. **Removal/Installation –** The defrost plenum is an integral part of the fuselage and contains no moving parts; therefore, no removal is required.

b. **Maintenance –** The defrost plenum is an integral part of the fuselage, therefore, no maintenance is required.

21-8. EYEBALL VENTS (INSTRUMENT PANEL)

a. **Eyeball Vent Removal –** To remove the left and/or right eyeball vent(s) from the instrument panel use the following instructions. Steps 1 and 2 do not apply for the Garmin G1000 system instrument panel.

1. Remove the left side instrument panel and/or the right side kidney panel screws.
2. Gently pull the left side instrument panel and/or the right side kidney panel away from the main panel until the eyeball vent flexible ducting can be seen.
3. Remove the cable tie securing the tubing to the back of the plenum cap. For the Garmin G1000 system, the back of the vent may be accessed from underneath the instrument panel.
4. Loosen the clamp securing the plenum cap to the backside of the eyeball vent, and gently pull away from the vent.
5. Loosen and remove the threaded eyeball vent retaining collar from the backside of the panel.

6. Pull the eyeball vent out from the front of the panel. See Figure 21 - 3 for a schematic of the eyeball vent.

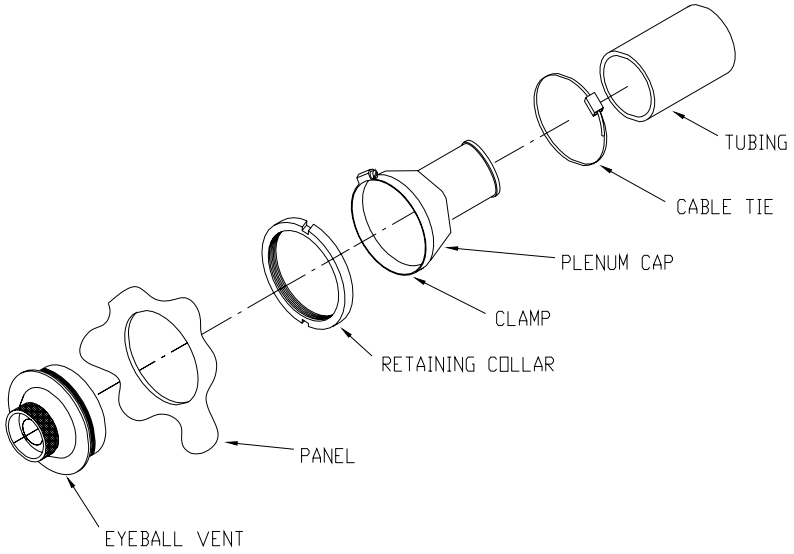


Figure 21 - 3 Eyeball Vent Installation

b. **Eyeball Vent Installation –** To install the left and/or right eyeball vent(s) into the instrument panel use the following instructions. Step 5 does not apply to the Garmin G1000 system instrument panel.
   1. Insert the eyeball vent(s) into the front of the left side instrument panel, into the front of the right kidney panel, or into the lower instrument panel (Garmin G1000 system), as applicable.
   2. Install the threaded eyeball vent retaining collar to the backside of eyeball vent.
   3. Press the plastic plenum cap onto the backside of the eyeball vent, and secure with clamp. When the clamp is secured, the screw housing must remain at the 10 to 12 o'clock position for the left eyeball vent and at the 12 to 3 o'clock position for the right eyeball vent, as viewed from the rear.
   4. Slide the tubing over the back of the plenum cap and secure with a new cable tie.
   5. Install the instrument panel and/or the kidney panel to the main panel with screws.

21-9. EYEBALL VENTS (REAR INTERIOR SIDE PANELS)

a. **Removal –** To remove the rear interior side panel eyeball vents use the following instructions.
   1. Remove the rear interior panel attachment screws, then gently pull the interior side panel up and away from the fuselage until access is gained to the rear of the eyeball vent.
   2. Loosen and remove the threaded eyeball vent retaining collar from the backside of the panel.
   3. Pull the eyeball vent out from the front of the panel.

b. **Installation –** To install eyeball vents into the rear interior side panel use the following instructions.
   1. Insert the eyeball vent into the side panel hole.

2. Install the threaded eyeball vent retaining ring to the backside of the eyeball vent.
3. Press the panel back into place ensuring the vent is seated in the ducting channel.
4. Install the rear interior panel to the fuselage, and attach it with screws.

c. **Maintenance –** The rear interior side panel eyeball vents are self-contained, therefore, no maintenance is required.

21-10. FLOOR VENT (FRONT)

a. **Removal/Installation –** The front floor vent is an integral part of the radio rack frame with no moving parts; therefore, no removal is required.

b. **Maintenance –** The front floor vent is self-contained; therefore, no maintenance is required.

21-11. FLOOR VENT (REAR)

a. **Floor Vent Removal –** To remove the rear floor vent use the following instructions.
   1. Gently pull the center console panel out and away from the main console.
   2. With the rear heat vent flexible ducting exposed, loosen the clamp securing the plastic plenum cap to the backside of the eyeball vent, and gently pull the plastic housing away from vent.
   3. Loosen and remove the threaded eyeball vent retaining collar from the backside of the panel.
   4. Pull the eyeball vent out from the front of the panel.

b. **Floor Vent Installation –** To install eyeball vents onto the center console, use the following instructions.
   1. Insert the eyeball vent into the side panel hole.
   2. Install the threaded eyeball vent retaining collar to the backside of the eyeball vent.
   3. Press the plastic plenum cap onto the backside of the eyeball vent, and secure with a clamp.
   4. Install the center console panel onto the main panel until the duel lock stripping is fully engaged.

c. **Maintenance –** The rear floor vent is self-contained; therefore, no maintenance is required.

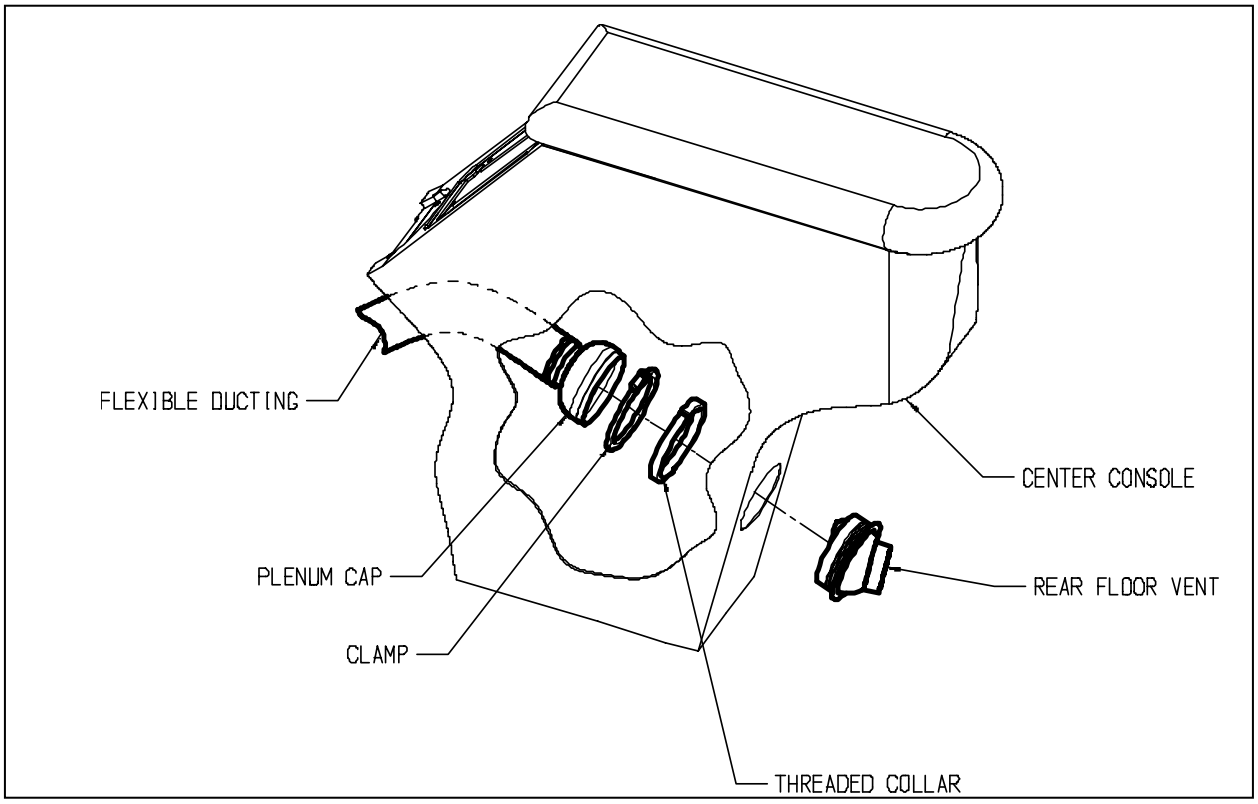


Figure 21 - 4 Rear Floor Vent

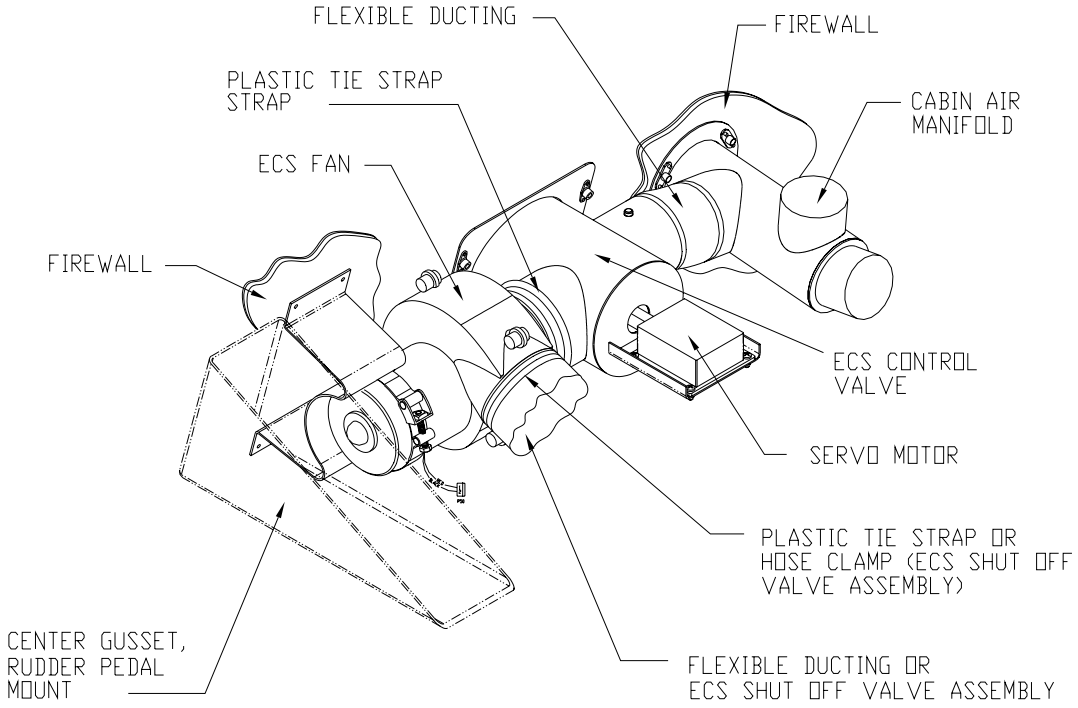
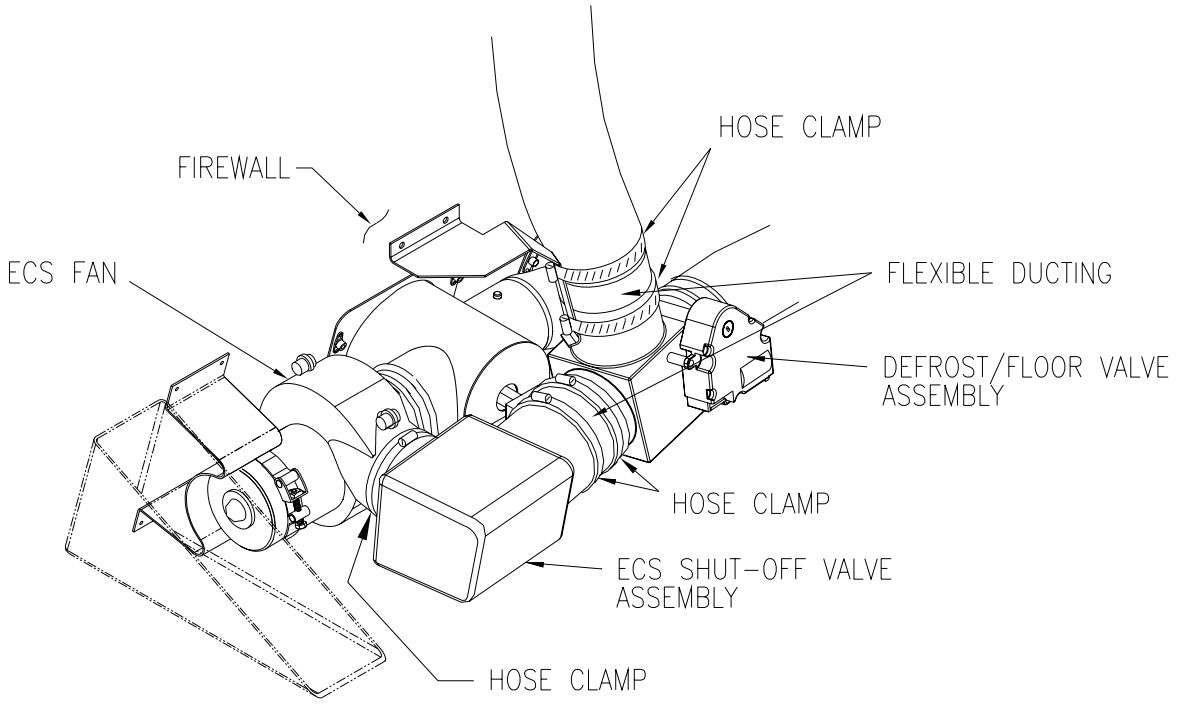
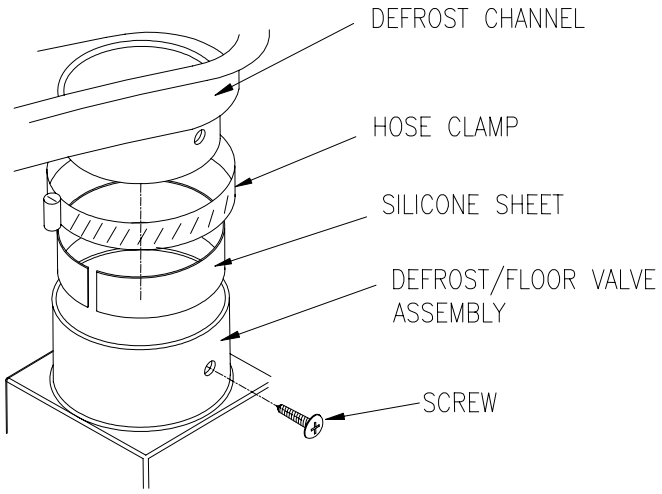


Figure 21 - 5 ECS System Attachment



CONFIGURATION WITH DUCTING BETWEEN THE
DEFROST/FLOOR VALVE ASSEMBLY AND THE DEFROST CHANNEL



CONFIGURATION WITH THE DEFROST/FLOOR VALVE ASSEMBLY
ATTACHED DIRECTLY TO THE DEFROST CHANNEL

Figure 21 - 6 ECS Shut-Off Valve and Defrost/Floor Valve Assemblies

21-12. HEATING AND ECS CONTROL VALVE

a. The following procedures are applicable to the heater and ECS control valve.

21-13. ECS CONTROL VALVE

a. **Removal** – To remove the ECS control valve from the airplane, use the following instructions.

1. Loosen the steel clamp securing the flexible ducting to the ECS control valve inlet tube located in the engine compartment, and pull the ducting away from the tube. See Figure 21 - 7.
2. Cut and remove the plastic tie straps from the inlet and outlet tubes of the ECS control box, and pull the flexible ducting away from the tubes (see Figure 21 - 5).

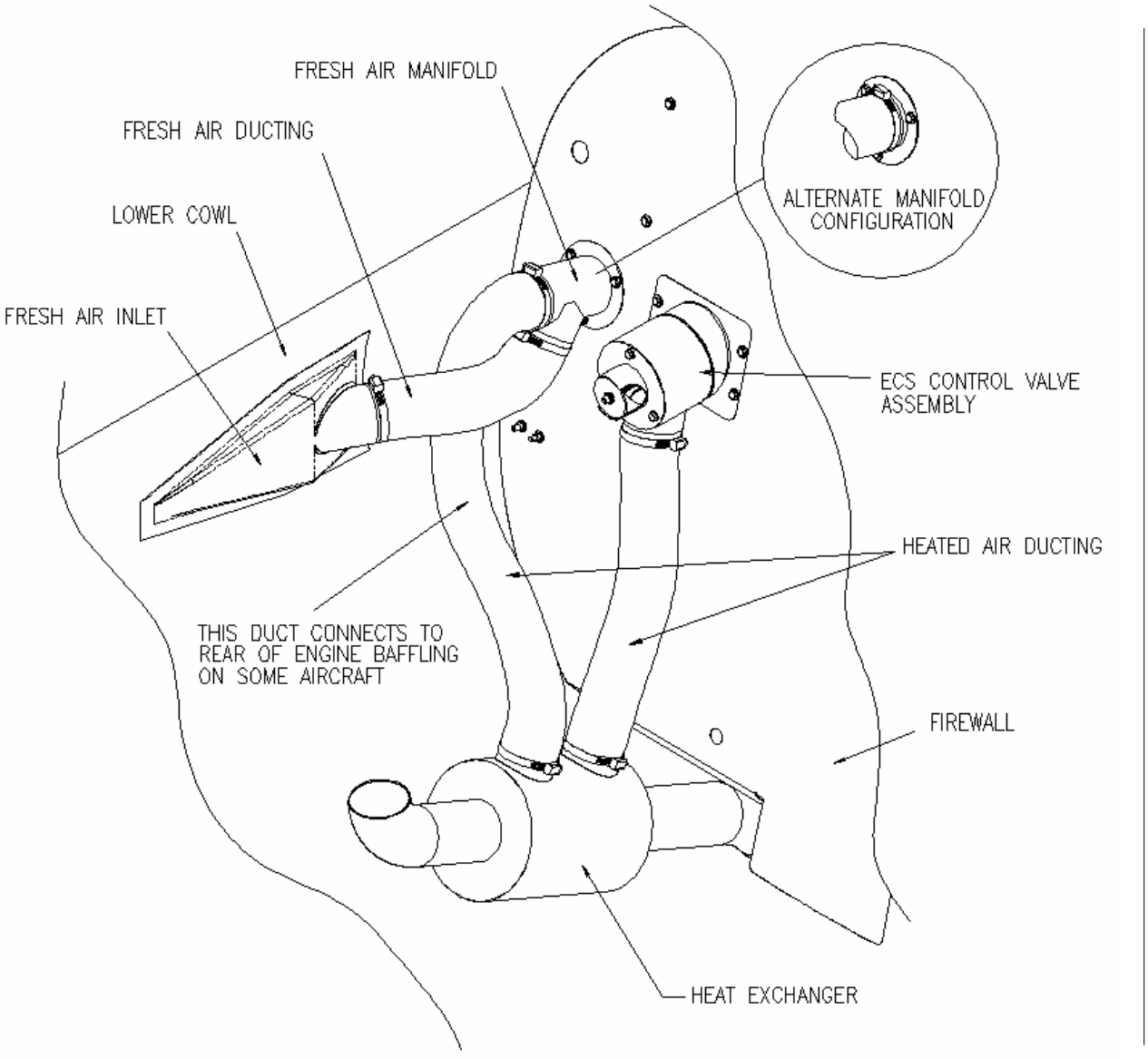


Figure 21 - 7 ECS Assembly in Engine Compartment

3. Disconnect the heater control valve servo motor ELC plug.
4. Remove the faceplate screws and washers, and pull the faceplate outward until the inner plunger seal can be seen.
5. Remove the control box bolts and washers. Pull the control box out from under the instrument panel (see Figure 21 - 8).

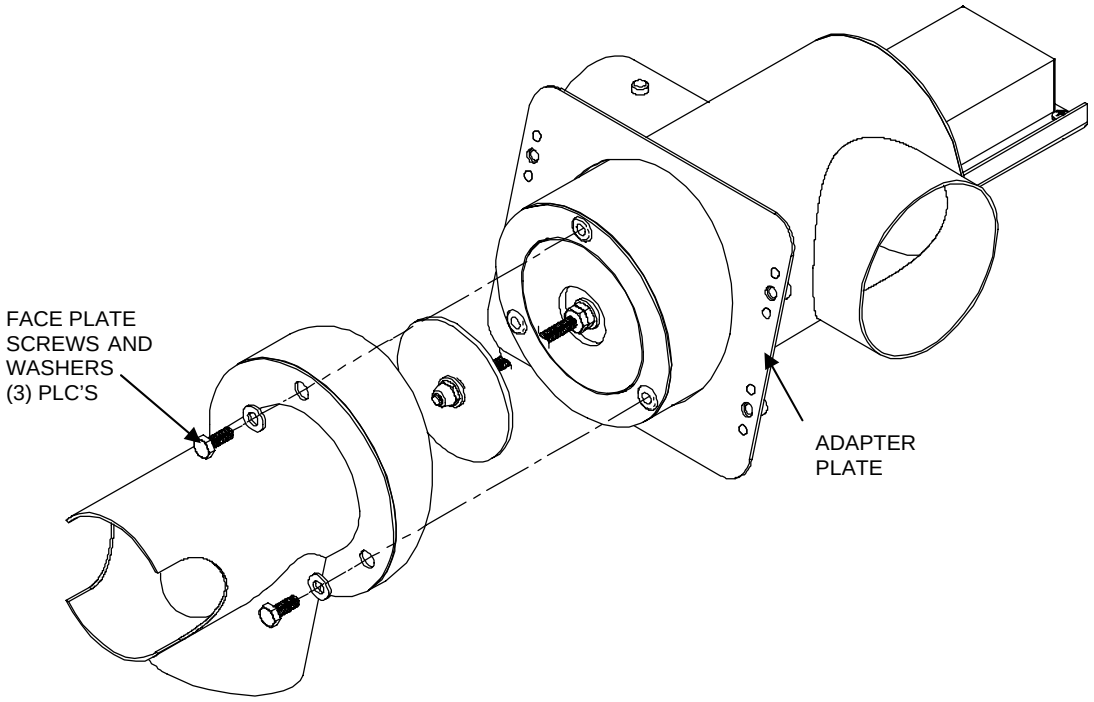


Figure 21 - 8 ECS Control Valve and Faceplate Attachment

b. **ECS Control Valve Installation** – To install the heater control box use the following instructions:
   1. Insert the control box into the opening in the firewall from inside of the cabin area with the servo motor facing up.
   2. Install the control valve mounting bolts and washers from the engine compartment side of the firewall.
   3. Install the faceplate onto the front of the control box, then install the faceplate mounting screws and washers.
   4. Reattach the flexible ducting to the faceplate inlet tube with a steel clamp.
   5. Turn on the power and test the heater control box for proper valve operation and adjust the servo motor per paragraph 21-4.b. if necessary.

c. **Maintenance** – The heater control box is a self-contained unit. Only a periodic check for proper valve operation is required.

21-14. TEMPERATURE AND AIR CONTROL UNIT – One of three types of temperature and air control units will be present in the aircraft depending upon avionics installed (basic, Avidyne or Garmin G1000) and upon installation of optional Automatic Climate Control System (ACCS). For aircraft with Garmin G1000 avionics and without ACCS, the temperature and air control unit is similar to the control head of the ACCS with identical removal and installation procedures (see section 21-100-00). For aircraft with basic instruments only or Avidyne avionics, and without ACCS see below. For aircraft with optional ACCS see section 21-100-00.

a. **Removal of Air Control Unit** – To remove the temperature and air control unit use the following instructions.

1. Turn the power off and remove the temperature and air control knobs from the control unit faceplate located on the instrument panel by removing the set screws at the bottom of each knob (see Figure 21 - 9).
2. Remove the fan rocker switch by gently squeezing together the locking tabs located on the outer perimeter of the switch from behind the instrument panel.
3. Carefully pull the rocker switch away from the instrument panel until wiring is exposed.
4. Disconnect the wiring from the back side of the rocker switch, and remove the switch.
5. Disconnect the ELC plug between the motor control card and the heater control box servo motor.
6. Cut the plastic tie straps located at the air control unit inlet and outlet tube locations.
7. Pull the flexible ducting away from the air control unit.
8. Remove the four 6-32 self-locking nuts and washers attached to the Pem studs on the face of the air control unit. See Figure 21 - 10.

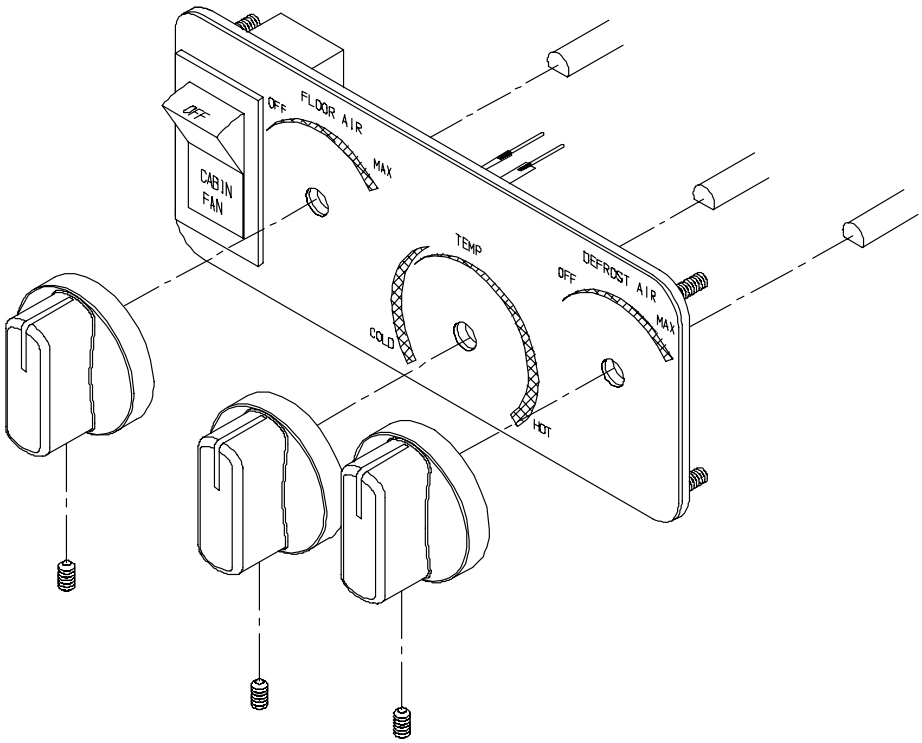


Figure 21 - 9 ECS Faceplate

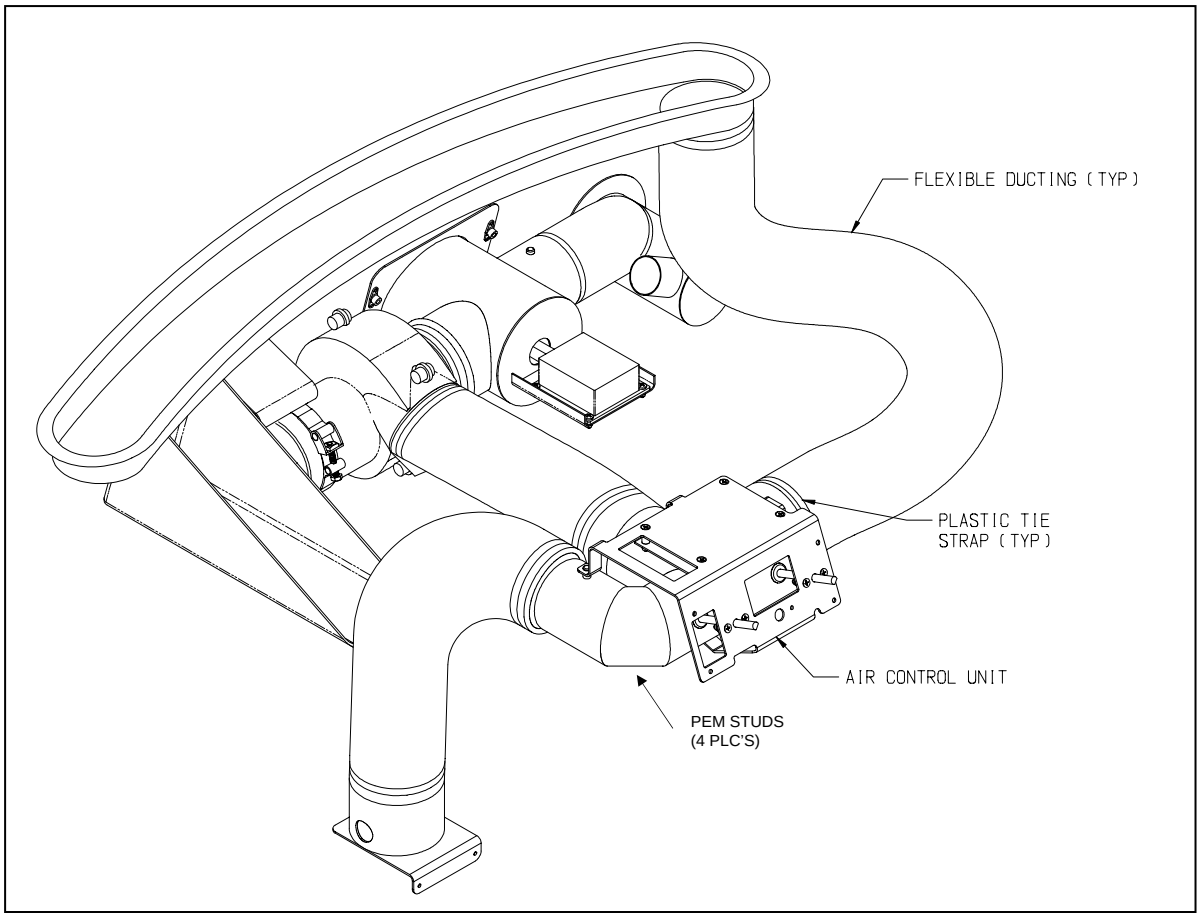


Figure 21 - 10 Air Control Unit Attachment

9. From the back side of the instrument panel, remove the nuts and washers securing the air control unit tray to the back side of the instrument panel as shown in Figure 21 - 10 and Figure 21 - 11.
10. Pull the air control unit forward (toward the firewall) and away from the instrument panel until the knob shafts are behind the instrument panel.
11. Remove the air control unit by pulling it out from underneath the instrument panel.

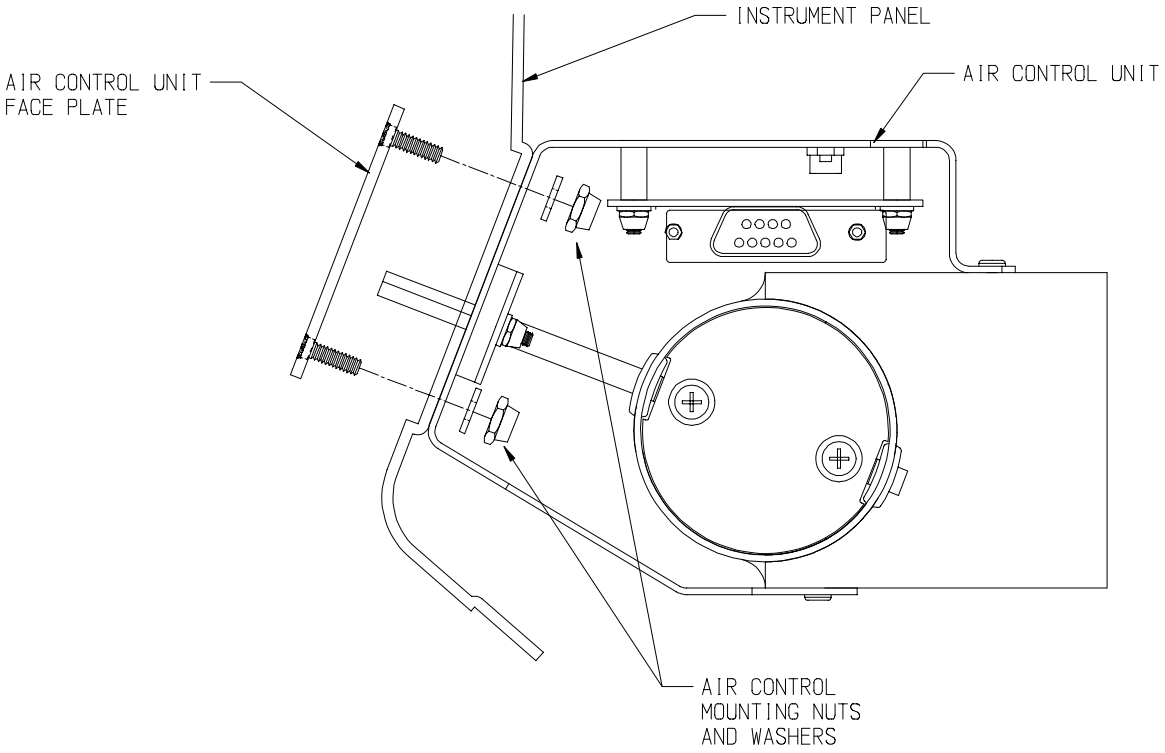


Figure 21 - 11 Air Control Unit Attachment (S/N 42003-42023)

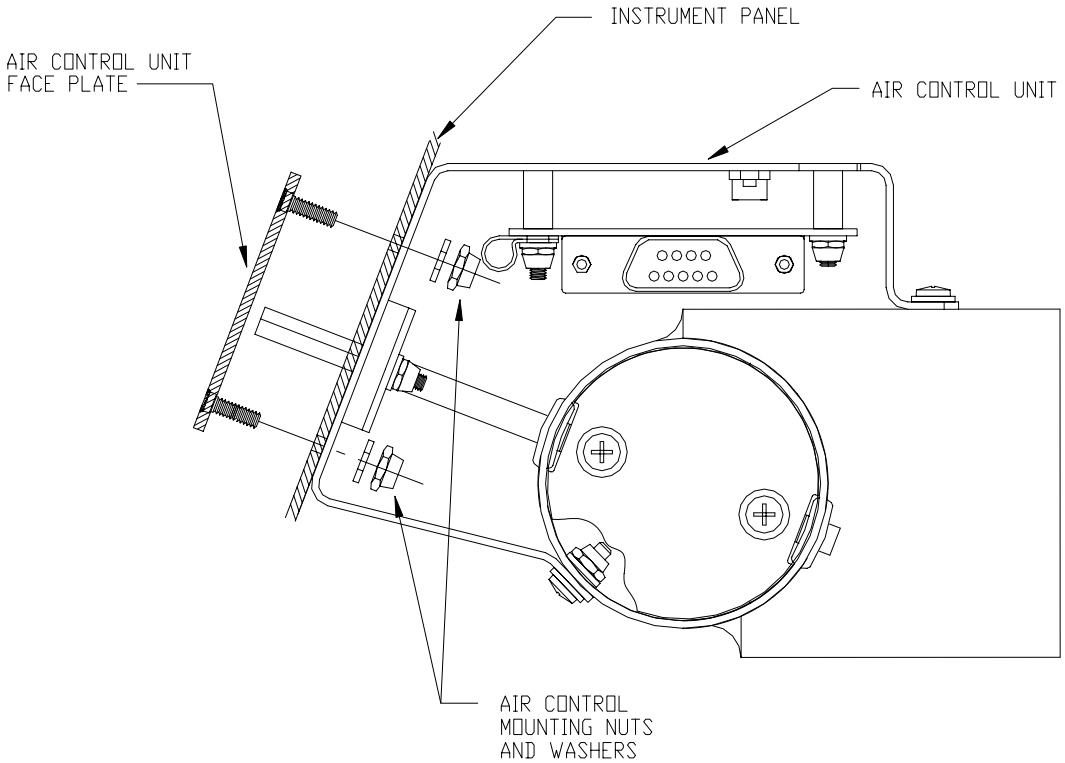


Figure 21 - 12 Air Control Unit Attachment (S/N 42024 and on)

b. **Installation of Air Control Unit** – To install the air control unit, use the following instructions.
   1. Place the unit onto the cover panel studs from the backside of the instrument panel, and secure with nuts and washers.
   2. Slide the flexible ducting onto the air control unit inlet and outlet tubes, and secure using plastic tie straps.
   3. Reconnect the motor control card to the servomotor ELC connector.
   4. Reconnect wiring to the backside of the fan rocker switch, and gently press the switch into the opening until the switch locking tabs are fully engaged.
   5. Install the temperature and air control knobs to the control shafts, checking for free knob rotation. Install set screws at bottom of knobs as shown in Figure 21 - 9.
   6. Turn the power on, and check for proper fan and heater control valve operation.

c. **Maintenance** – The temperature and air control unit is self-contained. A periodic check for fan, air control unit butterfly valve, and heater control valve operation is required. A functional check should be completed at each 100 hour or annual inspection using the following procedure. See 21-1 for description on control operation.
   1. With cabin fan off, check adequacy of airflow for the front floor vents, rear floor vent, and defroster.
   2. Check for proper operation of the air volume control, temperature control, and air output control.
   3. Evaluate eyeball fresh air vents for proper operation.
   4. Evaluate the adequacy of the system in terms of crew and passenger comfort, as well as the defroster's ability to control fogging of the windscreen.

21-15. AUTOMATIC CLIMATE CONTROL SYSTEM (ACCS)

a. **System Description** – The Automatic Climate Control System (ACCS) is a fully automatic system with manual operation capabilities comprised of a Control Head, Electronic Control Unit (ECU), temperature sensors, a Vapor Cycle System (VCS) air conditioning system, valves, hoses, and tubing. The air conditioning system consists of a compressor, a condenser with fans, a receiver-dryer with trinary pressure switch and an evaporator with an expansion valve and evaporator coil temperature sensor. Barrier hoses connect the components. A combination of steel and aluminum fittings are used. See Figure 21 - 13.

In manual or automatic operation, the system offers 11 choices for heater and air conditioning blower speed.

On 28 VDC aircraft with ACCS equipped with the optional electric compressor, pre-cooling of the aircraft cabin is possible using a ground power supply.

NOTE

Ground power operation of the air conditioning will require a ground power source that can deliver 100 amps during use.

After connecting a ground power source and switching the unit on, the ACCS can be activated by pushing the external switch found near the ground power receptacle. When activated, the aircraft power grid is disabled and the electric compressor and evaporator blower will run continuously while the condenser fans automatically cycle as needed. The external ACCS switch does not function when a battery master is on. The ACCS control panel is disabled during external power air conditioning operation and the ACCS cools at max capability. External power ACCS operation can be deactivated by pushing the external switch, removing the ground power source, or turning on either battery master switch. With a battery master on, the ACCS will be fully functional except the electric compressor will be off. An Electric A/C Interlock Assembly allows this integration of the ACCS with the airplanes electrical system. Refer to Chapter 24 for removal and installation of the interlock assembly.

The ACCS has self-diagnostic capabilities which can be displayed on the screen of the Control Head. See Figure 21 - 15.

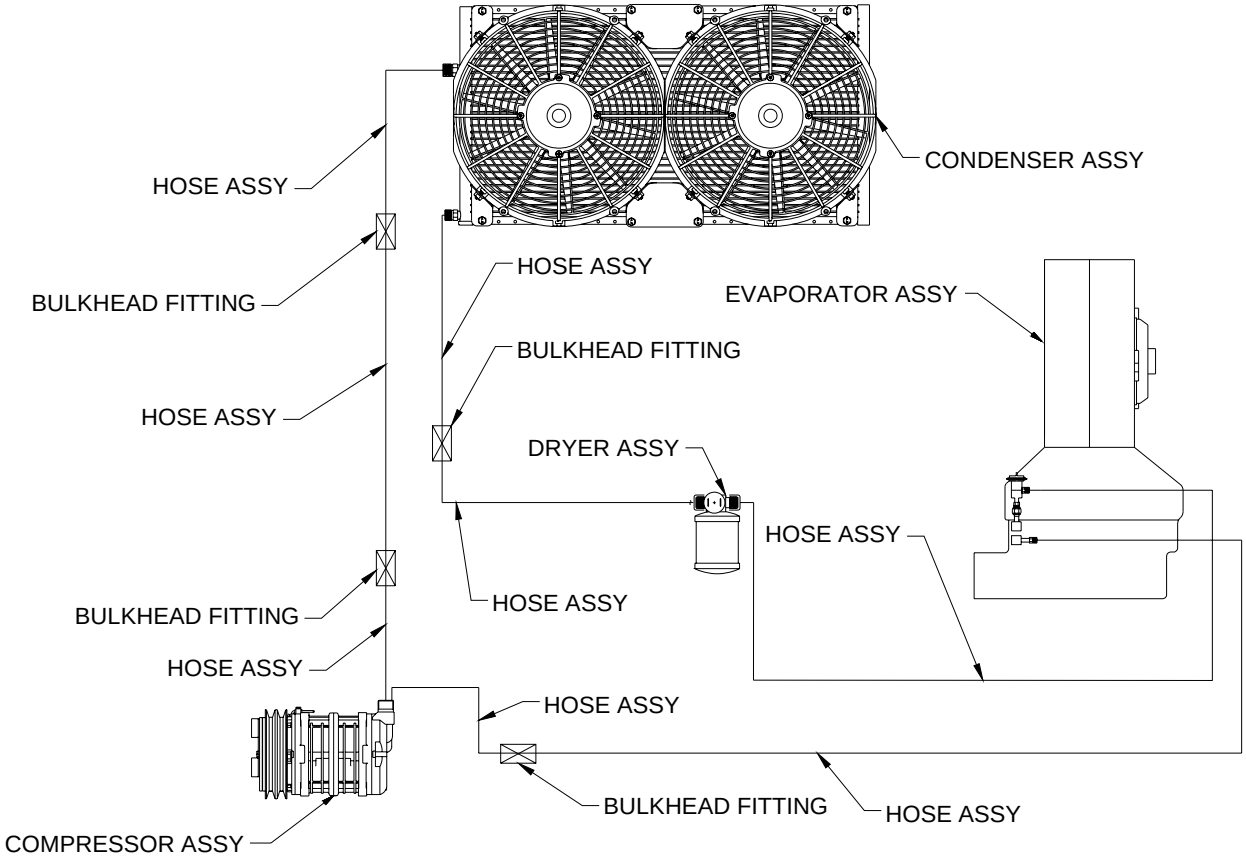


Figure 21 - 13 Vapor Cycle System (VCS) Air Conditioning

b. **Air Distribution –** Air distribution is controlled by the ACCS with pilot, or co-pilot, input through the control head. See Figure 21 - 14.

1. Air Conditioning – When air conditioned air is required, the evaporator blower is actuated and air is distributed through the overhead ceiling console vents and associated flood ducts.
2. Heat – When heat is required, the ACCS shuts off the compressor and evaporator blower and regulates the heater/defog blower in the forward cabin. Heated air is distributed approximately 90% to the floor ducts with the remaining air being diverted to the windscreen defog ducts.
3. Defog – When the Defog " " mode, on the Control Head, is selected the heater/defog blower is activated at maximum speed and all air is distributed to the windscreen. Temperature selection on the Control Head regulates the temperature of the defog air.

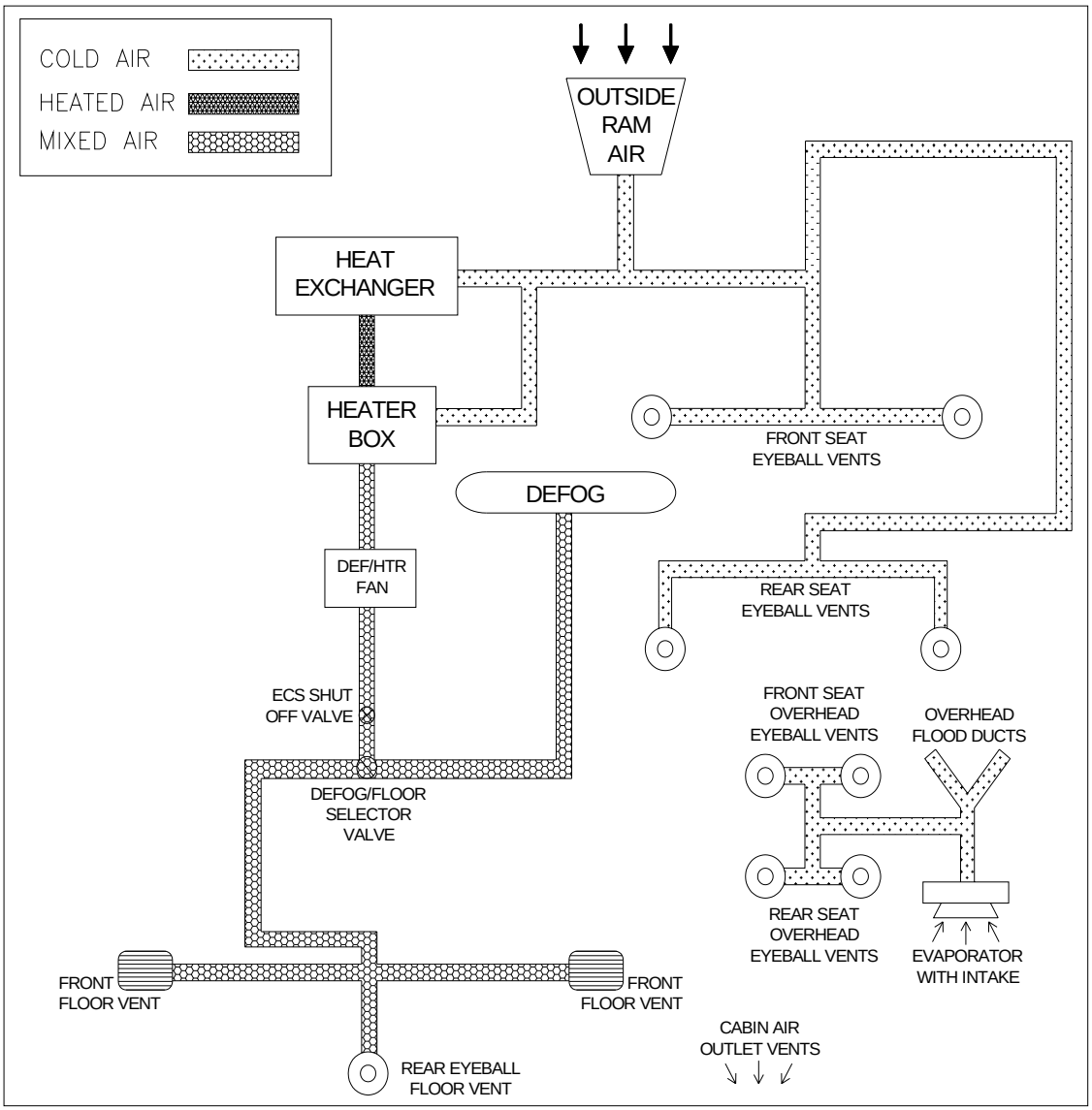


Figure 21 - 14 Airflow Diagram

c. **Diagnostics –** To display the Diagnostics Fault Code display, turn on the ACCS (or ECS on Garmin G1000 equipped aircraft)and depress the ON button three times in succession. The digital display on the Control Head will display any active fault codes. Repeatedly pressing the ON button will scroll through the active codes. Each code displayed should be addressed in turn until the "No Faults" code E00 is displayed. See Figure 21 - 15.

With the Garmin G1000 option and no ACCS, fault codes E04 and E08 will always display due to the absence of an evaporator temperature sensor or OAT sensor in the system. In this case, codes E04 and E08 may be ignored.

Fault Code	Information	Correction
E00	No Faults	-
E01	Cabin Temp Sensor-Shorted	Check cabin temp sensor wiring and sensor for correct operation. See Figure 21 - 16.
E02	Cabin Temp Sensor-Open	Check cabin temp sensor wiring and sensor for correct operation. See Figure 21 - 16.
E03	Evap Temp Sensor-Shorted	Check evaporator coil temp sensor wiring and sensor for correct operation. See Figure 21 - 18.
E04	Evap Temp Sensor-Open	Check evaporator coil temp sensor wiring and sensor for correct operation. See Figure 21 - 18.
E05	Not Currently Used	-
E06	Not Currently Used	-
E07	OAT Sensor-Shorted	Check OAT sensor wiring and sensor for correct operation. See Figure 21 - 17.
E08	OAT Sensor-Open	Check OAT sensor wiring and sensor for correct operation. See Figure 21 - 17.
E09	Not Currently Use	-
E10	Heater Blend Air Actuator-Open	Check actuator sensor wiring and sensor for correct operation.
E17	No Communication with ECU	Check ECU wiring and system circuit breakers for correct operation.

Figure 21 - 15 Fault Display Codes

1. **Advanced Diagnostics –** The Advanced Diagnostics display mode is activated by pressing and holding the "ON" button for five seconds while the Fault Code display is active.

   Press the "ON" button to cycle through the following displays:
   "CAb" Cabin Temperature Sensor temperature reading
   "EUAP" Evaporator Coil Temperature Sensor temperature reading
   "HEAt" ECS mixing valve position (percentage open)
   "A-C" VCS compressor command (ON or OFF)
   "FAn" Blower Speed-Evaporator or Heater/Defog (percentage of max)
   Press the "OFF" button to return to the normal display mode.

2. **Firmware Version –** The Firmware Version display mode is activated by pressing and holding the "ON" button for five seconds while the Advanced Diagnostics display is active.

   Press the "ON" button to cycle through the following displays:
   "ECU" ECU Module Firmware Version
   "CtrL" Control Head Firmware Version
   Press the "OFF" button to return to the normal display mode.

3. **Parameter Function** – The parameter function is activated by pressing and holding the "OFF" button for five seconds while the advanced diagnostics display mode is active.

   Press the FAN "+" or the FAN "-" buttons to select configuration (config)setting: "Pf-x" ("x" may be displayed as 0,1, or 2 depending on setting).

   Press the TEMP "+" or the TEMP "-" buttons to change the selected config setting to : "Pf-1".

   Exit and save the settings in the programming mode by pressing the "OFF" button.

NOTE

DO NOT CHANGE ANY OTHER SETTINGS within the configuration screen or damage to the system may occur.

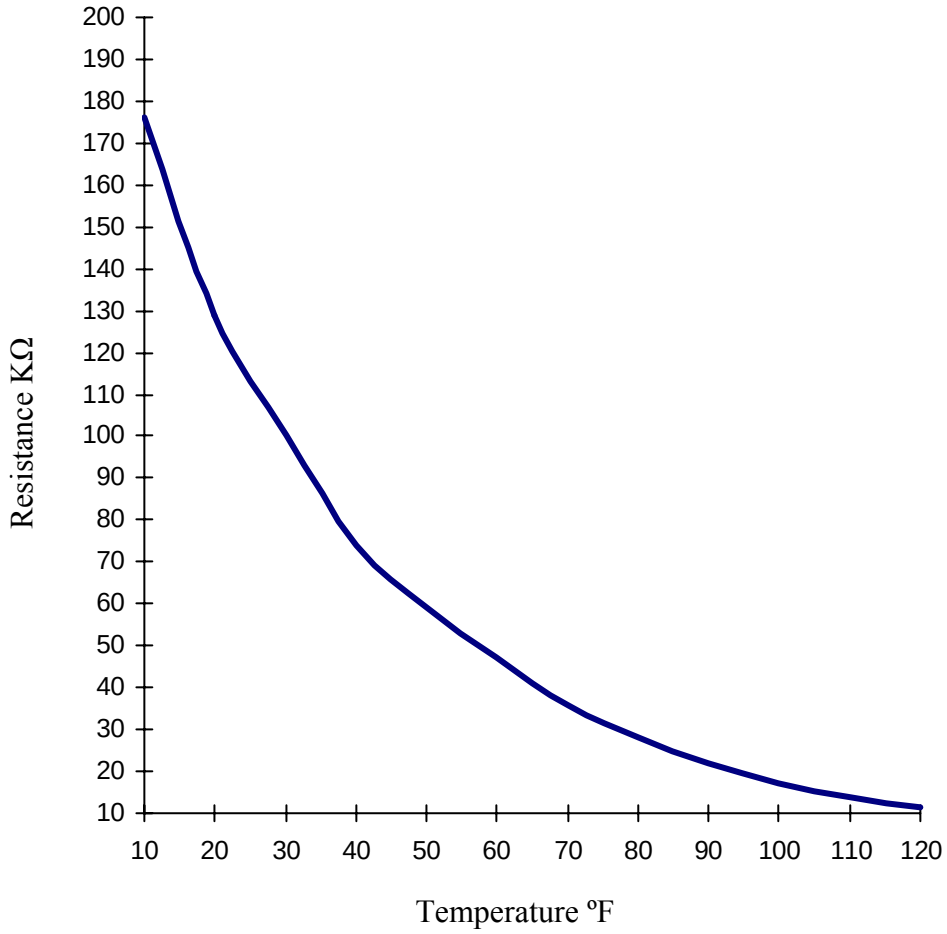


Figure 21 - 16 Cabin Temperature Sensor Resistance vs. Temperature Curve

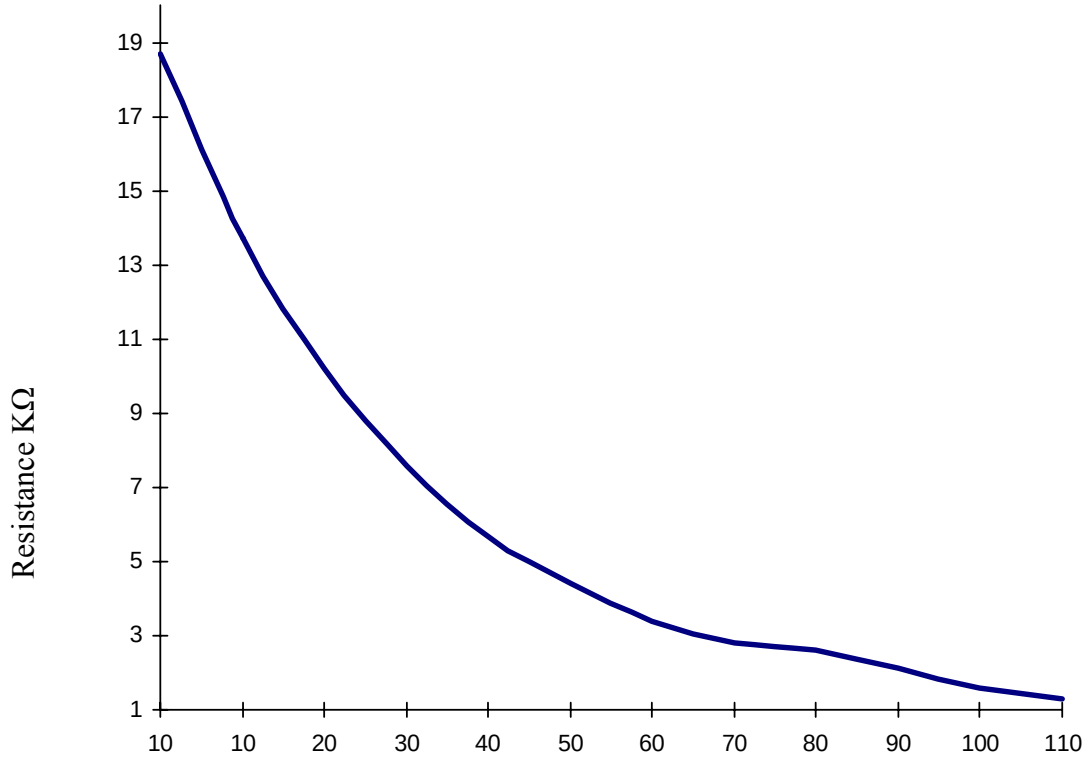


Figure 21 - 17 OAT Sensor Resistance vs. Temperature Curve

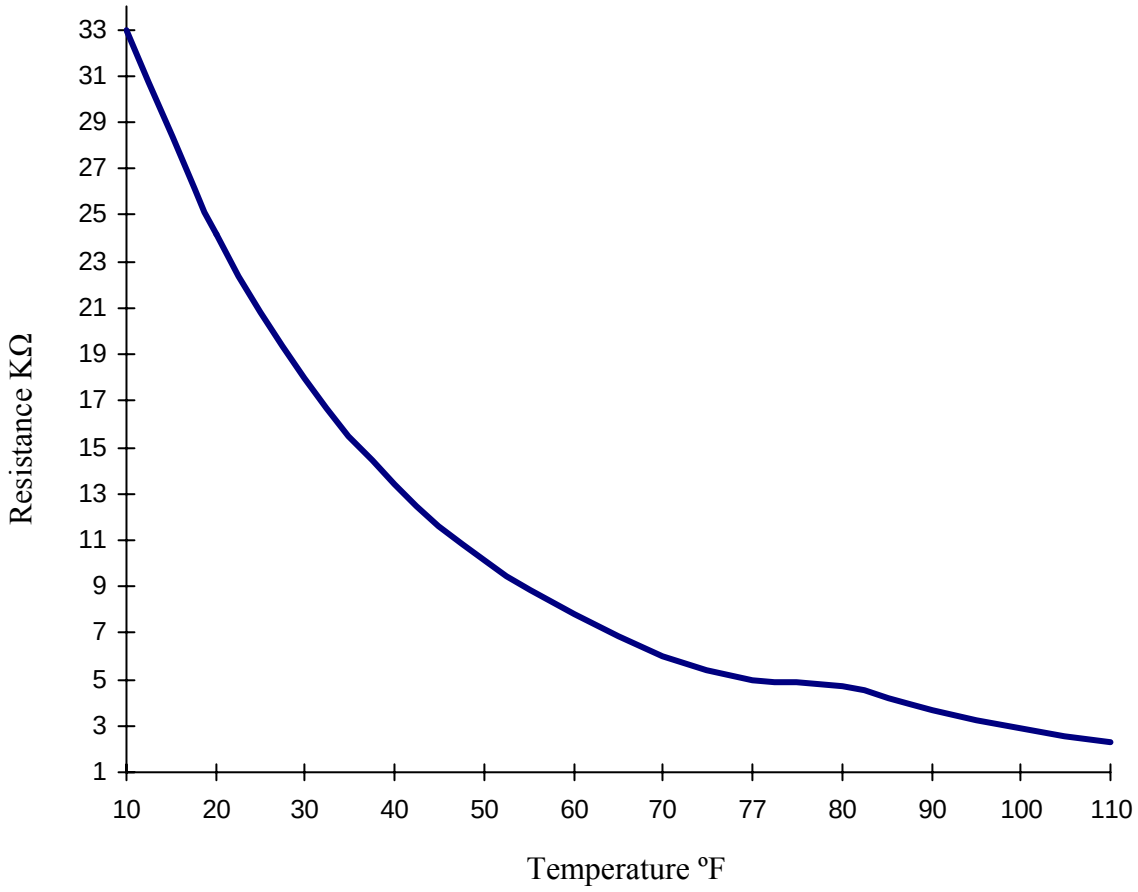


Figure 21 - 18 Evaporator Coil Temperature Sensor Resistance vs. Temperature Curve

d. **Safety and System Service Precautions**

1. **Cabin Sealing –** All of the following items must be sealed to ensure that carbon monoxide does not enter the cabin of the aircraft. Use McMaster-Carr foam tape P/N 8512K24 or LP Aero Plastic foam tape P/N SP-FT114, 3M 425 foil tape, and PR 1428 B1/2 or RTV5818 sealant as indicated.
   a) Two clear access panels, or one plug, located on the aft condenser bulkhead – Ensure that foam tape is intact around the edges of the panels. When reinstalling the panels apply foam tape as required to form an airtight seal. When reinstalling the plug, apply Wacker Elastosil 952, Dow Corning 732, or GE RTV-108 sealant.
   b) Baggage floor at the aft condenser attachment – Sealed with .25" +/- bead of sealant. Apply sealant as required.
   c) Components penetrating the forward condenser bay bulkhead, and hollow shank rivets in the condenser compartment – Apply sealant as required.
   d) Lower corners of condenser compartment and gear member seals – Apply sealant as required

e) Between gearbox and fuselage on exterior near forward side of gear leg attachment – Apply sealant as required.

f) Air conditioning system bay perimeter– Ensure foam tape is intact and installed on both sides of the attachment holes. Apply foam tape as required.

g) Air conditioning system bay access cover – Seal the air conditioning system bay access cover to the floor using foil tape to create a leak proof seal.

2. **Warnings and Cautions for R-134a Refrigerant**

a) Avoid breathing R-134a refrigerant, lubricant vapor, or mist. Exposure may irritate eyes, nose, and throat. Wear eye protection when servicing the air conditioning system. Serious eye injury can result from eye contact with refrigerant. If eye contact is made, seek medical attention immediately.

b) If accidental system discharge occurs, ventilate the work area before resuming service. Large amounts of R-134a refrigerant will displace oxygen and cause suffocation. Work only in well ventilated areas.

c) Do not heat refrigerant containers above 125ºF or expose refrigerant to open flame. Do not use open flame to heat refrigerant containers.

d) Do not intentionally drop, puncture, or incinerate refrigerant containers.

e) The evaporation rate of R-134a refrigerant at average temperature and altitude is extremely high. As a result, anything that comes in contact with the refrigerant will freeze. Always protect skin or delicate objects from direct contact with refrigerant. For personal protection, goggles and protective gloves should be worn and clean cloth wrapped around fittings, valves and connections when doing work that includes opening the refrigerant system. If R-134a refrigerant comes in contact with any part of the body, severe frostbite and personal injury can result. Flush exposed zone immediately with cold water and obtain prompt medical assistance.

f) R-134a service equipment or aircraft air conditioning system should not be pressure tested or leak tested with compressed shop air. Though R-134a is considered non-flammable, some mixtures of air and R-134a have been shown to be combustible at elevated pressures. These mixtures are potentially dangerous and may result in fire or explosion.

g) Never add R-134a refrigerant to a system that has not been evacuated to 29 in. Hg. vacuum pressure.

CAUTION

Liquid refrigerant is corrosive to metal surfaces. Follow the operation instructions supplied with equipment being used.

CAUTION

Never add R-12 refrigerant to an air conditioning system designed to use R-134a. Damage to the system may result.

CAUTION

R-12 compressor oil cannot be mixed with the R-134a compressor oil. They are NOT compatible

CAUTION

Do NOT use R-12 servicing equipment or parts on an R-134a air conditioning system.

3. **System Service Precautions**
   a) Never open or loosen a connection before discharging the system refrigerant.
   b) Do NOT open a refrigerant system or uncap a replacement component unless its temperature is as close as possible to room temperature. This will prevent condensation from forming inside of a component which is cooler than the surrounding air.
   c) Before disconnecting a component from the system, clean the outside of the fittings thoroughly.
   d) Immediately after disconnecting a component from the system, seal the open fittings with a cap or plug.
   e) Before connecting an open fitting always install a new seal or gasket designed for the specific component. Coat the fitting and seal with clean refrigerant oil before connecting.
   f) Do NOT remove the sealing caps from a replacement component until ready to install.
   g) When installing a refrigerant line avoid bends which produce radiuses smaller than specified below for the hose being installed.

Hose Size	Minimum Bend Radius
1/2” 13mm (#10)	3.0 inch
13/32” 10mm (#8)	2.5 inch
5/16” 8mm (#6)	2.0 inch

   h) Position refrigerant lines to avoid exhaust components, flight controls, or items which may chafe the line. Refer to A.C. 43-13, Acceptable Methods, Techniques, and Practices for compliant methods of installation.
   i) Apply a light coat of Ester RL-500S (POE) (engine driven compressor) or FVC68D (PVE) (electric compressor) refrigerant oil to the o-rings of all fittings. Tighten fittings only to the specified torque listed below. Use an anti-torque back-up wrench on ALL component fittings. The refrigerant fittings will not tolerate over-tightening.

Tube O.D.	Thread Size	Torque Ft-lbs.
3/8" (.375")	5/8-18	11-13
1/2" (.500")	3/4-16	15-20
5/8" (.625")	7/8-14	21-27

Trinary Switch Torque: 7 ft.-lbs.

j) When disconnecting a fitting use a wrench on BOTH halves of the fitting. This will prevent twisting of the refrigerant lines or tubes.

k) Refrigerant oil will absorb moisture from the atmosphere if left uncapped. Do NOT open an oil container until ready to use and install the cap immediately after using. Store the oil only in a clean moisture-free container.

l) Keep service tools and the work area clean. Contamination of the VCS air conditioning system through careless work habits must be avoided.

m) Plumbing systems in aircraft are subject to vibration and extreme temperature and atmospheric pressure changes. Therefore, strict adherence to procedures and correct use of material is mandatory.

e. **System Components**

1. Refrigerant Oil – There is either POE or PVE refrigerant oil used in the air conditioning system depending upon the type of compressor installed on the airplane. POE refrigerant oil is used with the engine driven compressor and PVE refrigerant oil is used with the electrically driven compressor. It is important to have the correct amount of oil in the air conditioning system. This will ensure proper lubrication of the compressor. Too little oil will result in damage to the compressor. Too much oil will reduce the cooling capacity of the system.

CAUTION

Do not mix POE and PVE refrigerant oils. Verify the type of oil used in the refrigerant system before starting any work on the system. If the type of oil used is in doubt, contact Cessna.

a) The oil used in the system is either Ester RL-500S (POE) which is wax-free synthetic ester refrigerant oil or FVC68D (PVE) which is a polyvinylether refrigerant oil. Do NOT use any other oil when servicing or repairing the system. The oil container should be kept tightly capped until it is ready for use and then capped immediately after use to prevent contamination. Refrigerant oil will quickly absorb any moisture it comes in contact with.

b) When an air conditioning system is assembled at the factory all components except the compressor are refrigerant oil free. After the system has been charged and operated, the oil in the compressor is dispersed through the system. The evaporator, condenser, receiver/dryer, and compressor will retain a significant amount of oil.

c) When a receiver/dryer, condenser, or evaporator core is replaced, 1 fluid oz. (29 cc) of refrigerant oil must be added.

d) When the compressor is replaced, the new compressor is factory filled with the correct amount of oil. It will not be necessary to check the oil level in the compressor or to add oil unless there has been an oil loss. This may be due to a rupture or leak from a line, shaft seal, evaporator, or condenser. Oil loss at a leak point will be evident by the presence of a wet shiny surface around the leak.

2. VCS Compressor – The air conditioning compressor will be either engine driven or electrical, depending upon the configuration of the aircraft.

   a) Engine Driven Compressor – The compressor is a swash or wobble plate axial compressor with a displacement of 147 cm^3. The compressor cycles to regulate cabin temperature and to avoid evaporator coil freeze-up. The compressor is engaged by a magnetic clutch driven by a V-Belt off the accessory drive at the back of the engine. The magnetic clutch is switched off during air conditioning operation anytime the refrigerant system pressure drops below 29 psi (2 bar) at the trinary switch or anytime the high-pressure side of the refrigerant system rises above 397 psi (27 bar). See Figure 21 - 19 and Figure 21 - 20.

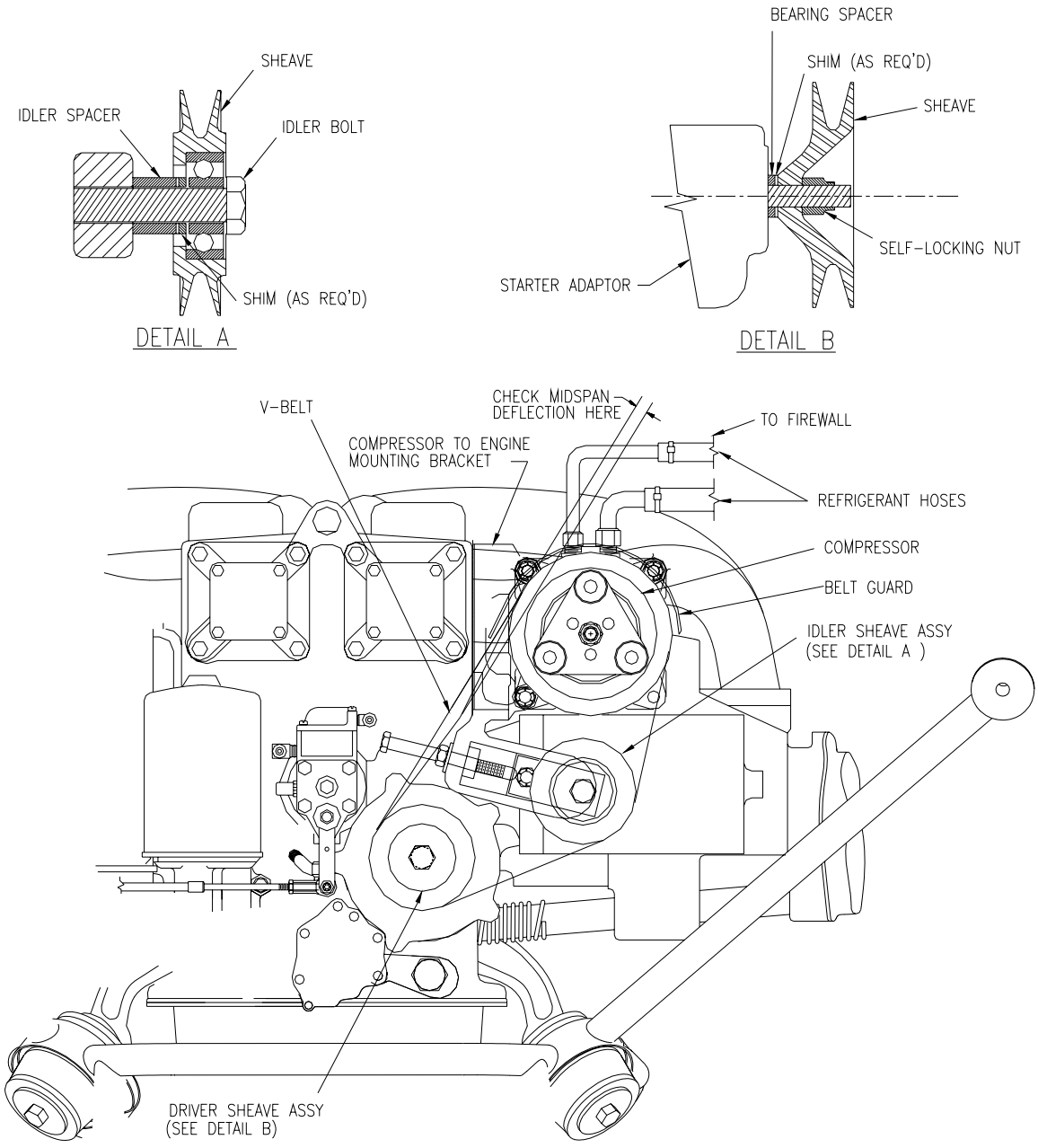


Figure 21 - 19 Engine Driven Compressor, Engine Rear View

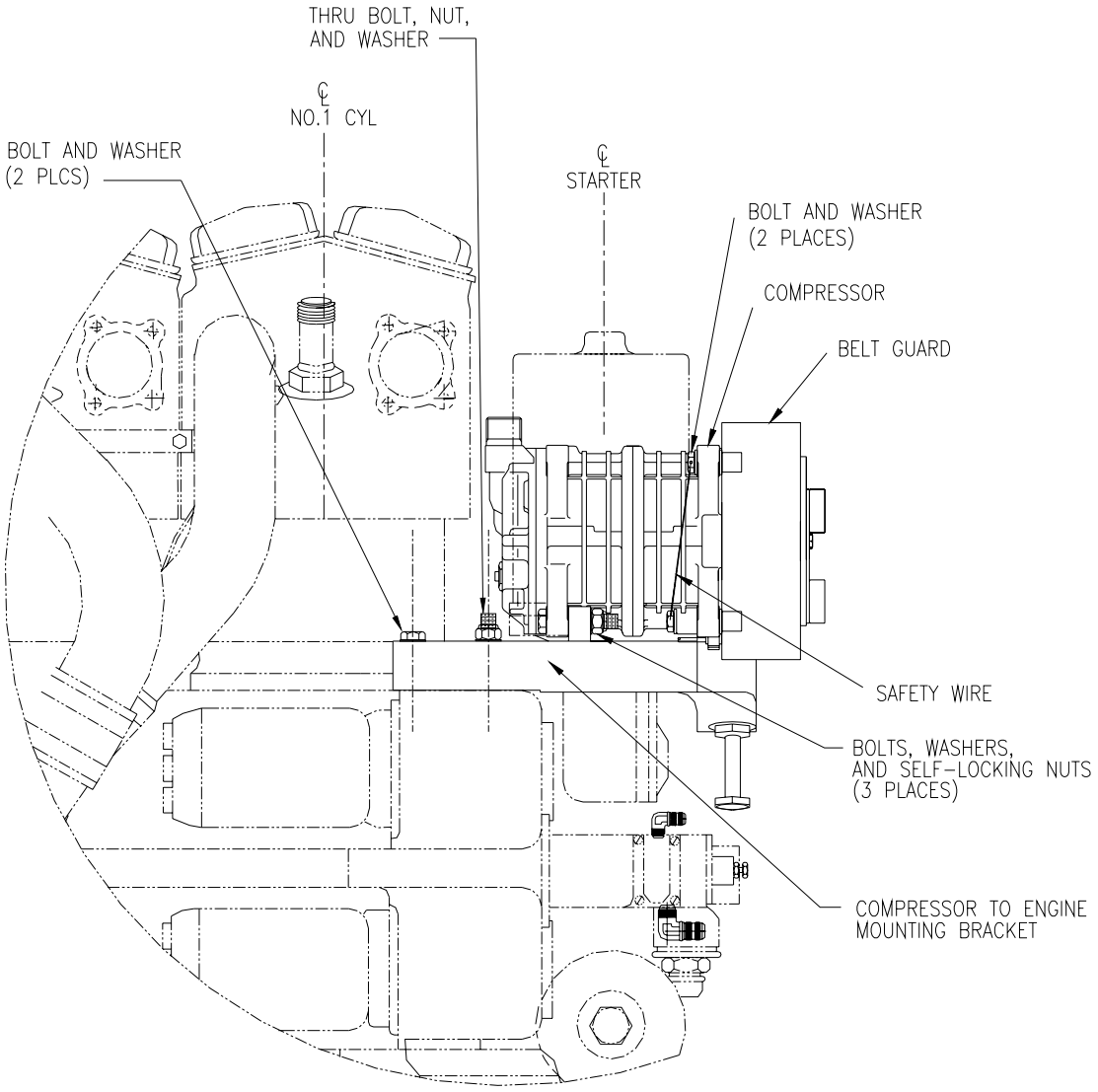


Figure 21 - 20 Engine Driven Compressor, Engine Top View

(i) Removal– To remove the compressor use the following instructions.

1. Remove the upper and lower engine cowling per the instructions in Chapter 71.
2. Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.
3. Remove the Evaporator/Blower Assembly cover. See Figure 21 - 24.
4. Evacuate the air conditioning system per instructions in paragraph 21-15.h.
5. Remove the belt guard.
6. Loosen the V-belt idler sheave and remove the V-belt.

7. Disconnect the two refrigerant hoses from the compressor and seal all open fittings with a cap or plug.
8. Remove 3 AN6-16A bolts, washers, and self-locking nuts attaching the compressor to the engine.
9. Remove the compressor.

(ii) Installation– To install the compressor use the following instructions.

1. Attach the compressor to the engine using 3 AN6-16A bolts, washers, and self-locking nuts. Torque 20 to 25 ft.-lbs.
2. Place the V-belt onto the compressor pulley.
3. Install the belt guard using AN6H-10A and AN6H-14A bolts, and washers. Torque 275 to 325 in.-lbs. Safety wire the bolts.
4. Tighten the V-belt idler sheave to provide the belt tension recommended by Teledyne Continental Motors service procedures.
5. Connect the two refrigerant hoses to the compressor.
6. Replace the receiver/dryer if required per 21-15.e.6.
7. Evacuate and charge the system per paragraphs 21-15.h. and 21-15.i.
8. Replace the Evaporator/Blower Assembly cover.
9. Replace the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

(iii)Maintenance – The compressor is self-contained and maintenance is not required. Inspection of the drive belts and pulleys should be completed at each 100 hour or annual inspection.

1. Inspect the compressor drive belt for chafing and/or wear. The belt must be replaced if there is any indication of fraying or dryrot. Check the belt tension and adjust per Teledyne Continental Motors recommended service procedures as required.

CAUTION

Do NOT over-tighten the belt. After approximately 5 hours of initial operation, recheck the belt tension and adjust as required .

2. Inspect the compressor drive belt pulleys for security, tightness, rotational smoothness, and freedom.

b) Electrical Compressor – The electric compressor is a brushless, hermetic DC variable speed compressor. The compressor operates at a fixed high speed determined by an electronic motor controller which cycles the compressor between high speed and idle speed depending on requirements of the system. Electric power to run the compressor is provided by an accessories alternator mounted on the aft of the engine. The compressor is protected by the trinary switch which senses out of limit system pressures. Any time the system pressure drops below 29 psi or rises above 397 psi, the compressor is shut down. See Figure 21 - 21.

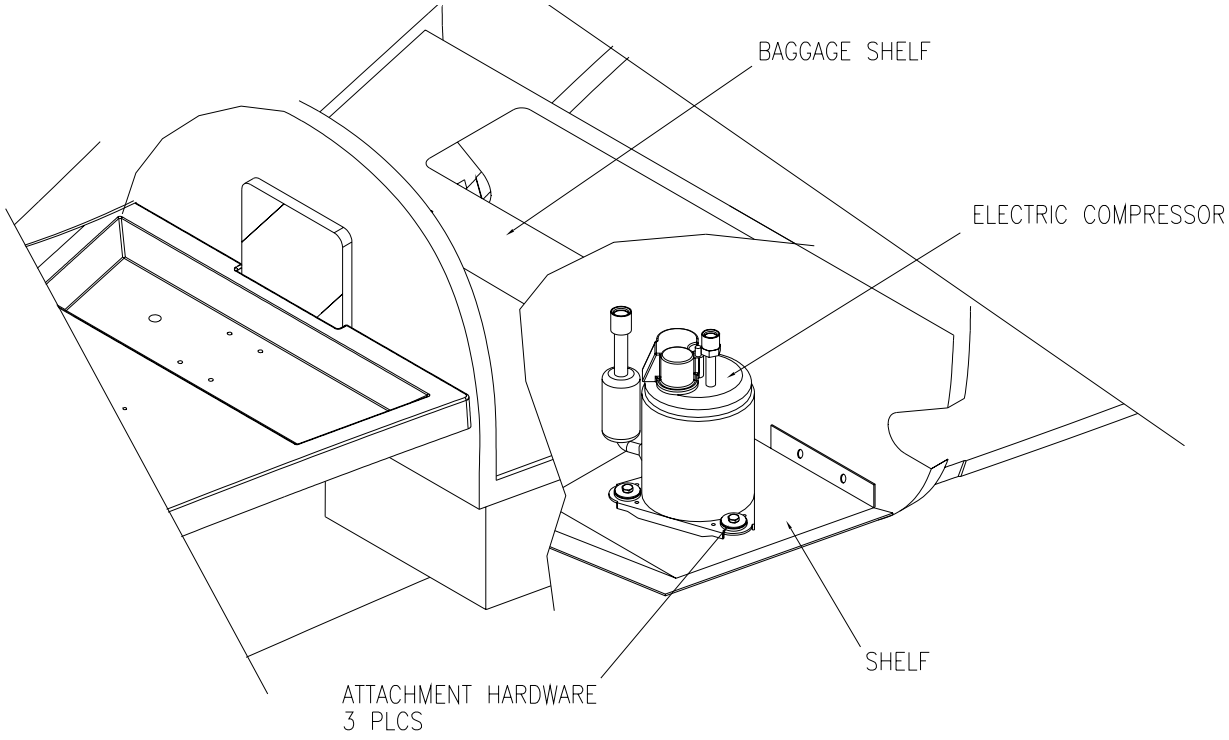


Figure 21 - 21 Electrical Compressor

(i) Removal

1. Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.
2. Remove the Elevator Interconnect Access Panel per instructions in Chapter 25.
3. Remove the Evaporator/Blower Assembly cover. See Figure 21 - 24.
4. Evacuate the air conditioning system per instructions in paragraph 21-15.h.
5. Disconnect all electrical connections to the compressor.
6. Disconnect the two refrigerant hoses from the compressor and seal all open fittings with a cap or plug.
7. At each of three locations remove the AN4-10 bolt, washers, isolation ring, isolation bushing, and bolt sleeve spacer attaching the compressor to the shelf.
8. Remove the compressor.

(ii) Installation

1. Ensure that the wet weight of the compressor is 14.68 lbs. when installed in a dry system. Add refrigerant oil as required.

2. At each of three locations attach the compressor to the shelf using the AN4-10 bolt, washers, isolation ring, isolation bushing, and bolt sleeve spacer.
3. Connect the two refrigerant hoses to the compressor.
4. Connect all electrical connections to the compressor.
5. Replace the receiver/dryer if required per 21-15.e.6.
6. Evacuate and charge the system per paragraphs 21-15.h. and 21-15.i.
7. Replace the Evaporator/Blower Assembly cover.
8. Replace the Elevator Interconnect Access Panel per instructions in Chapter 25.
9. Replace the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

(iii) Maintenance – The VCS Compressor is self-contained and maintenance is not required.

3. Condenser/Fan Assembly – The condenser receives high-temperature, high-pressure vapor from the compressor. The vapor enters at the upper fitting of the condenser. As the vapor flows through the condenser, it releases heat to the cooler ambient air flowing over the condenser. As the vapor releases heat it changes to a liquid. Under average load, 2/3 of the condenser contains refrigerant vapor and the other 1/3 contains liquid refrigerant. The liquid refrigerant in the condenser has lost much of its heat, but still remains at a relatively high temperature and pressure. The assembly is located beneath the baggage compartment floor with vents for inlet and outlet directly below, through the belly of the aircraft. See Figure 21 - 22.

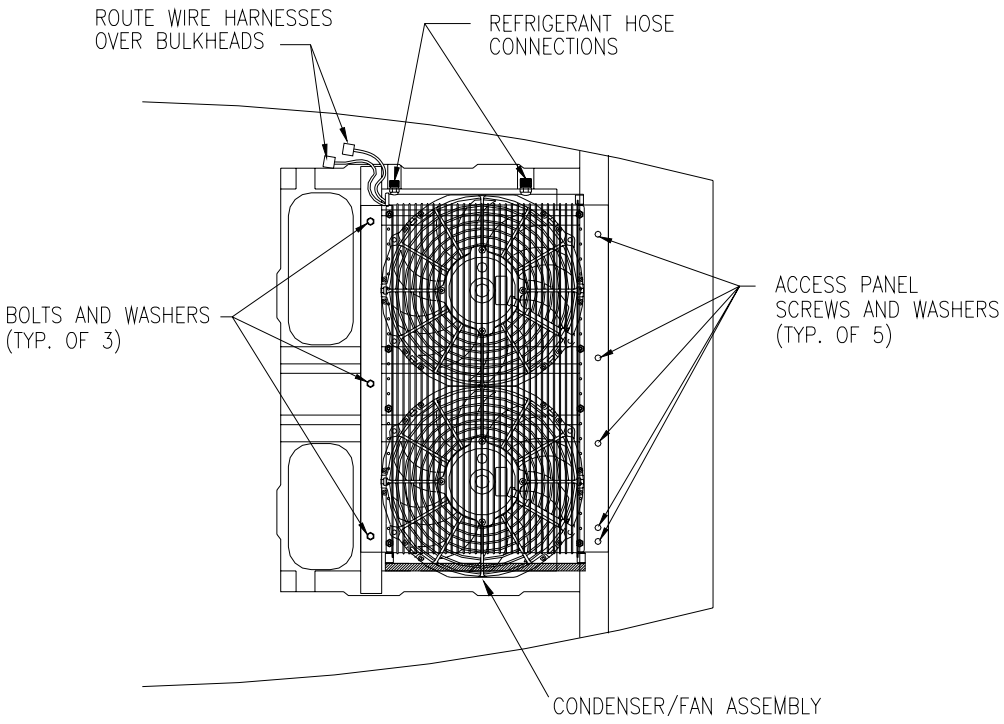


Figure 21 - 22 Condenser/Fan Assembly

a) Removal of the Condenser/Fan Assembly – To remove the Condenser/Fan Assembly use the following instructions.
   1) Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.
   2) Remove the Evaporator/Blower Assembly cover, the avionics shelf access panel and the air conditioning system bay access cover. See Figure 21 - 24
   3) Evacuate the air conditioning system per instructions in paragraph 21-15.h.
   4) Disconnect the wiring harness.
   5) Disconnect the two refrigerant hoses from the Condenser/Fan assembly and seal all open fittings with a cap or plug.
   6) Remove 5 screws and 3 bolts mounting the Condenser/Fan assembly to the fuselage bulkheads.
   7) Lift the Condenser/Fan assembly out of the compartment.

b) Installation of the Condenser/Fan Assembly – To install the Condenser/Fan Assembly use the following instructions.
   1) Install PR1428 B1/2 or RTV5818 sealant, as required, at all openings between the air conditioning bay and the cabin area (including underlying floors) to completely seal off the cabin area from the air conditioning bay. Installation is the reverse of steps (iv) through (vii) above.
   2) Install McMaster-Carr foam tape P/N 8512K24, as required, around the perimeter of the air conditioning system bay.
   3) Installation of the Condenser/Fan Assembly is the reverse of removal steps 4) through 7) of paragraph a) above. Install a .25" +/- bead of PR1428 B1/2 or RTV5818 sealant between the baggage floor and the aft condenser attachment.
   4) Replace the receiver/dryer if required per 21-15.e.6.
   5) Evacuate and charge the system per paragraphs 21-15.h. and 21-15.i. Add 1 fluid oz. (29 cc) of refrigerant oil when installing a new Condenser/Fan assembly.
   6) Replace the air conditioning system bay access cover. Seal the air conditioning system bay access cover to the floor using 3M 425 adhesive foil tape to create a leak proof seal.
   7) Replace the avionics shelf access panel, and the Evaporator/Blower Assembly cover.
   8) Replace the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

c) Maintenance – The Condenser/Fan Assembly is a self-contained unit and maintenance is not required.

4. Expansion Valve – The expansion valve is located at the inlet to the evaporator. It is the connection point between the high-pressure and low-pressure side of the refrigeration system. The expansion valve determines the flow rate of the refrigerant and provides a pressure drop, reducing pressure on the liquid refrigerant before it enters the evaporator. See Figure 21 - 23 and Figure 21 - 25.

Figure 21 - 23 Expansion Valve

a) Removal of the Expansion Valve – To remove the Expansion Valve use the following instructions.
   1) Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.
   2) Remove the Evaporator/Blower Assembly cover. See Figure 21 - 24.
   3) Evacuate the air conditioning system per instructions in paragraph 21-15.h.
   4) Disconnect the refrigerant hose from the Expansion Valve and the Expansion Valve from the Evaporator/Blower Assembly, and seal all open fittings with a cap or plug.

b) Installation of the Expansion Valve – To install the Expansion Valve use the following instructions.
   1) Connect the Expansion Valve to the Evaporator/Blower Assembly and connect the refrigerant hose to the Expansion Valve. Torque fittings per the values in paragraph 21-15.d.3.i).
   2) Replace the receiver/dryer if required per 21-15.e.6.
   3) Evacuate and charge the system per paragraphs 21-15.h. and 21-15.i.
   4) Replace the Evaporator/Blower Assembly cover.
   5) Replace the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

c) Maintenance – The Expansion Valve is a self-contained unit and maintenance is not required.

5. Evaporator/Blower Assembly – The evaporator uses a plate and fin evaporator coil. The tubing is hydraulically expanded into the fins to ensure good thermal conductivity. The evaporator removes heat and moisture from air entering the aircraft cabin. Low-pressure liquid refrigerant from the expansion valve enters the evaporator. Due to the reduced pressure, the refrigerant boils into a vapor. The boiling refrigerant absorbs the heat from the incoming air as it changes from a liquid to a vapor. The unit is located under a cover against the aft wall of the baggage compartment. See Figure 21 - 25.

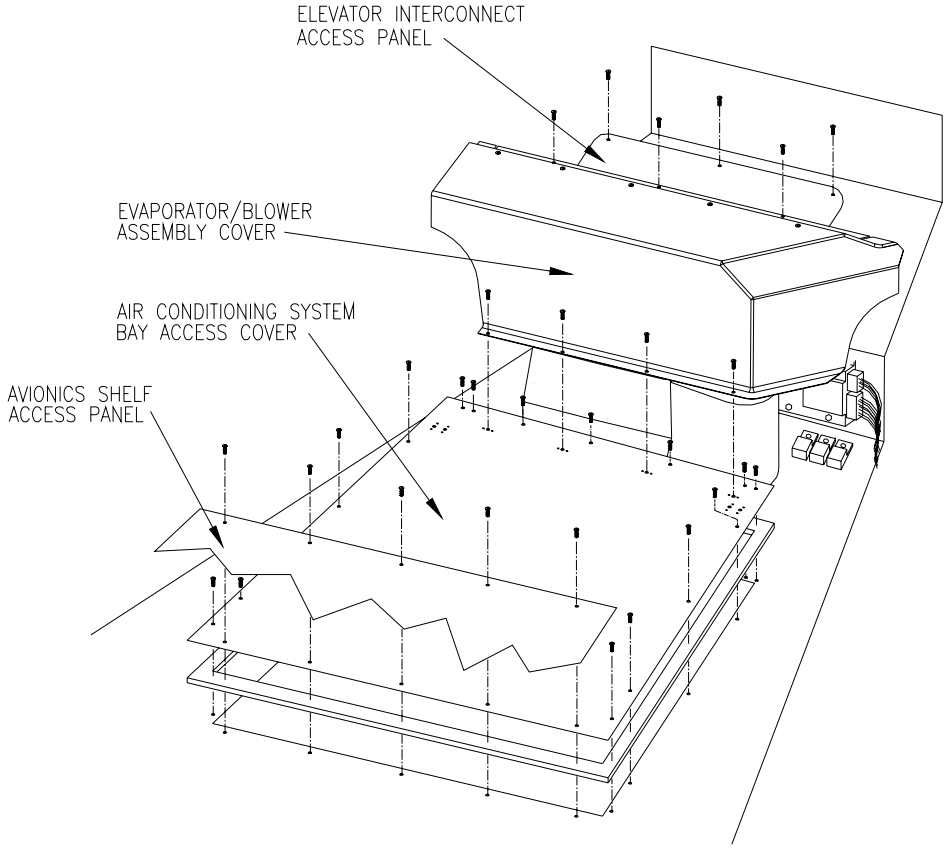


Figure 21 - 24 A/C Access Panels and Covers

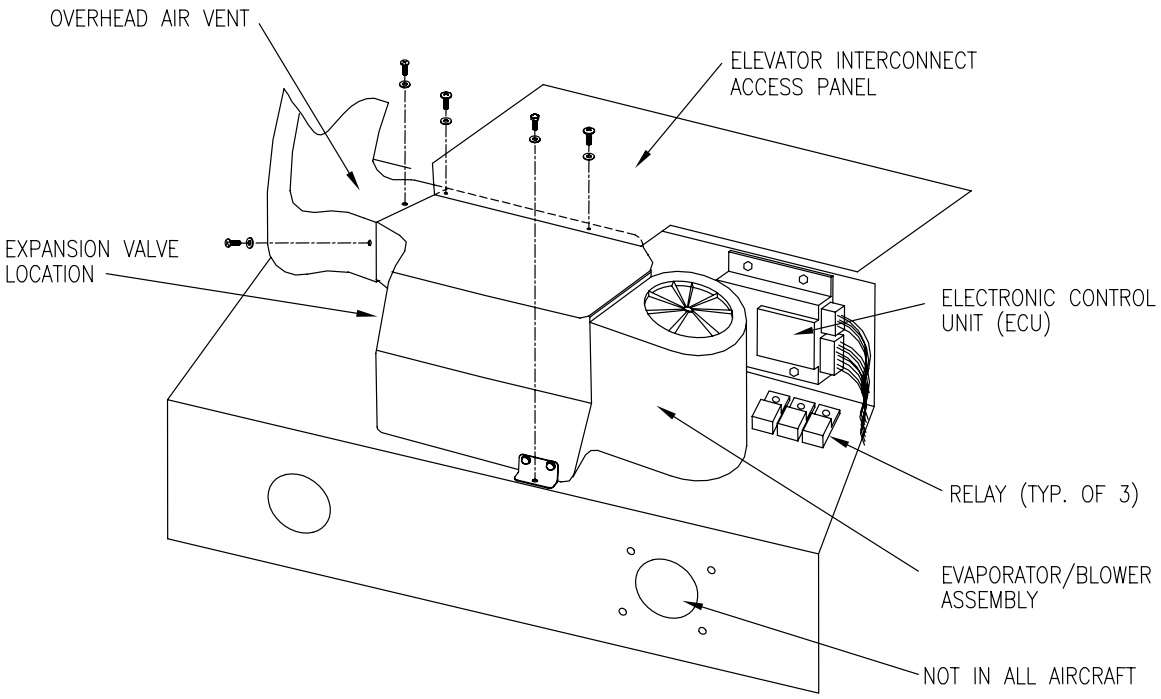


Figure 21 - 25 Evaporator/Blower Assembly

a) Removal of the Evaporator/Blower Assembly – To remove the Evaporator/Blower Assembly use the following instructions.
   1) Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.
   2) Remove the Evaporator/Blower Assembly cover and the elevator interconnect cover. See Figure 21 - 24.
   3) Evacuate the air conditioning system per instructions in paragraph 21-15.h.
   4) Disconnect all electrical connections to the Evaporator/Blower Assembly.
   5) Remove the Expansion Valve per paragraph 21-15.e.4.
   6) Disconnect the low-pressure refrigerant hose from the Evaporator/Blower Assembly and seal all open fittings with a cap or plug.
   7) Remove 2 screws and washers connecting the overhead air duct to the Evaporator/Blower Assembly.
   8) Remove 3 screws attaching the Evaporator/Blower Assembly to the aircraft.
   9) Remove the Evaporator/Blower Assembly.

b) Installation of the Evaporator/Blower Assembly – To Install the Evaporator/Blower Assembly use the following instructions.
   1) Slide the Evaporator/Blower Assembly outlet into the overhead air duct.
   2) Install 3 screws attaching the Evaporator/Blower Assembly to the aircraft.
   3) With 2 screws and washers, attach the overhead air duct to the Evaporator/Blower Assembly.

4) Install the Expansion Valve per paragraph 21-15.e.4.
5) Attach the low-pressure refrigerant hose to the Evaporator/Blower.
6) Reconnect all electrical connections to the Evaporator/Blower Assembly.
7) Replace the receiver/dryer if required per 21-15.e.6.
8) Evacuate and charge the system per paragraphs 21-15.h. and 21-15.i. Add 1 fluid oz. (29 cc) of refrigerant oil when installing a new Evaporator/Blower Assembly.
9) Replace the elevator interconnect cover and Evaporator/Blower Assembly cover.
10) Replace the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

c) Maintenance – The Evaporator/Blower Assembly is a self-contained unit and maintenance is not required.

6. Receiver/Dryer with Trinary Switch – The receiver/dryer acts as a storage and filter unit for refrigerant oil. The trinary switch serves two functions. One portion of the switch cycles the condenser fans once the refrigerant pressure reaches 235 psi (16 bar) and turns off the fans at 190 psi (13 bar). The other portion of the switch protects the system from damage due to low refrigerant or excessive pressures by turning off the compressor. Power to the compressor is interrupted once the refrigerant pressure drops below 29 psi (2 bar) or exceeds 397 psi (27 bar) at the trinary switch. The trinary switch automatically resets once pressures are restored to a safe level.

   The receiver/dryer must be replaced on the following conditions:

   - Compressor replaced due to failure or contamination.
   - Evaporator module replaced due to contamination.
   - Condenser replaced due to contamination.
   - Any hose or line replaced due to contamination.
   - Expansion valve replaced due to contamination.
   - The system has been opened to the ambient environment for more than 24 hours, e.g., components were not capped or plugged to preclude moisture contamination during service

   Generally, replace the receiver/dryer whenever a contamination event occurs or if the system has been opened and exposed to ambient environment for more than 24 hours.

   The Trinary Switch is considered an LRU (Line Replaceable Unit) which allows it to be replaced independently of the receiver/dryer assembly. The switch is sealed with an o-ring on the receiver/dryer and releases a Schrader valve upon removal thus allowing replacement without evacuating the refrigeration system. It is important to note that the 14VDC switch and the 28VDC switch, although identical electrically, have different electrical interface connectors. The correct switch must be ordered based on the configuration (14VDC or 28VDC) of the aircraft being serviced.

The receiver/dryer assembly is attached to the landing gear tab located beneath the baggage compartment, forward of the condenser bay on the co-pilot's side. See Figure 21 - 26.

Figure 21 - 26 Receiver/Dryer with Trinary Switch

a) Removal of the Receiver/Dryer and/or the Trinary Switch – To remove the Receiver/Dryer and/or the Trinary Switch use the following instructions.
   1) Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.
   2) Remove the Evaporator/Blower Assembly cover and the avionics shelf access panel. See Figure 21 - 24.
   3) Disconnect the electrical connections from the Trinary Switch.
   4) Remove the Trinary Switch, if desired. The Trinary Switch may be removed without evacuating the system. See description above. For removal of the Receiver/Dryer continue with steps 5) through 7) below.
   5) Evacuate the air conditioning system per instructions in paragraph 21-15.h.
   6) Disconnect the refrigerant hoses from the Receiver/Dryer and seal all open fittings with a cap or plug.
   7) Remove 2 bolts, washers, and self locking nuts and remove the Receiver/Dryer.

b) Installation of the Trinary Switch Only– To install the Trinary Switch use the following instructions.
   1) Screw in the Trinary Switch. The tightening torque for the Trinary Switch is 7 ft.-lbs.
   2) Reconnect the electrical connections to the Trinary Switch.
   3) Replace the avionics shelf access panel and the Evaporator/Blower Assembly cover.
   4) Replace the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

c) Installation of the Receiver/Dryer– To install the Receiver/Dryer use the following instructions.

1) Attach the Receiver/Dryer to the landing gear tab with two bolts, washers, and self locking nuts.
2) Attach the refrigerant hoses to the Receiver/Dryer.
3) Reconnect the electrical connections to the Trinary Switch.
4) Evacuate and charge the system per paragraphs 21-15.h. and 21-15.i. Add 1 fluid oz. (29 cc) of refrigerant oil when installing a new Receiver/Dryer.
5) Replace the avionics shelf access panel and the Evaporator/Blower Assembly cover.
6) Replace the baggage floor and the baggage shelf carpet per instructions in Chapter 25.

d) Maintenance – The Receiver/Dryer with Trinary Switch Assembly is a self-contained unit and maintenance is not required.

7. Cabin Temperature Sensor – The cabin temperature sensor assembly contains a calibrated thermistor and fan used to draw cabin air over the thermistor. The sensor assembly provides input data to the ECU. Cabin temperature is determined by comparing the inside temperature of the aircraft with the desired cabin temperature setting. The temperature is regulated by cycling the compressor and regulating the evaporator blower speed during cooling requirements and by regulating the heater temperature door and heater/defogger blower speed during heating requirements. If the sensor fails, a fault will be noted in the ACCS and manual temperature control will be required through the control head. The sensor assembly is located behind the co-pilot's knee bolster. If the Garmin G1000 system is installed, the sensor assembly is at the back of the instrument panel behind the MFD or mounted behind the headset holder at the front face of the Tower.

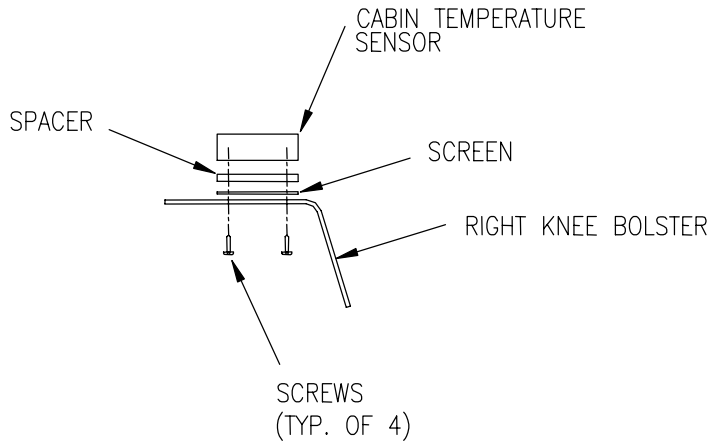


Figure 21 - 27 Cabin Temperature Sensor (Basic or Avidyne Option)

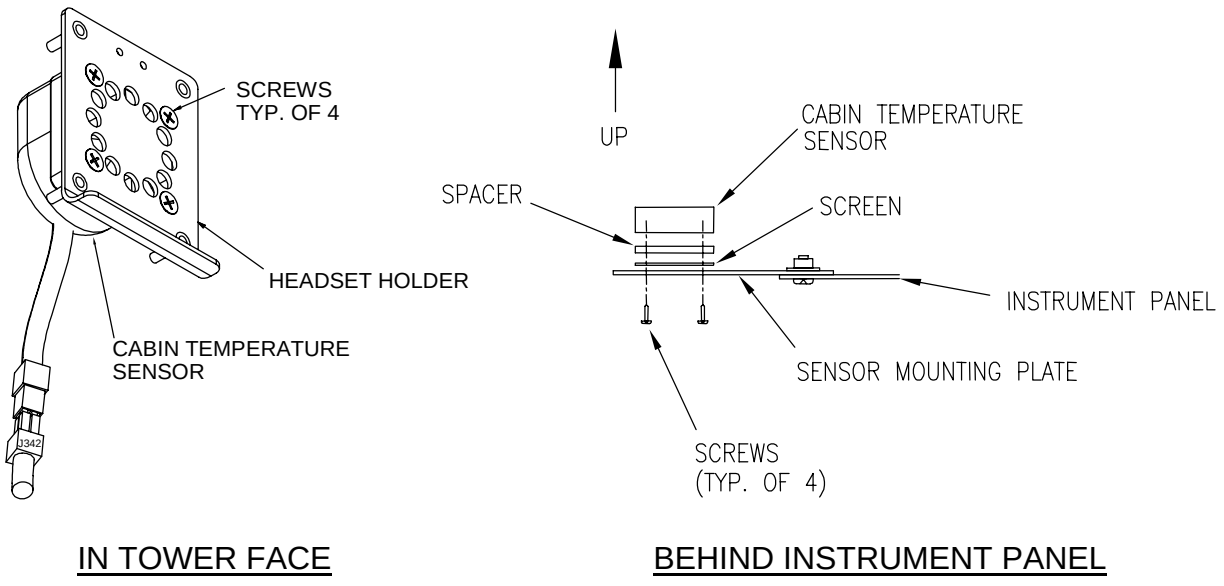


Figure 21 - 28 Cabin Temperature Sensor (Garmin G1000 Option)

a) Removal of the Cabin Temperature Sensor – To remove the Cabin Temperature Sensor use the following instructions.
   1) Remove the Right Knee Bolster per instructions in Chapter 25. If the Garmin G1000 system is installed the sensor may be accessed from underneath the instrument panel or by removing the tower per Chapter 25, as applicable.
   2) Disconnect the electrical connection from the Cabin Temperature Sensor.
   3) Remove 4 screws attaching the Cabin Temperature Sensor to the top of the Right Knee Bolster, to the sensor mounting plate, or to the headset holder, as applicable.
   4) Remove the Cabin Temperature Sensor, spacer, and screen.

b) Installation of the Cabin Temperature Sensor – Installation is the reverse of removal.

c) Maintenance – The Cabin Temperature Sensor is a self-contained unit and maintenance is not required.

8. Outside Air Temperature (OAT) Sensor – The OAT temperature sensor continuously measures the temperature of the air passing by the outside of the aircraft. The sensor is used as a basic variable for temperature regulation. The ECU also uses the signal from this sensor to calculate the temperature for the OAT indication on the control head display. The temperature sensor is located in the condenser bay on the forward bulkhead. See Figure 21 - 29.

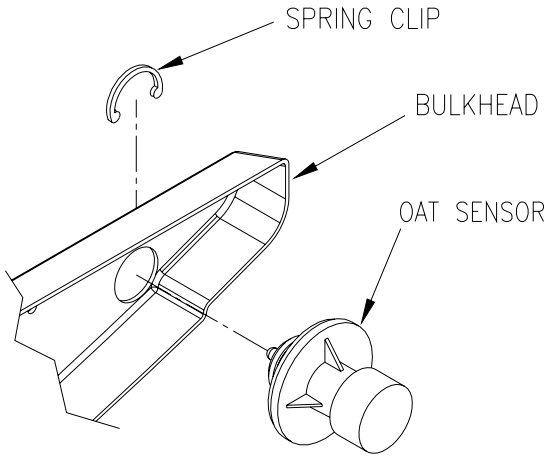


Figure 21 - 29 OAT Sensor

a) Removal of the OAT Sensor – To remove the OAT Sensor use the following instructions.
   1) Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.
   2) Remove the Evaporator/Blower Assembly cover, the avionics shelf access panel and the air conditioning system bay access cover. See Figure 21 - 24.
   3) Disconnect the electrical connection to the OAT.
   4) Remove the "C" clip attaching the OAT sensor to the bulkhead and remove the sensor.

b) Installation of the OAT Sensor – Installation is the reverse of removal.
   1) Install PR1428 B1/2 or RTV5818 sealant, as required, at all openings between the air conditioning bay and the cabin area (including underlying floors) to completely seal off the cabin area from the air conditioning bay.
   2) Install McMaster-Carr foam tape P/N 8512K24, as required, around the perimeter of the air conditioning system bay.
   3) Replace the air conditioning system bay access cover. Seal the air conditioning system bay access cover to the floor using 3M 425 adhesive foil tape to create a leak proof seal.
   4) Replace the avionics shelf access panel, and the Evaporator/Blower Assembly cover.

c) Maintenance – The OAT Sensor is a self-contained unit and maintenance is not required.

9. Defog/Floor Vent Valve Assembly – During heater operation this valve normally distributes air to the floor vents at a volume of approximately 90% with the remaining 10% going to the windscreen defog outlets. When the Defog Ⓦ mode is selected the valve distributes nearly 100% of the air to the windscreen defog outlets. The valve

operates in either a “full on” or “full off” position. A minimum of 10% of the air distributed by the heater/defog blower is distributed to the defog ducts anytime the heater/defog blower is operating. This valve is attached to the upper windscreen defrost channel/plenum, beneath the center of the instrument panel. See Figure 21 - 30 or Figure 21 - 31.

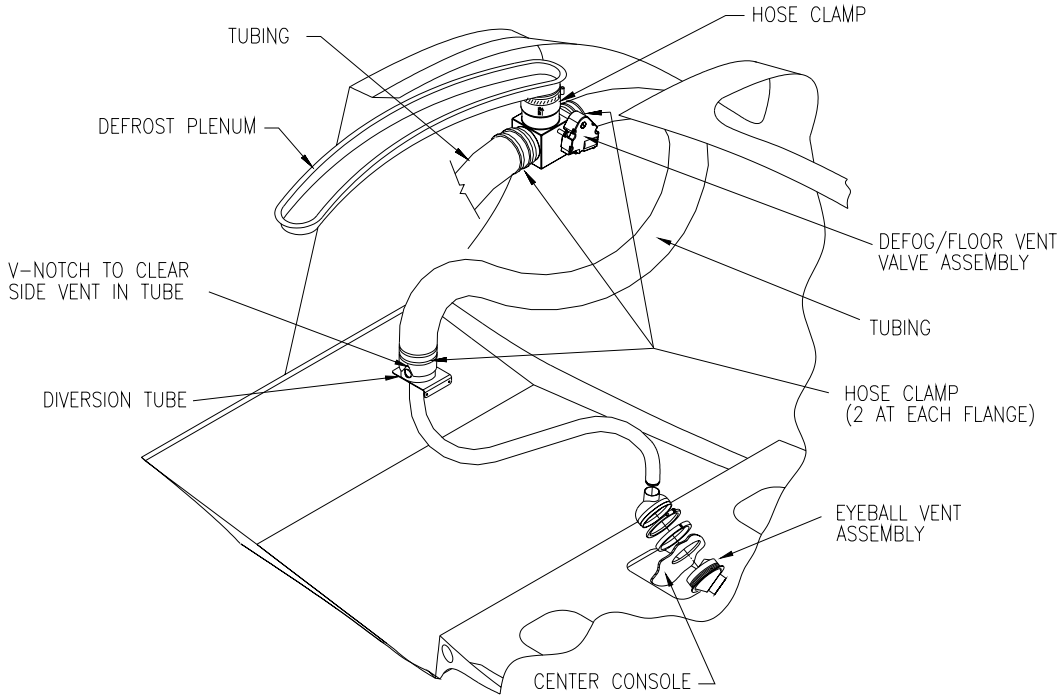
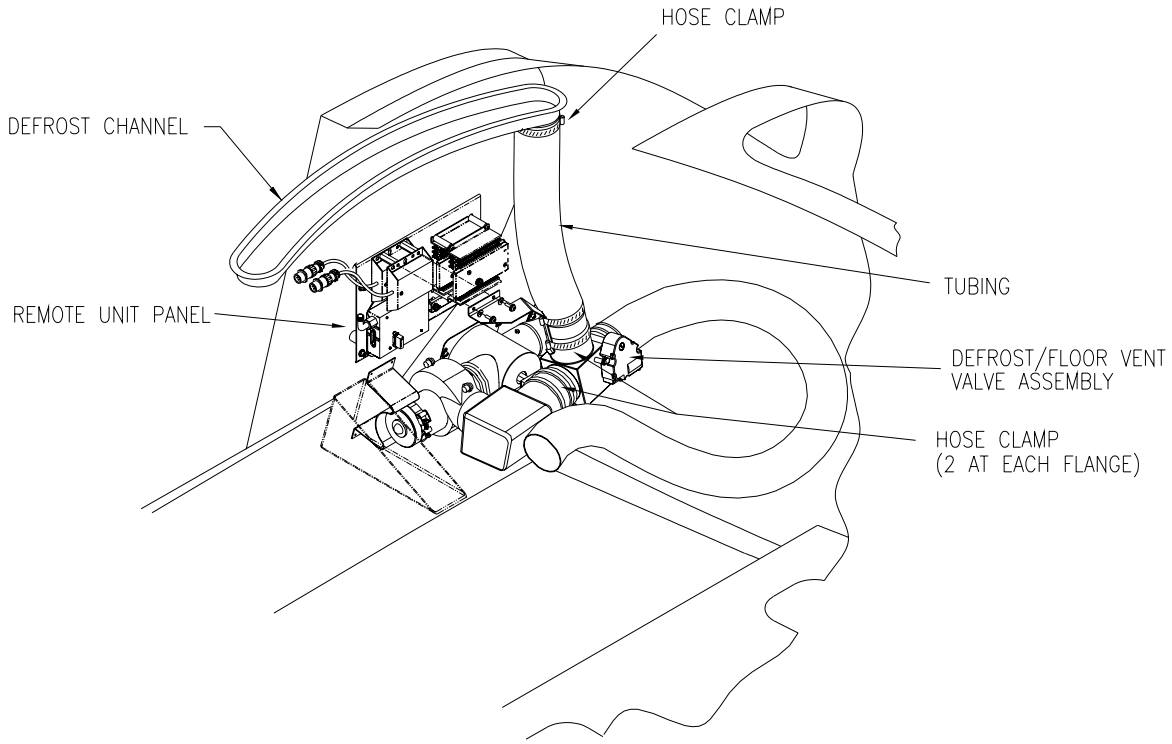
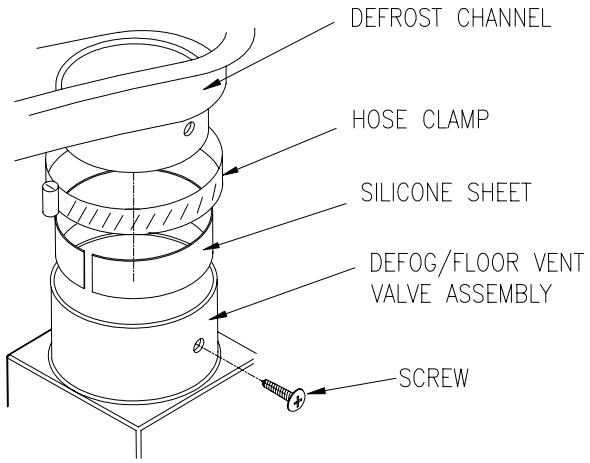


Figure 21 - 30 Defog/Floor Vent Valve Assembly (S/N 42001 to 42500)



CONFIGURATION WITH DUCTING BETWEEN THE
DEFOG/FLOOR VENT VALVE ASSEMBLY AND THE DEFROST CHANNEL



CONFIGURATION WITH THE DEFOG/FLOOR VENT VALVE ASSEMBLY
ATTACHED DIRECTLY TO THE DEFROST CHANNEL

Figure 21 - 31 Defog/Floor Vent Valve Assembly (S/N 42501 and on)

a) Removal of the Defog/Floor Vent Valve
   1) Disconnect the electrical connector from the defog/floor vent valve.
   2) Remove the hose clamps attaching the defog/floor vent valve to the flexible ducting and defrost channel.
      (a) On Garmin G1000 aircraft, the defog/floor vent valve may be attached directly to the defrost channel. With this condition, remove the hose clamp and a MS51861-25C screw. There is a length of silicone sheeting between the defog/floor vent valve and the defrost channel.
   3) Pull the flexible ducting away from the defog/floor vent valve.
   4) Remove the RTV5818 silicone sealant applied between the defog/floor vent valve and the defrost channel and remove the valve.

b) Installation of the Defog/Floor Vent Valve
   1) Defog/floor vent valve attached directly to the defrost channel.
      (a) Fill the gap between the inner diameter of the valve flange and the outer diameter of the defrost channel flange with .062" or .093" AMS3320 silicone sheet.
      (b) When installing a new defog/floor vent valve, match drill a 0.166" diameter (#32) hole in the valve flange.
      (c) Install a MS51861-25C screw through the valve flange, the defrost channel flange, and the silicone sheet. Dry torque hand tight plus 1/4 turn.
      (d) Apply RTV5818 silicone all around the inlet port, install flexible ducting, and secure with a hose clamp.
      (e) Reconnect the electrical connector.
      (f) Turn on the power, and test the defog/floor vent valve for proper operation.
   2) Defog/floor vent valve not attached directly to the defrost channel.
      (a) Apply RTV5818 silicone all around the inlet and outlet ports of the defog/floor vent valve.
      (b) Slide the flexible ducting over the inlet and outlet ports of the defog/floor vent valve. Rotate the valve as required to clear the rudder pedal tubes and ECS servo motor actuator screw, and secure with hose clamps.
      (c) Reconnect the electrical connector.
      (d) Turn on the power, and test the defog/floor valve for proper operation.

a) Maintenance – The Defog/Floor Vent Valve Assembly is a self-contained unit. Only a periodic check for proper valve operation is required.

10. ECS Shut-Off Valve Assembly – When the ACCS is operating in the air conditioning mode, this valve remains closed to prevent outside air from entering through the ECS mixing valve and entering the cabin. During the heat mode the valve opens allowing air from the ECS mixing valve to enter the cabin. This valve is attached to the heater blower, beneath the center of the instrument panel. See Figure 21 - 32 or Figure 21 - 33.

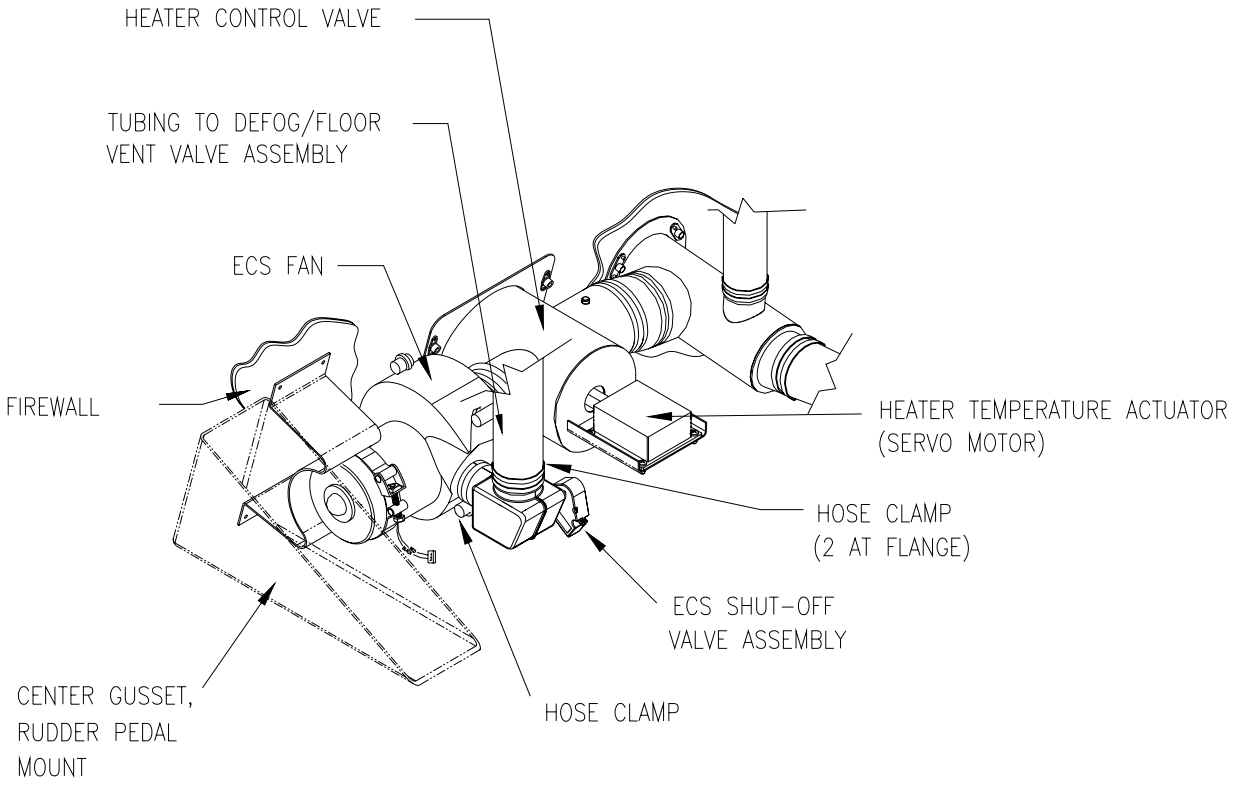


Figure 21 - 32 ECS Shut-Off Valve Assembly (S/N 42001 to 42500)

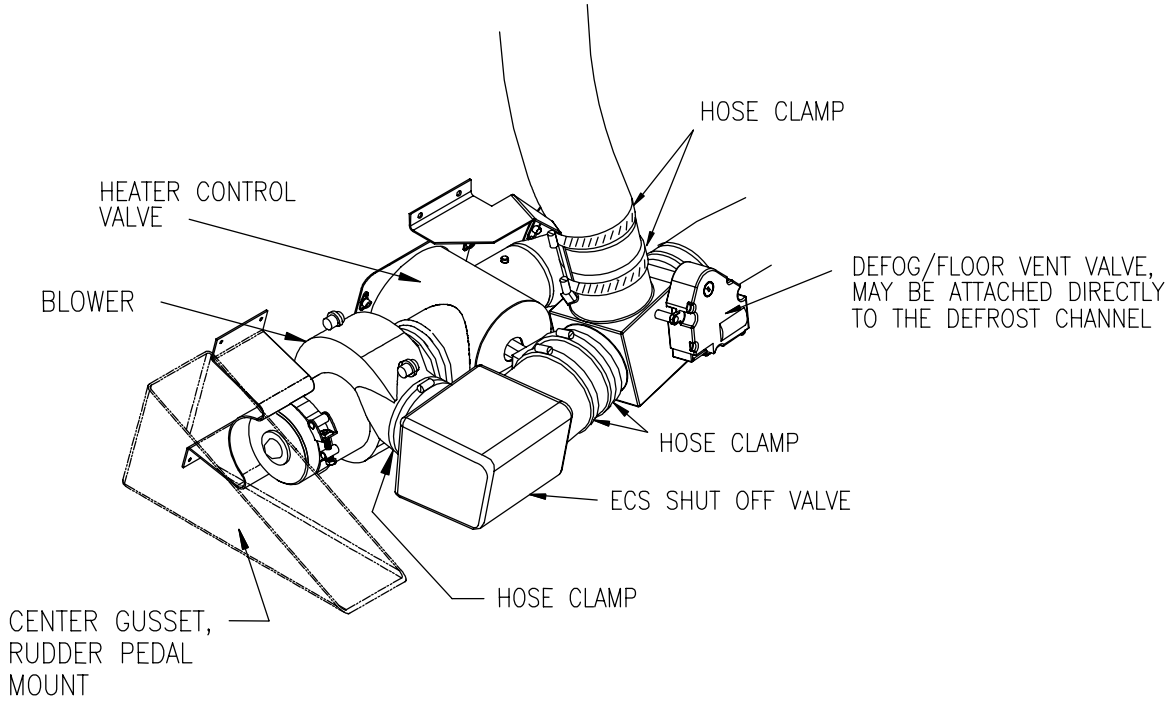


Figure 21 - 33 ECS Shut-Off Valve Assembly (S/N 42501 and on)

a) Removal of the ECS Shut-Off Valve Assembly – To remove the ECS Shut-Off Valve Assembly use the following instructions.
   1) Disconnect the electrical connector from the ECS Shut-Off Valve Assembly.
   2) Remove 3 tangential worm drive hose clamps attaching the ECS Shut-Off Valve Assembly to the flexible ducting and to the ECS Fan.
   3) Pull the flexible ducting away from the ECS Shut-Off Valve Assembly.
   4) Remove the RTV5818 silicone sealant applied between the Shut-Off Valve Assembly and the ECS Fan and remove the Shut-Off Valve.

b) Installation of the ECS Shut-Off Valve Assembly – To Install the ECS Shut-Off Valve Assembly use the following instructions.
   1) Installation is the reverse of removal.
   2) Apply RTV5818 silicone sealant at the connection between the defrost plenum and the Shut-Off Valve Assembly before securing the clamp.

c) Maintenance – the ECS Shut-Off Valve Assembly is a self-contained unit. Only a periodic check for proper valve operation is required.

11. Heater Temperature Actuator (Servo Motor) – Heated air temperature in the cabin is adjusted via the heater temperature actuator. Its adjusting range extends from "heating final stop" (all air channeled from the heat exchanger) to "cooling final stop" (no air channeled from the heat exchanger)." The potentiometer reports the position of the actuator to the ECU as a feedback value. The heater temperature actuator is activated by the ECU according to the temperature selected on the control head. The heater temperature actuator is located on the heater/ECS mixing box in the forward cabin. See Figure 21 - 32, Figure 21 - 33, and Figure 21 - 5.
    a) Removal and installation of the Heater Temperature Actuator – Removal and installation of the Heater Temperature Actuator is the same as removal and installation of the Servo Motor per paragraph 21-4.
    b) Maintenance – The Heater Temperature Actuator is a self-contained unit and requires only a periodic check for proper operation.
12. Electronic Control Unit (ECU) – The ECU is the control unit of the ACCS incorporating all the electronic circuitry and utilizing digital processing logic. The ECU receives signals from the control head and supplies outputs accordingly. The ECU also contains the blower control unit which regulates the evaporator and the 11 heater blower speeds. Voltage at the blowers is reported back to the control head. For S/N 42001 to 42501, the ECU is located on the pilot's side, beneath the evaporator cover, in the baggage compartment. See Figure 21 - 25. For S/N 42502 and on, the ECU is mounted on the fuselage skin under the baggage bulkhead access panel.
    a) Removal of the ECU – To remove the ECU use the following instructions.
       1) Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.
       2) For S/N 42001 to 42501, remove the Evaporator/Blower Assembly cover. See Figure 21 - 24. For S/N 42502 and on, remove the baggage bulkhead access panel.

3) Disconnect all electrical connections to the ECU.
4) Remove 3 screws (S/N 42001 to 42501) or 4 self-locking nuts and washers (S/N 42502 and on) and remove the ECU.

b) Installation of the ECU – Installation of the ECU is the reverse of removal.

c) Maintenance – The ECU is a self-contained unit and maintenance is not required.

13. Control Head – For the Basic or Avidyne option, the control head is located to the right of the flap panel. For the Garmin G1000 option the control head is located in the tower assembly. The control head receives inputs from the pilot, or co-pilot, and the ACCS components. It is the user interface for the ACCS. Fault codes generated by the ACCS are displayed on the control head LED screen for troubleshooting. See Figure 21 - 34.

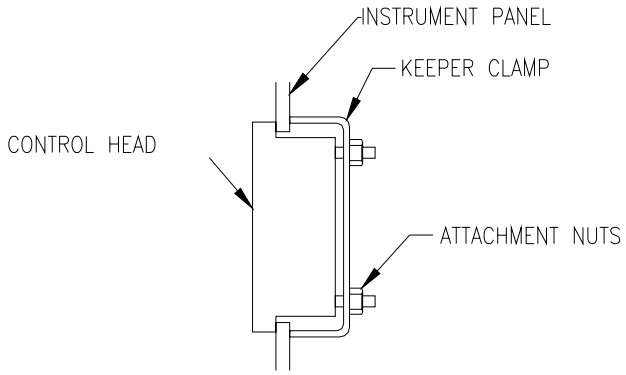


Figure 21 - 34 Control Head Attachment

a) Removal of the Control Head – To remove the Control Head use the following instructions.
1) Turn the power off and disconnect the wiring harness from the back of the Control Head.
2) From the back side of the instrument panel remove the four nuts and keeper clamp attaching the Control Head to the instrument panel.
3) Remove the Control Head by pulling it forward from the front of the instrument panel.

b) Installation of the Control Head – Installation of the Control Head is the reverse of removal.

c) Maintenance –The Control Head is a self-contained unit. A periodic check for Defog/Floor Vent Assembly, ECS Shut-Off Valve Assembly, and Heater Temperature Actuator operation is required. A functional check should be completed at each 100 hour or annual inspection.
1) Evaluate the adequacy of the system in terms of crew and passenger comfort, as well as the defroster's ability to control fogging of the windscreen
2) Check for proper operation of the fresh air, air conditioning, heat, defog., air volume control, temperature control, and air output control.

3) Evaluate the eyeball vents for proper operation.

f. **Tools and Equipment –** The following tools and equipment are required for leak testing, evacuating, and charging the refrigerant system:

a) Gaseous dry nitrogen, regulated source (0-500 psig or 0-35 bar).
b) R-134a refrigerant
c) Refrigerant oil, Ester RL-500S.
d) R-134a refrigerant charging manifold with gauges and hoses with quick-disconnect couplers. The coupler has a knob on top which is used to engage and disengage the Schrader valve in the aircraft service ports.
e) R-134a charging station and recovery/recycling station (SAE standard J2210).
f) Air conditioning vacuum pump.
g) Hose adapter, for nitrogen bottle (1/2 in. male acme to 1/4 in. SAE female flare).
h) Hose adapter for vacuum pump (1/2 in. male acme to 1/4 in. SAE female flare).
i) Electronic leak detector.
j) Burroughs tool No. BT-33-73F (or equivalent) belt tension gauge.
k) Torque wrench capable of reading in.-lbs or Nm.
l) Small hand tools and socket set.
m) Thermometer, 0-150ºF.
n) Inspection Mirror, adjustable.
o) Service light or flashlight.
p) Shop hand towels
q) Weight scale, 0-50 lbs (0-25 kg) capable of 0.1 pound (5 gram) precision.

g. **Leak Check Procedure** – System leak check is required whenever the air conditioning system is not cooling properly, there is a system component replacement or a loss of R-134a refrigerant charge.

1. Determine if the system is fully charged. See Figure 21 - 37, Figure 21 - 38, Figure 21 - 39, or Figure 21 - 40.
2. If the system is empty evacuate and charge the system with 0.6 lbs. (296 grams) of R-134a refrigerant.
3. Position the aircraft in a wind free work area.
4. Remove the engine cowling per instructions in Chapter 71.
5. Remove the baggage floor and the baggage shelf carpet per instructions in Chapter 25.
6. Remove the evaporator/blower assembly cover, the avionics shelf access panel and the air conditioning system bay panel.
7. Operate the aircraft engine with the air conditioning on for 5 to 10 minutes until the operating temperature and pressures are achieved based on the existing refrigerant charge.
8. Aircraft cabin doors and the baggage compartment door must be left open during test operation.
9. Shut off the aircraft engine. Use an R-134a electronic leak detector and search for leaks. Move the probe slowly along the bottom side of lines and fittings, because R-

134a is heavier than air. Fittings, lines, or components that appear to be oily usually are the site of a refrigerant leak.

10. To inspect for evaporator core leaks, ensure the aircraft engine is NOT running, set the ACCS temperature to 55ºF (13ºC), select "LO" for the blower speed and check the air conditioning outlet vents. Also inspect the evaporator drain tube outlet for the presence of refrigerant oil.
11. Tighten or repair joints as required to stop leaks.

CAUTION

Do not over tighten plumbing connections. Stripped threads or cracked flanges may result.

h. **System Evacuation Procedure –** If the air conditioning system has been open to the atmosphere, it must be evacuated before the system can be charged. Moisture and air mixed with refrigerant will raise the compressor head pressure above acceptable operating levels. This will reduce the performance of the air conditioning system and damage the compressor. Moisture will also boil at near room temperature when exposed to vacuum.

R-134a refrigerant is a hydro fluorocarbon (HFC) that does not contain chlorine. An R-134a refrigerant Recovery/Recycling Station that meets SAE standard J2210 must be used to evacuate the refrigerant system. Refer to the operating instructions provided with the equipment for proper operation.

1. Connect the charging manifold to the service ports.
2. Connect the manifold charging (yellow) hose to the vacuum pump and turn on the pump.
3. Open both valves of the charging manifold gauge set.
4. Open both valves of the charging hose quick connect fittings.
5. Observe the charging manifold gauges to verify vacuum.
6. Evacuate the system until 25 to 27 in. Hg (635 to 685 mm Hg) or greater vacuum is obtained.
7. Close all valves and turn off the vacuum pump.
8. If the system maintains the specified vacuum for 30 minutes, start the vacuum pump, open the suction and discharge valves, and evacuate the system for an additional 40 minutes, minimum.
9. If the specified vacuum is not maintained, locate and repair the refrigeration system leak before continuing.
10. Close all valves and turn off the vacuum pump.
11. Disconnect the manifold service hose from the vacuum pump. The system is now ready for refrigerant charging.

CAUTION

Any change in vacuum pressure, or failure to achieve a system pressure of 25 to 27 in. Hg (635 to 685 mm Hg) vacuum indicates the presence of a plumbing leak. Repeat the leak check procedure. Locate and fix all leaks.

i. **System Charging Procedure –** There are various methods of charging refrigerant into the air conditioning system. These include using a refrigerant recovery/recycle unit (refer to the manufacturer's instruction manual). The use of manifold pressure gauges, R-134a cylinder, and a weight scale is outlined in this section. Only R-134a refrigerant is to be used. Other refrigerants will damage the system. Overcharging of the system will result in reduced performance, reduced service life, and/or damage to the system components.

1. Leak check and evacuate the system per paragraphs 21-15.g. and 21-15.h.
2. Connect the yellow manifold charging hose to the R-134a cylinder and open the valve.
3. Crack open the charge hose fitting at the manifold gauge set and vent air from the hose until refrigerant is evidently escaping.
4. Place the refrigerant bottle on a 0-50 lb. (0-25 kg) scale and record the weight.
5. Open both manifold gauge set valves. Add refrigerant to the system until pressure stabilizes.
6. Close the manifold valves and verify that system pressure is 50 psig (3.4 bar) or greater.

NOTE

The system pressure must be above 50 psig (3.4 bar) to close the low-pressure cutoff switch. Otherwise the compressor will not turn on.

7. If necessary, warm the refrigerant bottle by immersion in warm water.

CAUTION

Do not use open flame to warm the refrigerant bottle. Do not heat the bottle above 125ºF (51ºC).

8. Move the aircraft to a run-up area.
9. Start the aircraft engine. Select "AUTO on the ACCS control head and set the temperature to 55ºF (13ºC). Idle the aircraft engine at 1200 to 1800 RPM.

CAUTION

Do not open the high-pressure (red) valve on the manifold gauge set.

10. With the system operating, observe the system discharge and suction pressures.
11. With the R-134a cylinder connected to the charging hose, charging container shutoff valve open and hose purged of air, slowly open the suction (blue) manifold valve. The suction pressure will increase to 60 to 70 psig (4.1 to 4.8 bar).

NOTE

As refrigerant enters the compressor, a slight increase in compressor discharge pressure will be noticed.

12. Continue to add refrigerant until 24 oz. (680 grams) of refrigerant has been added.
13. Close the suction manifold valve (blue) and let the system operate for 5 to 10 minutes to evaluate performance.

NOTE

Letting the system stabilize is required until the expansion valve can stabilize the system pressure.

NOTE

System charge is approximately 24 oz. (680 grams) of R-134a refrigerant.

CAUTION

Do NOT overcharge the air conditioning system. Overcharging will cause excessive compressor head pressure, loss of cooling, noise and system failure.

14. With the system fully charged and operating, observe the suction and discharge pressures. Typical values at various ambient temperatures, with hot cabins, are shown below. These values should be used as a reference for troubleshooting, not as a sole source. See Figure 21 - 35.

Ambient Temperature	21°C (70°F)	27°C (80°F)	32°C (90°F)	38°C (100°F)	43°C (110°F)
Air Temperature At Evaporator Outlet	7-9°C (45-48°F)	8-13°C (46-55°F)	11-16°C (52-61°F)	14-20°C (57-68°F)	17-25°C (63-77°F)
Low Side Service Port Pressure	18 psi 30	20 psi 35	22 psi 39	25 psi 43	26 psi 47
High Side Service Port Pressure	150 psi 220	160 psi 235	220 psi 310	250 psi 320	280 psi 340

Figure 21 - 35 Typical Suction and Discharge Pressure

15. Allow the system to operate for 10 minutes then shutdown.

NOTE

After shutdown, both suction and discharge pressures will begin to equalize.

16. Close the refrigerant container shutoff valve. Record the refrigerant container final weight and calculate the system refrigerant charge.

17. Turn the knob on the suction and discharge charging hose quick coupler to closed position and disconnect the hoses from the service ports.
18. remove the yellow charging hose from the refrigerant container and store the manifold gauge set.
19. Install the service port caps.

j. **System Flushing** – System flushing can be successfully done to systems where there is insignificant or no evidence of debris or FOD in the system. Failure to follow the actions and direction below will void all Seamech warranty considerations.

1. System flushing ***cannot*** be done to the complete system where a compressor has overheated and failed. The industry term for this type of failure is "Black Death" and is usually the result of low amounts of compressor lubricating oil in the system. This type of failure burns the remaining oil in the system and, combined with the polymers in the compressor, will form a hard, carbonized and impossible to remove buildup primarily in the receiver dryer and condenser. In this case, the hoses between the compressor and evaporator module, as well as the receiver dryer and condenser must be replaced. The expansion valve and evaporator module should be removed and inspected for evidence of black carbonized buildup. If black buildup exists, the expansion valve and/or evaporator module must also be replaced. If no evidence of black carbonized buildup exists, the expansion valve and evaporator module can be flushed using the techniques below.
2. If a contamination event is the result of an internal compressor mechanical failure or other debris in the system, and there is evidence of metal or debris in the VCS components, then all of the VCS hoses and components must be disconnected and inspected. Today's high efficiency condenser and evaporator coils are difficult to clean of all debris. These coils are parallel flow devices with deep wells on the manifolds where debris can and flushing material can pool making debris removal by flushing impossible. If there is evidence of debris in these devices, it is recommended that they be replaced to ensure the system can operate debris free when reassembled and serviced. In the event of a compressor mechanical failure, it is likely that the receiver/dryer, condenser and perhaps the evaporator module may require replacement. Hoses and other components of the system should be able to be flushed with success.
3. If flushing can be done, only DuPont TM Mobile AC Flush Fluid is authorized on Seamech systems. The following link will provide more information about the fluid and where it can be obtained. http://refrigerants.dupont.com/Suva/en_US/products/suva_mobile_flush.html. High pressure and flow flushing equipment is required. Traditional flush guns, aerosol flushes and pour in flush methods do not produce the velocity necessary to force contaminates out of the affected component.
4. VERY IMPORTANT - Flushing ***must*** take place in the reverse flow direction (low side to high side) to push contaminates back out of the affected components on a component by component basis. This means when an evaporator coil is flushed, the thermal expansion valve (TXV) must be removed and flushing material must enter into the downstream or suction side fitting of the evaporator coil. In a similar way,

flushing material must be applied to the bottom or discharge fitting of the condenser. Use gravity to assist in the cleaning.

CAUTION

It is critical that ALL of the flush be removed from the condenser. Any remaining flush in the condenser can cause the compressor to fail prematurely. If there is any doubt as to whether there is flush in the condenser, it is strongly advised that the condenser be replaced. Evidence of flush in a failed compressor will void any compressor warranties.

5. The thermal expansion valve must be removed from the evaporator module and thoroughly cleaned. Any contaminates in the expansion valve may cause the valve to operate improperly.
6. The receiver dryer can never be flushed because it is not possible to remove all of the flushing compounds. The receiver dryer contains desiccant which will absorb the flushing compound. The receiver dryer should be replaced if there was any type of contamination event.
7. All system hoses should be disconnected and flushed independently, opposite the direction of normal system refrigerant flow.
8. Compressors shall ***never*** be flushed. Flushing material will harm the compressor and displace any oil in the unit. Do ***not*** flush compressors.
9. Once all of the affected components of the system have been flushed, all components shall be blown dry with clean and very dry filtered shop air or nitrogen. VERY IMPORTANT – using air or nitrogen that is not dry will introduce moisture into the system components. This will cause internal corrosion over time.
10. It is vitally important to remove all of the flushing chemicals from all of the components. This may mean the components have to be rotated, shaken, or tilted to ensure all of the flushing material has been removed. Residual flushing material can absorb the compressor oils and can lead to premature failure of the compressor and degraded cooling performance. Repeat the “blow out” and removal of the flushing compounds until you are confident that all contamination debris and flushing chemicals have been removed.
11. All hose o-rings shall be replaced and the system reassembled per this maintenance manual.
12. Follow requirements for servicing the system with oil. Only Ester RL-500S system oil can be used. If a new compressor is used, they usually contain the required amount of oil internal to the compressor. The VCS can then be reserviced per this maintenance manual.

Condition	Possible Cause	Correction
Low side and high side pressure low	1. System refrigerant is low	1. Discharge, evacuate, leak test charge system
Low side pressure high and high side pressure low	1. Internal leak in compressor 2. Drive belt slipping	1. Replace compressor 2. Tension or replace belt
Low side and high side pressure high	1. Condenser fins obstructed 2. Air in refrigeration system 3. Condenser fans inoperative 4. Refrigerant system overcharged	1. Clean condenser fins 2. Evacuate, leak test and charge system 3. Troubleshoot condenser fans 4. Recover refrigerant and recharge
Low side pressure low and high side pressure high	1. Restriction in refrigerant hose 2. Restriction in receiver-dryer 3. Restriction in condenser 4. Expansion valve is defective	1. Check hoses for kinks and replace if necessary 2. Replace receiver-dryer 3. Replace condenser 4. Replace expansion valve
Low side and high side pressures normal (inadequate cooling)	1. Check for correct cooling at evaporator assembly outlet	1. If temperature is correct check for obstruction or leak in cabin ducting
Compressor noise	1. Internal compressor damage 2. Refrigerant system overcharged 3. Loose compressor mounting 4. Loose or worn compressor drive belt	1. Replace compressor 2. Recover refrigerant and recharge 3. Tighten compressor mounting bolts to correct torque values 4. Tension or replace belt
Excessive vibration	1. Incorrect belt tension 2. Compressor loose 3. Refrigerant system overcharged 4. Drive or idler pulley worn	1. Tension or replace belt 2. Tighten compressor mounting bolts to correct torque values 3. Recover refrigerant and recharge 4. Replace defective pulley
Condensation leaking inside Aircraft	1. Evaporator drain plugged or kinked	1. Clean drain hose and check for proper installation
Frozen evaporator coil	1. Faulty evap temp sensor 2. Obstructed evaporator coil	1. Replace evap temp sensor 2. Removed obstruction

Figure 21 - 36 Pressure and Performance Diagnosis

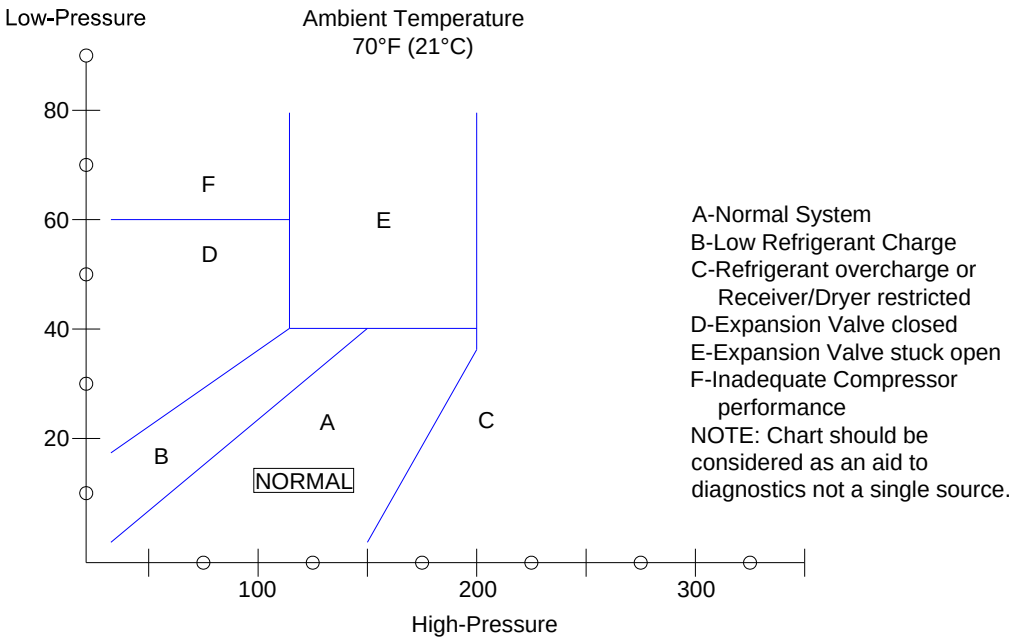


Figure 21 - 37 Pressure and Performance at 70ºF (21ºC)

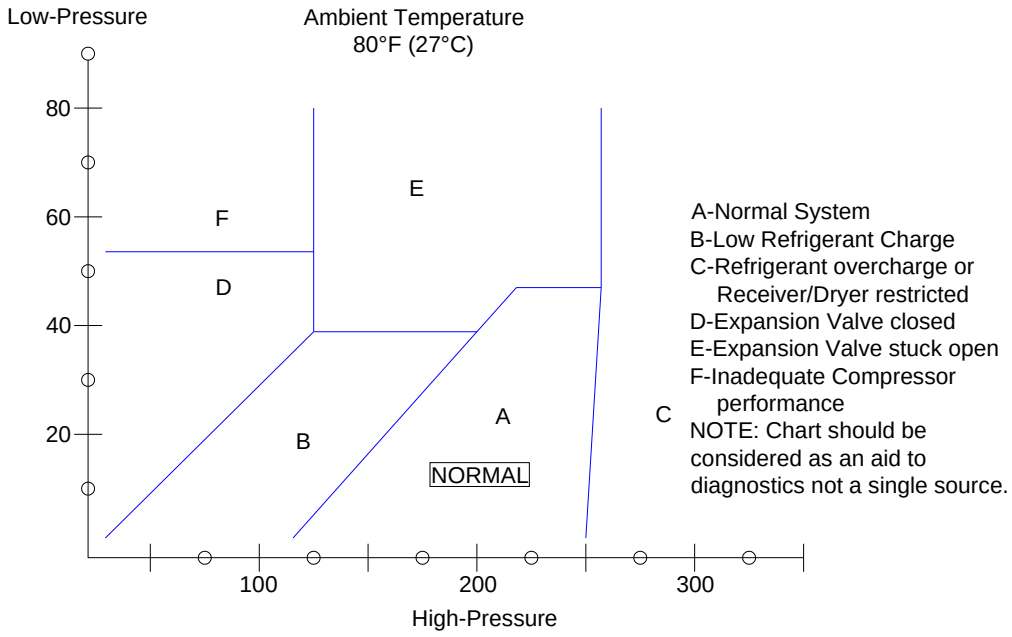


Figure 21 - 38 Pressure and Performance at 80ºF (27ºC)

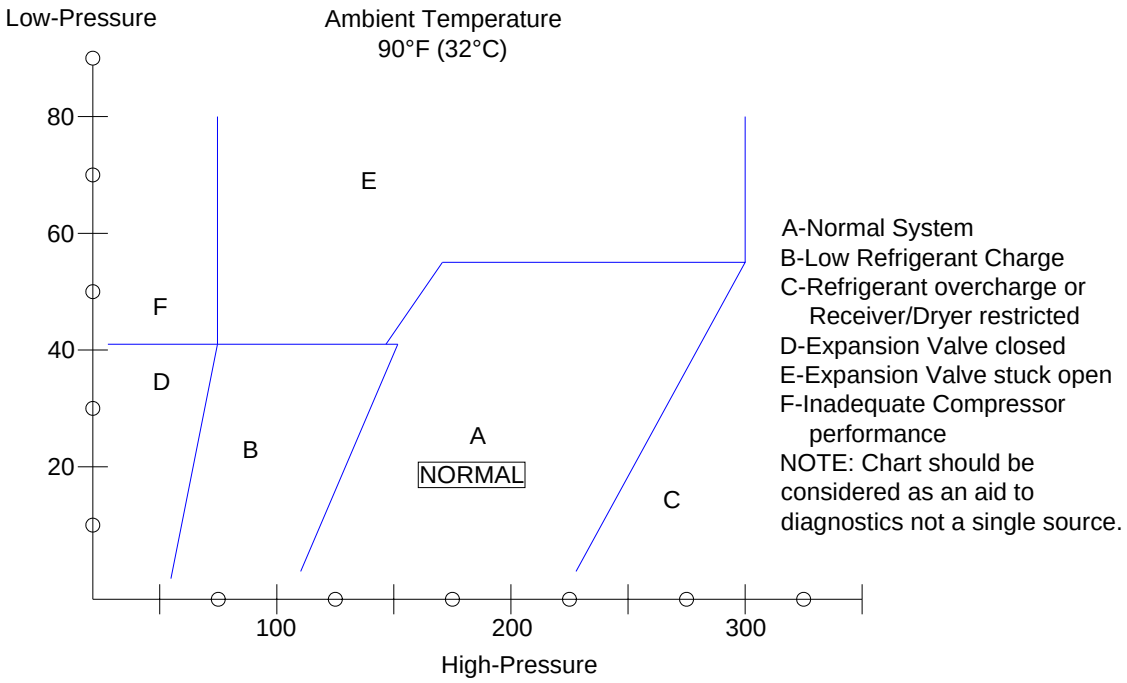


Figure 21 - 39 Pressure and Performance at 90ºF (32ºC)

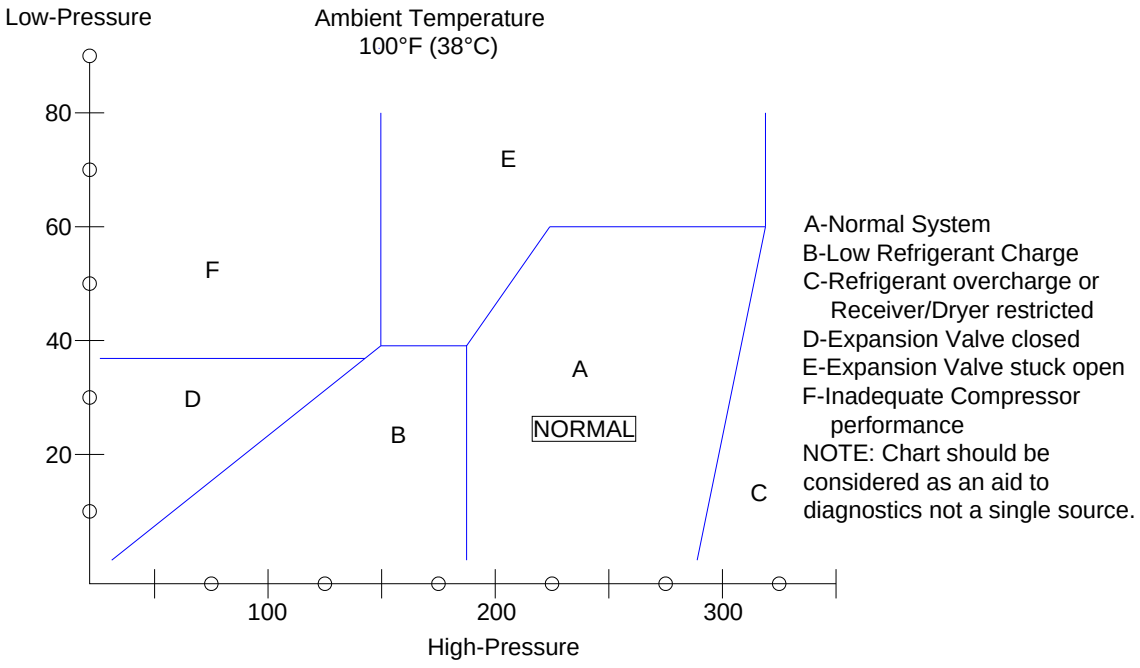


Figure 21 - 40 Pressure and Performance at 100ºF (38ºC)

CHAPTER 22

AUTO FLIGHT

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S-TEC 55X AUTOPILOT WITH AUTOTRIM

22-1. GENERAL

a. **System Overview** – The S-Tec 55X autopilot is a two axis system that controls movement around the pitch and roll axes. Information for movement around the roll axis is obtained from the turn coordinator of the airplane. The roll axis of the System 55X has heading select, VOR/Localizer front and back course intercept and tracking. It is also interfaced with the GPS and Nav system that provides standard autopilot outputs.

Figure 22 - 1 S-Tec 55X Control Panel

b. The System 55X derives pitch axis information from a solid-state absolute pressure transducer and an internal, sensitive accelerometer that provide precise pitch axis altitude hold, automatic/manual glideslope intercept and capture, and vertical speed commands totally independent of the aircraft's artificial horizon gyro. The System 55X autopilot delivers accurate altitude, vertical speed, and vertical acceleration data to the system's pitch computer, regardless of aircraft flight attitude.

22-2. THE SYSTEM 55X FLIGHT GUIDANCE PROGRAMMER/COMPUTER

a. The System 55X Flight Guidance Programmer/Computer serves the function of converting pilot commands to logic signals for the roll and pitch computer functions. As the pilot enters the desired mode by pressing the appropriate mode selector switch, the computer acknowledges the mode, causing the appropriate annunciator to illuminate.

b. The roll computer receives signal inputs from the Directional Gyro or HSI, VOR/LOC and GPS deviation indicators, and the turn coordinator. The roll computer computes roll servo commands for stabilization, turns, radial intercepts, and tracking.

c. The pitch computer receives signal inputs from the altitude pressure transducer; accelerometer; glideslope deviation indicator; and vertical speed modifier control, optional altitude selector/alerter, or altitude/vertical speed selector if installed. The pitch computer computes pitch servo commands for vertical speed, altitude hold, and glide slope intercept and tracking. Sensing for trim annunciation or automatic elevator trim is provided by the pitch servo. Drive for the elevator trim servo is provided by the pitch computer. See Figure 22 - 2 for a diagram of the autopilot system.

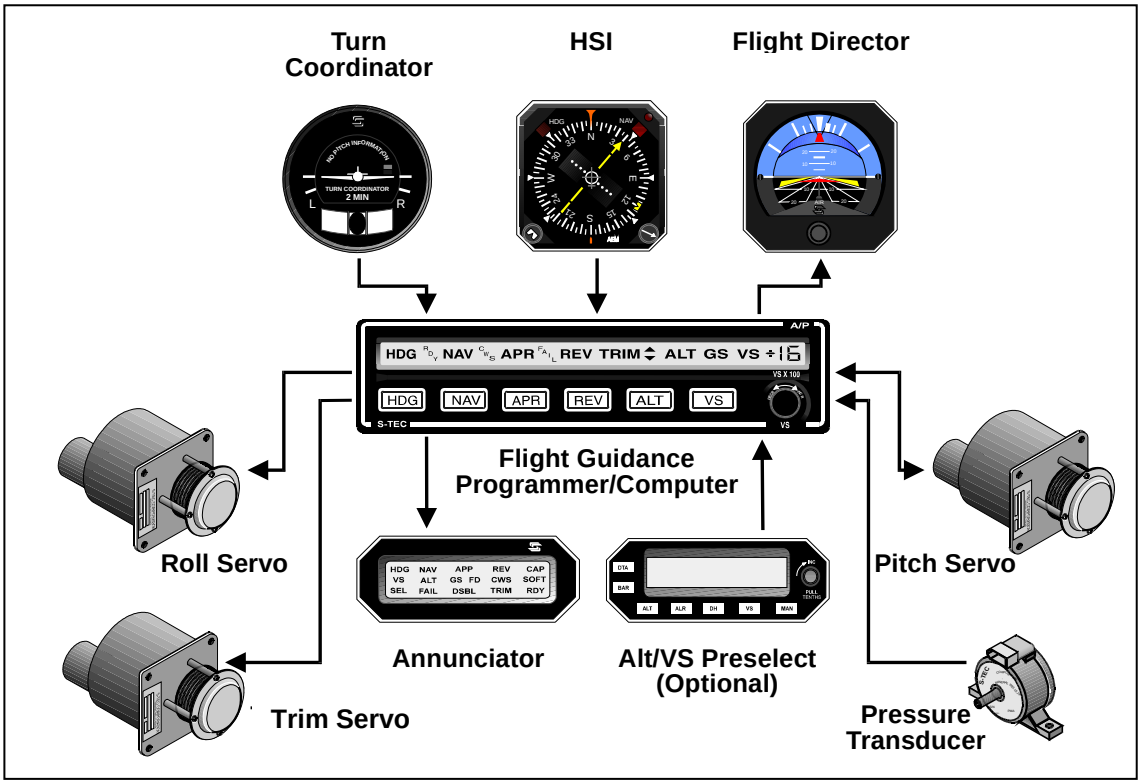

Figure 22 - 2 S-Tec 55X Autopilot System

22-3. PREFLIGHT SELF-TEST PROCEDURE (S-TEC 55X AUTOPILOT)

The System 55X incorporates a self-test that must pass before the autopilot can be engaged. To perform the test, aircraft DC electrical power must be on and the avionics master on. Place the Autopilot Master switch to the on position, and observe that all annunciators of the Programmer/Computer and remote display illuminate for approximately five seconds, then extinguish. When the system is first turned on, the Flight Guidance Programmer/Computer and Annunciator will display all functions as shown in Figure 22 - 1. When the self-test is complete, a RDY (ready) annunciation is displayed. Should a fault be detected, the FAIL annunciator will remain on and the auto pilot will not operate.

22-4. CHECKING THE AUTOTRIM SYSTEM (S-TEC 55X)

a. Place the Trim Master Switch in the on position.
b. Press and release the HDG and VS switches and ensure that HDG and VS illuminate on the annunciator.
c. Rotate the VS knob clockwise. The control stick should move slowly out (pilot may have to assist a heavy yoke).
d. Rotate the VS knob counterclockwise. The control stick should move slowly in.
e. Press the autopilot disconnect switch (APD) on the control stick (see Figure 22 - 3) and verify that the autopilot disconnects.
f. Engage HDG mode and move the DG or HSI HDG bug left and right. The control stick should follow the HDG bug.

1. If HSI equipped, center the course arrow under the lubber line and push the NAV button. Move the course arrow on the HSI left then right. The control stick should follow the course arrow. Channel a valid VOR signal and move course arrow just enough to deflect the left/right needle 1 or 2 dots. The control stick should follow the Course Deviation Indicator (CDI) left/right needle during the test. (This test is only valid if the left/right needle is centered with the course arrow under the lubber line)
2. If DG equipped, center the HDG bug under the lubber line. Channel a valid **VOR** signal. Move the OBS to cause left/right CDI needle deflection. The control stick should follow the left / right needle movement.

g. Press the REV mode switch. The control stick should respond opposite to the course arrow and CDI left/right needle inputs.
h. Press the ALT (Altitude Hold) switch. Slowly pull aft (nose up) on the control stick. Autotrim should run nose down with TRIM flashing on the remote annunciator and the autopilot computer/programmer after approximately 3 seconds. Slowly move the control stick forward (nose down). After 3 seconds, Autotrim should move nose up with TRIM flashing on the remote annunciator and the autopilot computer/programmer after approximately 3 seconds.
i. Place the Trim Master Switch in the off position.

22-5. MANUAL ELECTRIC TRIM TEST

a. Place the Trim Master switch in the on position.
b. Move the elevator trim switch (hat switch on the control stick) up and down. The autopilot should disengage. The *RDY* (ready) annunciation will flash, then display a steady indication, and the trim should operate in the commanded direction. (The elevator trim switch will disengage the autopilot only when a pitch mode is engaged.) Re-engage the heading and VS modes, and press the Autopilot disconnect switch (APD). The autopilot should disengage. As before, the *RDY* indication will flash, then annunciate steady. An audible tone should be heard when the autopilot is disconnected.

22-6. IF FLIGHT DIRECTOR EQUIPPED

a. Place the Autopilot Master Switch in SELECT FD position. Verify the roll, pitch, and trim servos are disengaged. The steering bar should be in view on the attitude indicator.
b. Engage HDG mode. Move the HDG bug 45 degrees left. The roll steering bar should slowly indicate a left steering command. Repeat the same test for the right side.
c. Engage VS mode. Select 1500 FPM rate of climb. Verify the pitch steering bar moves slowly up. Repeat the same test for the down direction.
d. Place the Autopilot Master Switch in SELECT FD/AP. The servos should re-engage.
e. Place the Trim Master switch in the on position.
f. Place the Manual Electric Trim Command switch to MOVE FWD or AFT. The autopilot should disconnect.
g. The Manual Electric Trim Command switch will disconnect the autopilot only if there is a Pitch Mode engaged.

22-7. AUTOPILOT DISCONNECT SWITCH

a. The autopilot disconnect switch is a spring loaded rocker switch on the top left side of the pilot's control stick and is normally operated with the thumb of the left hand. Pressing either the top or the bottom portion of the rocker switch will disengage the autopilot. (Note: Operating the elevator trim switch will also disconnect the autopilot.)

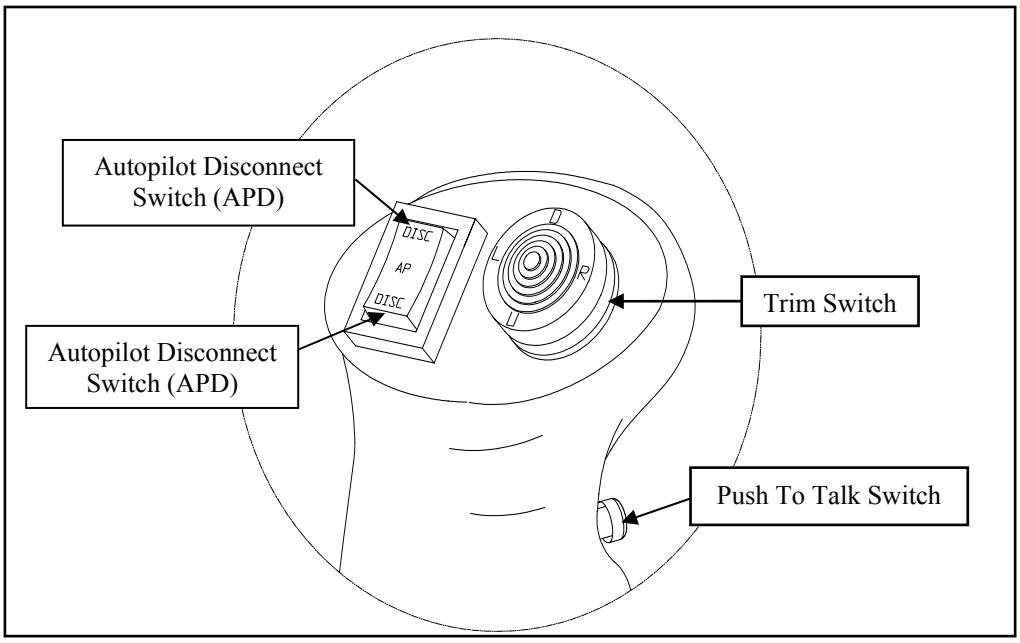

Figure 22 - 3 Autopilot Switches on the Control Stick

22-8. S-TEC 360 AUTOPILOT ALTITUDE PRESELECT (Optional Equipment)

a. **General Overview** – The S-Tec 360 Autopilot Altitude Preselect (AAP) enhances use of the S-Tec System 55X Autopilot, but is not a required component of the autopilot system. While the S-Tec 360 has a number of special functions like altitude and decision height alerting, the primary features of the AAP is the ability to preselect an assigned or desired altitude. When the altitude command is sent to the autopilot, the airplane will climb or descend at a preset rate to the preset altitude and thereafter maintain that altitude.

b. It is important to understand that the S-Tec 360 unit does not control the autopilot. Vertical speed and altitude commands for the autopilot are preprogrammed into the AAP unit but are not commanded until the corresponding functions on the autopilot display unit are engaged. The two primary advantages of the AAP unit are (1) the ability to preprogram altitude and climb settings for later execution, and (2) the barometric calibration feature discussed in the next paragraph.

c. Altitude information is obtained from the encoder altimeter of the airplane. The AAP unit has a "baro calibration" feature, which allows the pilot to correct the pressure altitude from the encoder for local pressure variations. Essentially, there are two Kollsman displays in the airplane, the static system altimeter and the S-Tec 360 display unit. If the encoding altimeter is properly calibrated and the pilot routinely updates the altimeter settings of the AAP unit, the autopilot will maintain the correct altitude above sea level.

d. The method of altitude hold is different when using the altitude hold feature of the S-Tec System 55X autopilot. When the altitude hold function is engaged on the autopilot without input from the AAP, the autopilot senses the current atmospheric pressure at the instant altitude hold is engaged, and thereafter, maneuvers the airplane to maintain that pressure. If the airplane flies into a low pressure area, the autopilot will command the airplane to descend. In this situation, the pilot must use the vertical speed control on the autopilot to compensate for the normal en route and diurnal changes in atmospheric pressure.

e. The S-Tec System 55X Autopilot and 360 AAP both have altitude (ALT) and vertical speed (VS) functions. During the study of the pilot's guide on the following pages, it is important to distinguish which unit the ALT or VS discussion is applicable. If the term **autopilot** is used, this refers to the buttons on the S-Tec System 55X autopilot. If the terms, **Selector**, **Selector Unit**, or **AAP** are used, it refers to the S-Tec 360 unit shown in Figure 22 - 5. In general, the AAP is used to **select**, **input**, or make altitude and/or vertical speed commands **active**. The autopilot, when **engaged**, executes the commands. Understanding the meaning of a few terms that are used in the S-Tec Pilot's Guide is helpful.

   1. The term **selected altitude** refers to the altitude selected or preset in the AAP Unit for the autopilot to intercept and/or maintain.
   2. The **operate mode** is the normal state of the S-Tec 360 system. In this mode, preset ALT and VS commands are executed when the pilot engages the corresponding commands on the autopilot. The **Entry Mode** is used to input altitude, or other related settings.

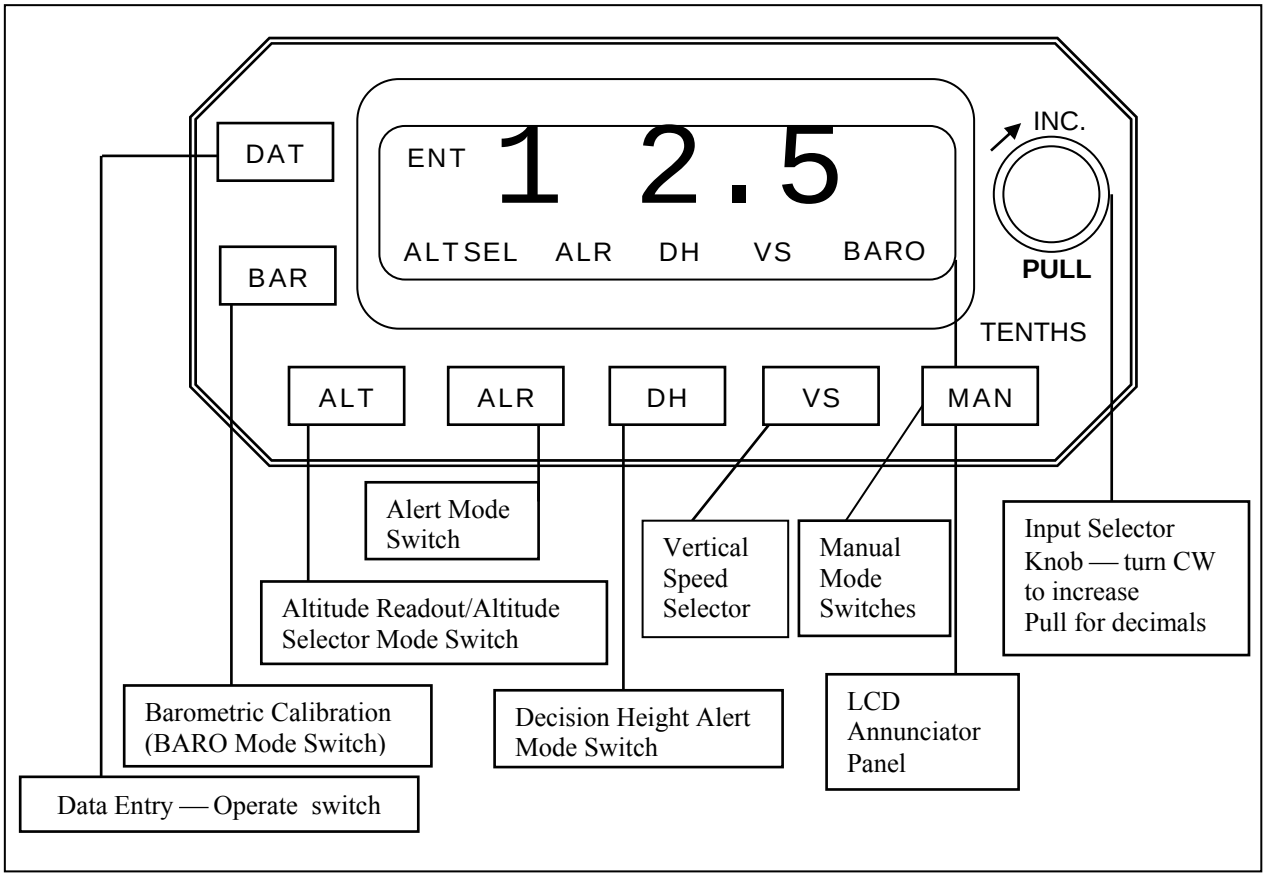

Figure 22 - 4 S-Tec 360 AAP

f. **S-Tec 360 System Description –** The S-Tec Liquid Crystal Display (LCD) Altitude and Vertical Speed Selector are designed for use with S-Tec 55X Autopilot. The system is used to pre-select an altitude, a rate of climb, or a rate of descent, which is then performed by the autopilot. In addition, the selector provides an altitude alert mode, a decision height (DH) alert mode, an altitude readout from the encoder, barometric calibration, and a manual mode. This supplemental section provides information on the features and functions of the system and operating instructions for its proper use. A labeled drawing of the display unit is shown in Figure 22 - 4.

g. The LCD Altitude and Vertical Speed Selectors are integrated into a single panel-mounted unit, which contains the display, the operating switches, and the computer electronics. The system is designed to interface with the autopilot and the encoder, which provides a standard altitude output in increments of 100 feet.

22-9. S-TEC 360 THEORY OF OPERATION

a. **Encoder Interface –** The Altitude Selector Computer reads and decodes raw altitude information from the altitude encoder. The uncorrected altitude information is adjusted by changing the setting in the barometric calibration. When the information from the selected altitude matches the corrected decoded altitude information from the encoder, the altitude selector computer signals the autopilot to electrically engage the altitude hold mode of the autopilot.

b. The vertical speed selector provides an electrical output to the autopilot pitch flight guidance computer that is proportional to the amplitude and polarity (direction) of the vertical speed. For example, + 500 FPM climb VS would produce a plus (+) voltage in an amount representing 500 FPM. The autopilot compares the existing vertical speed with the selected vertical speed and maneuvers the airplane to match these signals.

c. **System Diagram –** Figure 22 - 5 contains a block display of how the various avionics components function with the S-Tec 360 system.

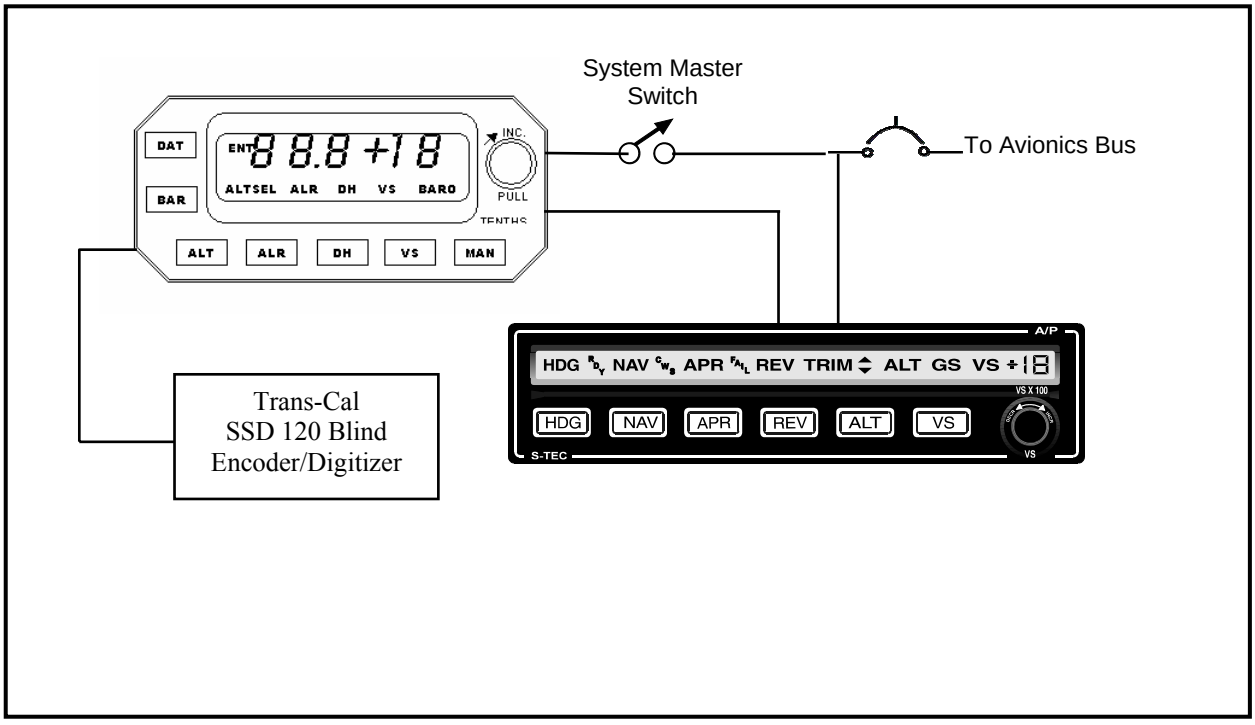

Figure 22 - 5 Block Diagram of S-Tec 360 System

22-10. S-TEC 360 GROUND TESTING

a. **Self-Test –** When power is applied to the system, an internal self-test of the computer electronic elements, the display, annunciators, and the altitude alerter audio tone is conducted. Successful test conclusion is indicated by a display of 29.9 in the selector unit. The self-test cycle does **not** check the encoder for proper operation. However, the preflight check procedures outlined in Chapter 34 provide a method to determine proper operation of the encoder.

b. **Power Switches** – There are no separate on/off switches for the S-Tec 360 Autopilot Altitude Preselect. The S-Tec 360 is powered when the system and avionics master switches are on. In addition, the Trans call SSD 120 Blind Encoder/Digitizer does not have a power switch. The encoder is turned on by the avionics master switch. There are certain operational limitations of the encoder which are discussed in Chapter 34.

c. **Preflight Inspections –** The following preflight procedure provides an operational test of the entire system, including the encoder, the altitude selector, and the autopilot. A successful test is indicated by the autopilot switching from VS Mode to ALT Hold Mode as the selected altitude is matched to field elevation.

1. System, Autopilot, and Avionics Master Switches – ON
2. Autopilot ALT Select Button – ON
3. Altimeter – Set to local altimeter setting or field elevation, as appropriate.
4. Altitude Selector –
   a) Observe that the self-test cycle is complete. When first powered, the system will display all annunciations for approximately five seconds, ending with an audible tone. Thereafter, the system will display a barometric setting of 29.9 with the barometric annunciator flashing.
   b) Rotate the selector input knob to set barometric setting to the nearest 0.1 in. of Hg (for millibars push on barometric switch).
   c) Push the ALT Switch to display ALT SEL. With the SEL flashing, rotate the selector knob to input an altitude 300 to 400 ft. lower or higher than the indicated altitude.
   d) Push the VS Switch to activate VS Selector, rotate the selector switch knob to input desired climb (+) or descent (-) speed.
   e) Push the ALT Switch to access the altitude set mode - ALT SEL will be annunciated.
5. Autopilot
   a) Engage the HDG Mode
   b) Simultaneously depress VS and ALT switches on the autopilot, and observe that the VS and ALT annunciators both illuminate.
   c) Rotate the altitude selector knob on the AAP unit and change the selected altitude so that it matches the field elevation. The VS annunciation on the autopilot should extinguish when the setting on the altitude selector is within 100 ft. of the altitude indicated on the altimeter. Extinguishing of the VS annunciation with the ALT remaining on indicates the altitude hold mode is engaged. If altitude engagement does not occur within 100 ft. of indicated altitude, readjust the barometric calibration.
6. Disengage Autopilot – Adjust the altitude selector to the desired cruise altitude and set the vertical speed to an appropriate climb setting for use after takeoff.

7. Autopilot Preflight – Conduct autopilot preflight per the instructions in S-Tec System 55X Pilot's Guide.

22-11. S-TEC AUTOPILOT INTEGRATION WITH AVIDYNE FLIGHTMAX ENTEGRA PRIMARY FLIGHT DISPLAY (PFD)

Whenever the S-Tec System 55X Autopilot or the Avidyne PFD are replaced, perform setup procedures indicated in Chapter 34.

22-12. INSTRUCTIONS FOR CONTINUED AIRWORTHINESS (S-Tec Corp.)

a. **Description –** The S-Tec flight control system includes the following items:
   1. Roll servo
   2. Pitch servo
   3. Trim servo (elevator)

b. **Installed Equipment –** Panel or remotely mounted pitch/roll/yaw computers include the following:
   1. Panel mounted controllers, indicators, switches, and breakers
   2. Barometric pressure transducer/static source

c. **Servos –** Servo installations utilize aluminum brackets to secure the servos to the airframe. Attachment to the aircraft primary flight controls and trim systems is accomplished through cable and push rod assemblies.

d. **Controls Operation Information –** Operation of the autopilot and other systems is described in FAA Approved S-TEC Flight Manual Supplement P/N 891830S. Specialized controls, annunciation, operation and interpretation are covered in this required document and in S-TEC Pilot's Operating Handbook S-TEC P/N 87109 that supplements the approved AFMS. This information is also contained in the POH/AFM.

e. **Servicing Information –** Approved S-TEC dealers holding the appropriate FAA certification must accomplish all servicing of items. Locations and access to the components installed are described and depicted in the installation drawings and Installation Instructions ST- 852. Removal and replacement of components should be determined by functional checks indicated in S-TEC AFM Supplement 891830S, Pilot's Operating Handbook document number, 87109 (55X) and the Post Installation Ground Checks document number 05110.

f. **Maintenance Instructions –** Condition and/or airworthiness inspections required under FAR Part 43, or other FAA approved programs, should include several items regarding the S-TEC autopilot system installed in the aircraft.
   1. Visual Inspections
      a) Flight Control Components – Check control cables, pulleys, servos, and associated equipment for condition, attachment, clearance, and proper operation. Replace cables that have broken strands or evidence of corrosion. Check cables for proper tension.
      b) Electrical Wiring and Components – Check for security, chafing, damage and attachment. For replacement of any of the autopilot wiring, cables or associated components, see the Master Drawing List 921139 for the appropriate drawing or document associated with the action to be performed.
   2. System Operation Check after Maintenance – See sections 22-3, 22-4, 22-5, and 22-6

22-13. TROUBLESHOOTING INFORMATION – Trouble-shooting this equipment should only be accomplished by authorized S-TEC Dealers, holding the appropriate FAA certification, with required test equipment and service data. System function should be determined through functional checks indicated in the AFM Supplement 891830S, Pilot's Operating Handbook document number 87109, and the Post Installation Ground Checks document number 05110.

22-14. SPECIAL ISSUES

a. **Removal and Replacement Information –** All components can be removed with common tools and practices.

b. **Diagrams/Special Inspections –** See Master Drawing List 921139., for all diagrams. There are no special inspection requirements.

c. **Application of Protective Treatments –** Servos and transducers should be removed prior to application of corrosion (or other) treatments. Panel mounted components should not be exposed to these treatments.

d. **Structural Fasteners –** See S-Tec document No. 761017, Installation, S-Tec System 55/55X Autopilot

e. **Special Tools**

1. **Crimp Tools**
   A crimp tool and positioner/locator meeting MIL Specification M22520/1-01 or equivalent are required to ensure consistent, reliable crimp contact connections for the rear d-sub connectors. These tools are available from ITT Cannon or other vendors:

Insertion Tool:	ITT part#274-7048-000 (Desc. CIET-22D-01)
Crimp Tool (HD):	ITT part#995-0001-584 (Desc. M22520/2-01)
Locator Tool:	ITT part#995-0001-244 (Desc. M22520/2-07)
Locator Tool (HD):	ITT part#995-0001-739 (Desc, M22520/2-06)
Locator Tool (HD):	Desc. M22520/2-09

2. **S-TEC Special Tool**

Clutch Adjustment Spanner	Part Number 6622	Drawing No. 6622 & 1
Clutch Adjustment Spanner	Part Number 66228	Drawing No. 66228 & 1N/A

f. **Normal Category Aircraft –** Electrical load must be maintained within 80% of generator capacity for systems installed (AC 43.13-1B). Electrical loads are described in the Component Weights and Current Drain section of the S-Tec Installation Instructions ST-852.

g. **Overhaul Time Limitations –** N/A

h. **Airworthiness Limitations –** Limitations are listed in the Limitations Section of AFM Supplement(s).

i. **Revisions –** The S-Tec Service Letter program will be utilized to inform aircraft operators of significant changes to this ICA. Contact S-Tec Corporation at 1-800-USA-STEC.

22-15. AUTOPILOT ROLL SERVO

a. **General –** The autopilot roll servo is located under the flap drive bracket. It is accessed through the wing controls access panel located under the fuselage at the aft end of the wing saddle, near the wing trailing edge. The servo is attached to the bracket at an angle. This angle aligns the servo drive capstan grooves with the bridle cable. The bridle cable is attached to the aileron interconnect pushrod with two saddle assemblies. See Chapter 27 for a discussion of the aileron system. See Figure 22 - 6 for a picture of the aileron roll servo.

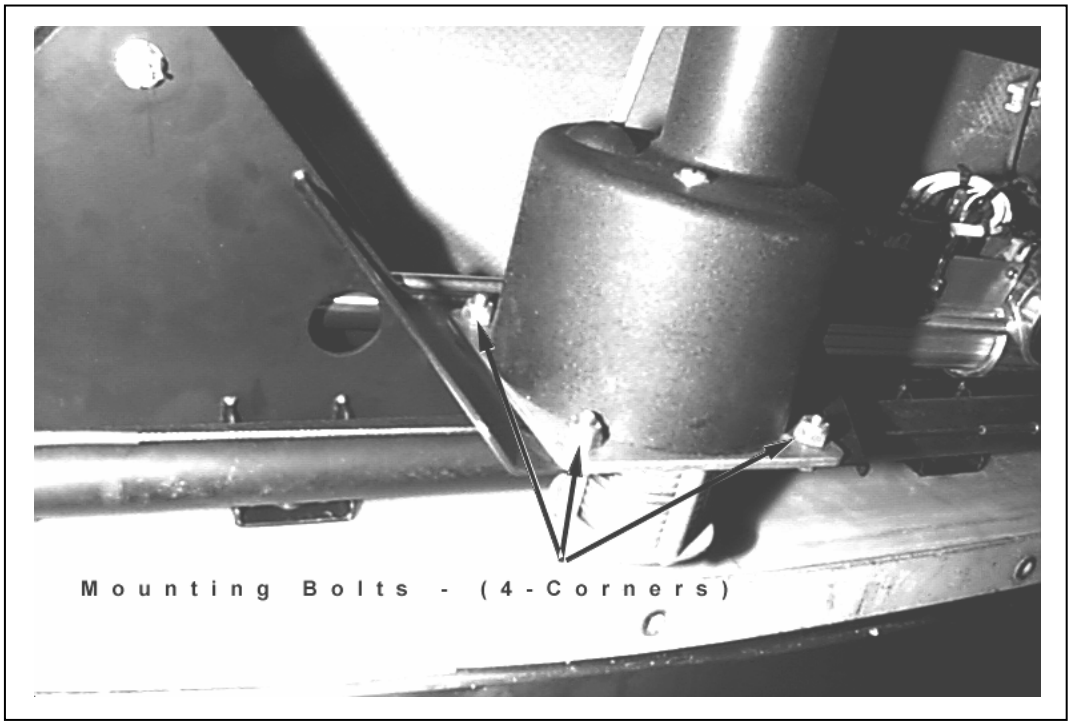

Figure 22 - 6 Autopilot Roll Servo (View looking up)

b. **Removal** – The roll servo bridle cable must be detached to remove the roll servo. Remove the MS-21256-1 locking clips from the barrel of the turnbuckle. Detach this cable from both ends by loosening the turnbuckle and removing the clevis pins. Retain the MS20392-2C-11 clevis pin and NAS1149F0332P washer for reuse. A new MS24665-134 cotter pin will be required for reinstallation.

1. **Bridle Cable Removal and Inspection** – The bridle cable does not have to be removed from the servo drive capstan for most maintenance procedures. Check cable for signs of wear or corrosion. Replace cable assembly if required. Autopilot installation bulletins must be followed for this process.
2. **Unplug Roll Servo** – Cut the nylon tie used to secure the electrical wiring. Unplug the autopilot roll servo electrical connection at the connector.
3. **Servo Removal** – The servo is attached to the bracket with four AN3-4A bolts and MS20365-1032 nuts with NAS1149F0332P washers. These are located on the corners of the servo attach flange. Three of these bolts can be seen in Figure 22 - 6. When the

four AN3-4A bolts and MS20365-1032 nuts with NAS1149F0332P washers are removed, the servo can be removed.

c. **Servo Installation** – Installation is the reverse of removal.
   1. Adjust bridle cable tension to proper 30 lbs. ± 2 lbs. value.
   2. The four AN3-4A bolts and MS20365-1032 nuts are torqued 30 to 36 in.-lbs.
   3. Be sure to check cable tension and installation per autopilot installation requirements.

22-16. AUTOPILOT PITCH SERVO

a. **General** – The autopilot pitch servo is located under the baggage bulkhead. It is accessed through the baggage bulkhead access panel located under the upholstery on the hat rack. The servo is attached to a composite bracket mounted at an angle to the fuselage centerline. This angle aligns the servo drive capstan grooves with the bridle cable. The bridle cable is attached to the elevator final drive pushrod with two saddle assemblies.

b. **Removal** – The pitch servo bridle cable must be detached to remove the roll servo. Remove the MS-21256-1 locking clips from the barrel of the turnbuckle. Detach this cable from both ends by loosening the turnbuckle and removing the clevis pins. Retain the MS20392-2C-11 clevis pin and NAS1149F0332P washer for reuse. A new MS24665-134 cotter pin will be required for reinstallation.

Figure 22 - 7 Autopilot Pitch Servo (View looking down)

1. **Bridle Cable Removal and Inspection** – The bridle cable does not have to be removed from the servo drive capstan for most work. Check cable for signs of wear or corrosion. Replace cable assembly if required. Autopilot installation bulletins must be followed for this process.
2. **Unplug Pitch Servo** – Cut the nylon tie used to secure the electrical wiring. Unplug the autopilot pitch servo electrical connection at the connector.
3. **Servo Removal** – The servo is attached to the bracket with two MS24694-S50 flat head screw on the upper two mounting locations and two AN3-4A bolts with NAS1149F0363P washers on the lower two mounting locations. Each screw and bolt is installed into a clip nut (ESNA RM52LHA4972-8-02) attached to the servo mounting flange. These mounting bolts are located on the corners of the servo attach flange, and the bolts can be seen in Figure 22 - 8. Remove the servo.

c. **Servo Installation** – Installation is the reverse of the removal.
1. Adjust bridle cable tension to proper 30 lbs. ± 2 lbs. value.
2. Torque all bolts 30 to 36 in.-lbs.
3. Be sure to check cable tension and installation per autopilot installation requirements.

Figure 22 - 8 Autopilot Pitch Servo

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GARMIN GFC 700 AUTOPILOT

22-17. GENERAL

The GFC 700 is a two-axis fail-safe digital flight control system. There is no single LRU with the name "GFC 700;" rather "GFC 700" refers to an integrated autopilot and flight director system, with functions provided by multiple G1000 LRUs and servos. See Figure 22 - 9. The following functions are provided by the GFC 700 in this installation:

- Flight Director
- Autopilot
- Manual Electric Trim

a. **Flight Director –** The Flight Director operates within the #1 GIA 63/GIA 63 W and uses data from the G1000 system, including air data, attitude, and navigation data, to calculate commands for display to the pilot and for the Autopilot. Flight Director command bars and mode annunciations are sent to the PFD through a high-speed Ethernet connection for display to the pilot. The Flight Director operates independently of the Autopilot, and allows the pilot to hand-fly the command bars, if desired.

b. **Autopilot**: **–** The Autopilot operates within the two GSA 81 servos. Flight Director data is processed within the two servos and turned into aircraft flight control surface commands. The autopilot cannot operate unless the Flight Director is engaged.

The following is a summary of the autopilot functions provided by each LRU:

- GDU 1040 PFD – Displays the Flight Director command bars and the autopilot mode annunciations.
- GDU 1042/GDU 1044 MFD – Provides controls for the autopilot functions.
- GIA 63/GIA 63W – Performs the calculations to display the Flight Director command bars on the PFD and sends control movement commands to the GSA 81 autopilot servos.
- GSA 81 – Actuates the control surfaces based on commands received from the GIA 63/GIA 63W.
- GSA 82 Trim Adapter – Operates the elevator trim tab.

c. **Configuration and Testing –** After removal and replacement, LRUs must be configured and tested per the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

CAUTION

When removing and/or replacing any G1000 component, always ensure that aircraft power is off. Unplug any auxiliary power supplies.

Before removing any G1000 LRU, it is required that the technician verify the LRU software part number and version against the Required Equipment List RB011002.

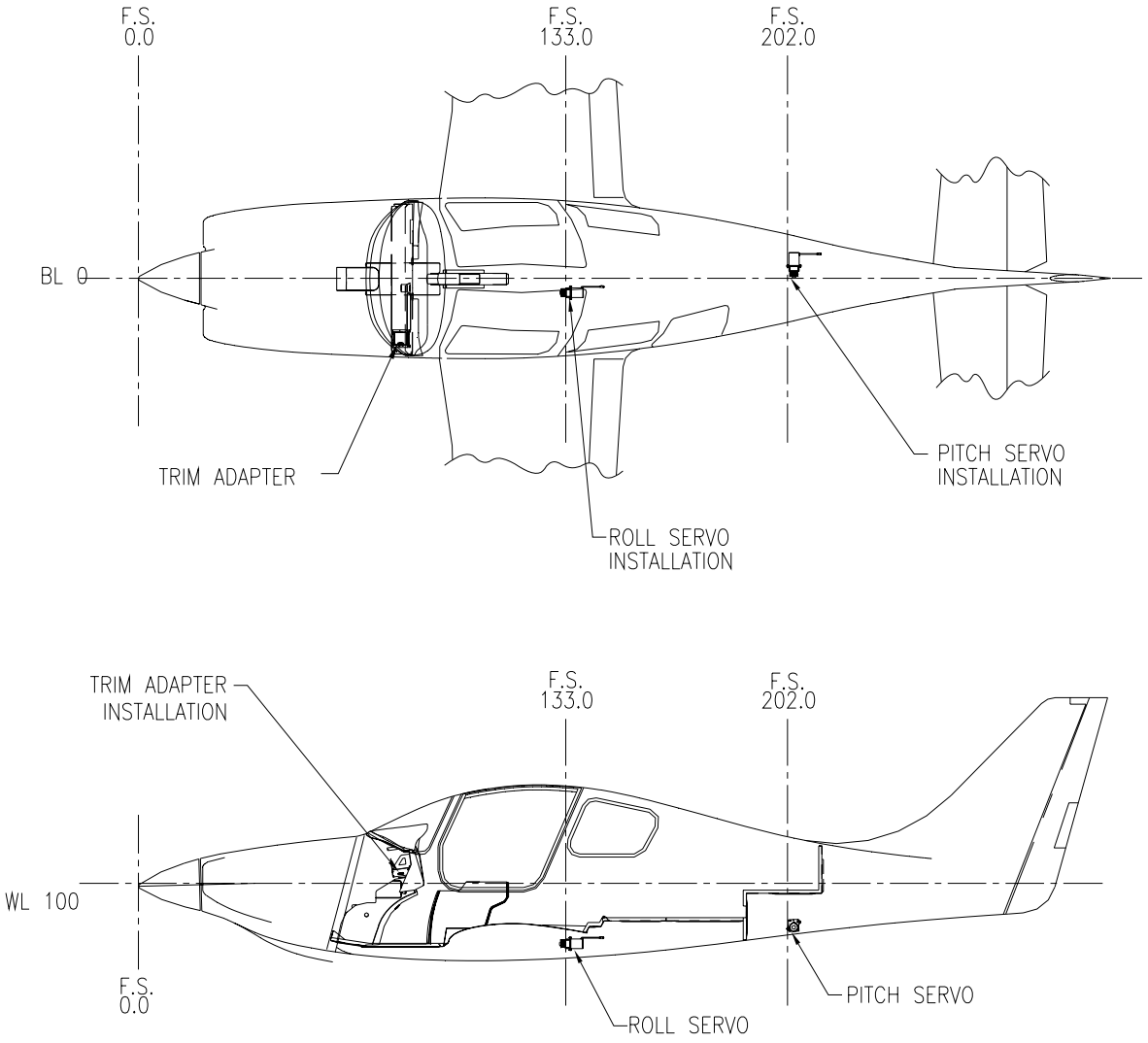

Figure 22 - 9 GFC 700 Autopilot

22-18. AUTOPILOT DISCONNECT SWITCH

The autopilot disconnect switch is a push button switch on the top left side of the pilot's control stick and the top right side of the co-pilot's control stick, and is normally operated with the thumb. Pressing the switch will disengage the autopilot and trim system. (Note: Operating the elevator trim switch in the overhead will also disconnect the autopilot.)

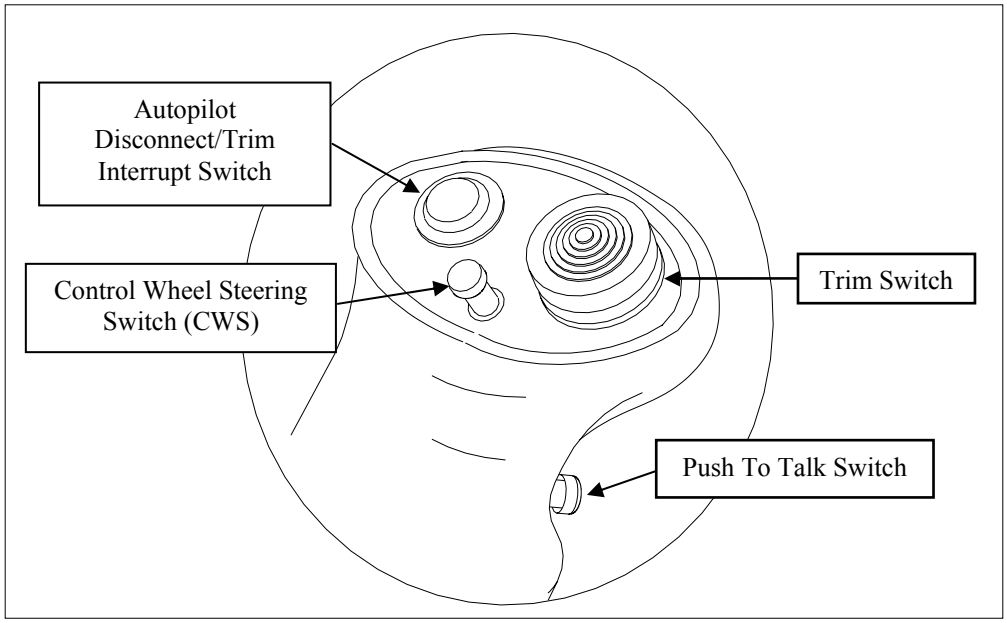

Figure 22 - 10 Autopilot Switch on the Control Stick

22-19. TROUBLESHOOTING

Troubleshoot the GFC 700 system per the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

22-20. GSA 81 AUTOPILOT ROLL SERVO (See Figure 22 - 11)

The GSA 81 is mated to the GSM 85 Servo Mount to form a single servo unit

The design of the servo assembly allows the servo portion (GSA 81) to be removed from the capstan (GSM 85) without the need to de-rig the aircraft control cables. The roll servo is located beneath the rear passenger seats.

a. **Removal**

1. Ensure aircraft power is off. Unplug any auxiliary power supplies.
2. On the underside of the wings at the fuselage remove the Flap and Aileron Push-Rod Access Panel.
3. Disconnect servo connector P680.
4. Remove the two bolts holding the servo to the servo mount. See Figure 22 - 12.
5. Remove the unit.
6. Remove excess grease build-up from the single servo output gear using a lint-free cloth.

NOTE

It is not necessary to remove all of the grease from the output gear, only the excess grease. D0 NOT USE SOLVENTS TO CLEAN THE OUTPUT GEAR!

7. Using a brush or other applicator, apply a thin coat of grease to the servo output gear. Use Aeroshell 17 or equivalent (Synthetic Diester, Low Temp; Must meet MIL-G-21164D).

b. **Installation**

1. Set unit in place.
2. Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
3. Install the two bolts holding the servo to the servo mount.
4. Connect servo connector P680.
5. Configure and test the GSA 81 per the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.
6. Operate all controls through full travel and ensure that no binding or restriction occurs.
7. Reinstall the Flap and Aileron Push-Rod Access Panel.

22-21. GSM 85 AUTOPILOT ROLL SERVO MOUNT

a. **Removal**

1. Remove the GSA 81 servo per paragraph 22-20.
2. Loosen the bridle cable turnbuckle and remove the clevis pins.
3. Remove the four screws holding the mount to the servo bracket.
4. Remove the unit with the bridle cable

b. **Installation**

1. Prior to installation, adjust and verify the slip clutch torque is set to 24 ± 4 in-lb. for the roll servo mount or 50 ± 7 in-lb. for the pitch servo mount.
   (a) Place the servo mount onto the Garmin Servo Adjustment Fixture (011-01085-XX) and secure with toggle clamps. See Figure 22 - 13.
   (b) Install the cable, supplied with the fixture, between the capstan under test and the fixture capstan.
   (c) Remove slack in the cable with the tension adjust knob.
   (d) Tighten the thumbscrews on the fixture capstan.
   (e) Attach a socket to a calibrated torque wrench of the appropriate range and place the assembly on top of the fixture capstan. Adjust the wrench support as required to support the torque wrench in a level, horizontal position. Ensure the torque wrench is set to “ZERO” while the motor and solenoid are “OFF”.
   (f) Connect a 24 VDC power supply to the fixture.
   (g) Place the solenoid switch to “ON” and move the direction switch to the “CW” or “CCW” position. Allow the capstan to rotate about 1 revolution in each direction. Adjust the castle nut on the servo mount until the torque value indicates 24 ± 4 in-lb. for the roll servo mount or 50 ± 7 in-lb. for the pitch servo mount.

(h) Switch the motor direction back and forth as necessary to allow adjustment until the required torque range is achieved in both directions. Ensure the castle nut aligns with one of the holes in the shaft.

(i) Observe the slip clutch torque reading for at least two full revolutions in each direction.

(j) Ensure the minimum and maximum torque readings are within tolerance of 24 ± 4 in-lb. for the roll servo mount or 50 ± 7 in-lb. for the pitch servo mount.

(k) When finished, remove the servo mount from the servo adjustment fixture.

1. Install a new cotter pin through the capstan nut and servo mount shaft.
2. Set the unit in place.
3. Install the four screws holding the mount to the servo bracket.
4. Install the bridle cable. Wrap the turnbuckle end approximately 270º over the capstan. Wrap the fork end approximately 450º over the capstan. Set the cable tension to 25 to 60 lbs. The bridle cable should make two complete revolutions around the capstan and the lock ball should be mated in its seat. System rigging to be accomplished with controls neutral.
5. Safety wire the turnbuckle using a single wrap method with .040 safety wire or a double wrap method using .030 safety wire.
6. Install the GSA 81 servo per paragraph 22-20.

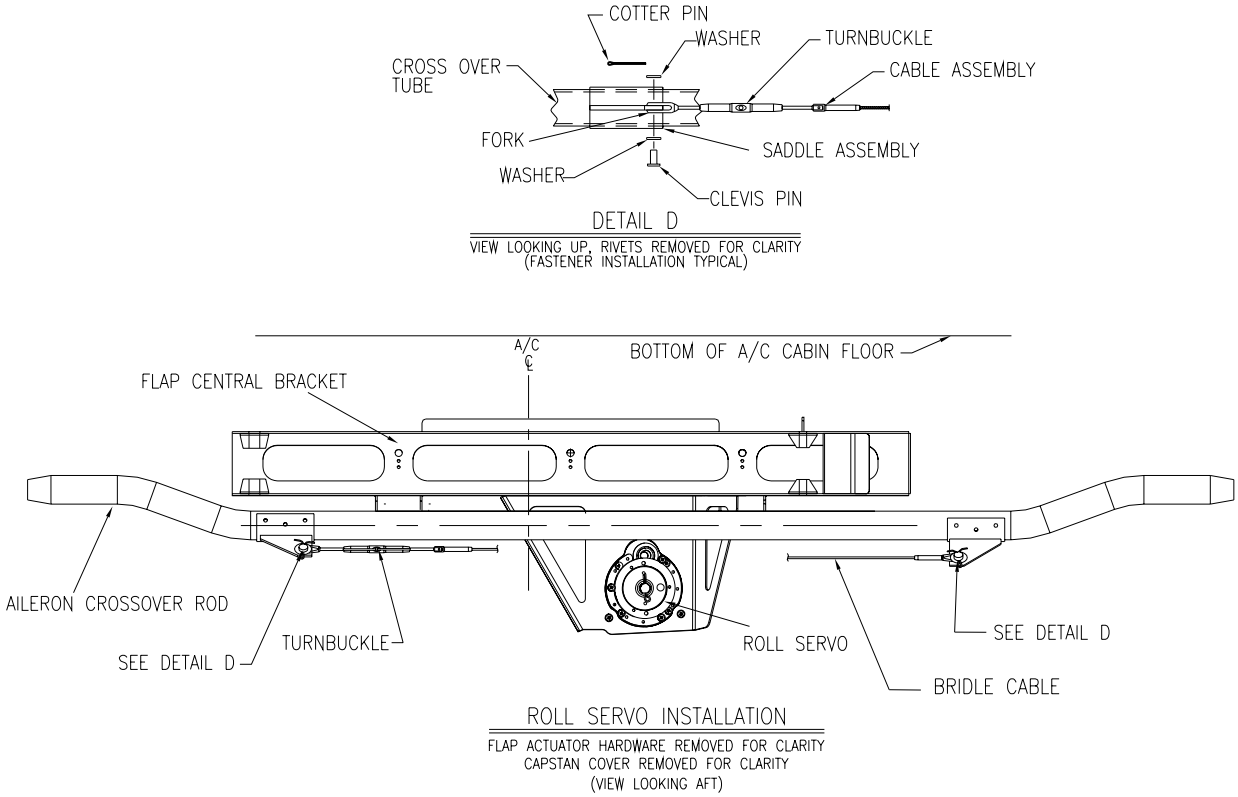

Figure 22 - 11 Roll Servo

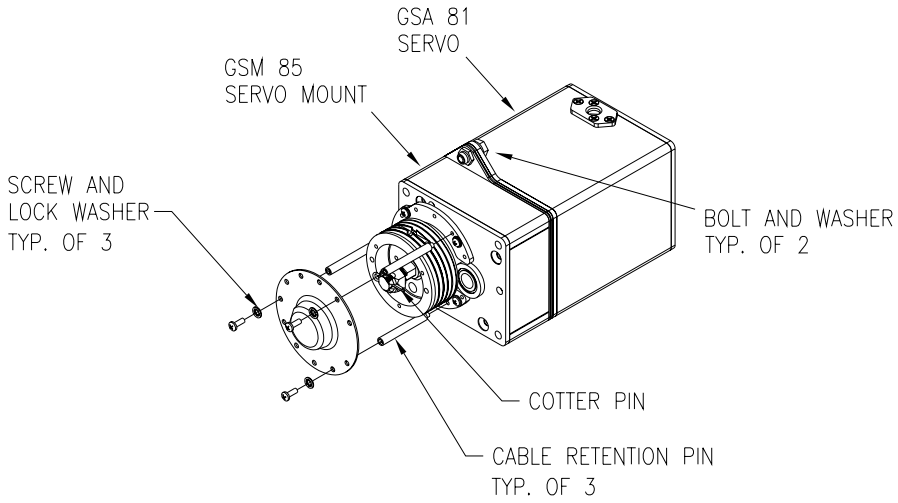

Figure 22 - 12 Servo Assembly

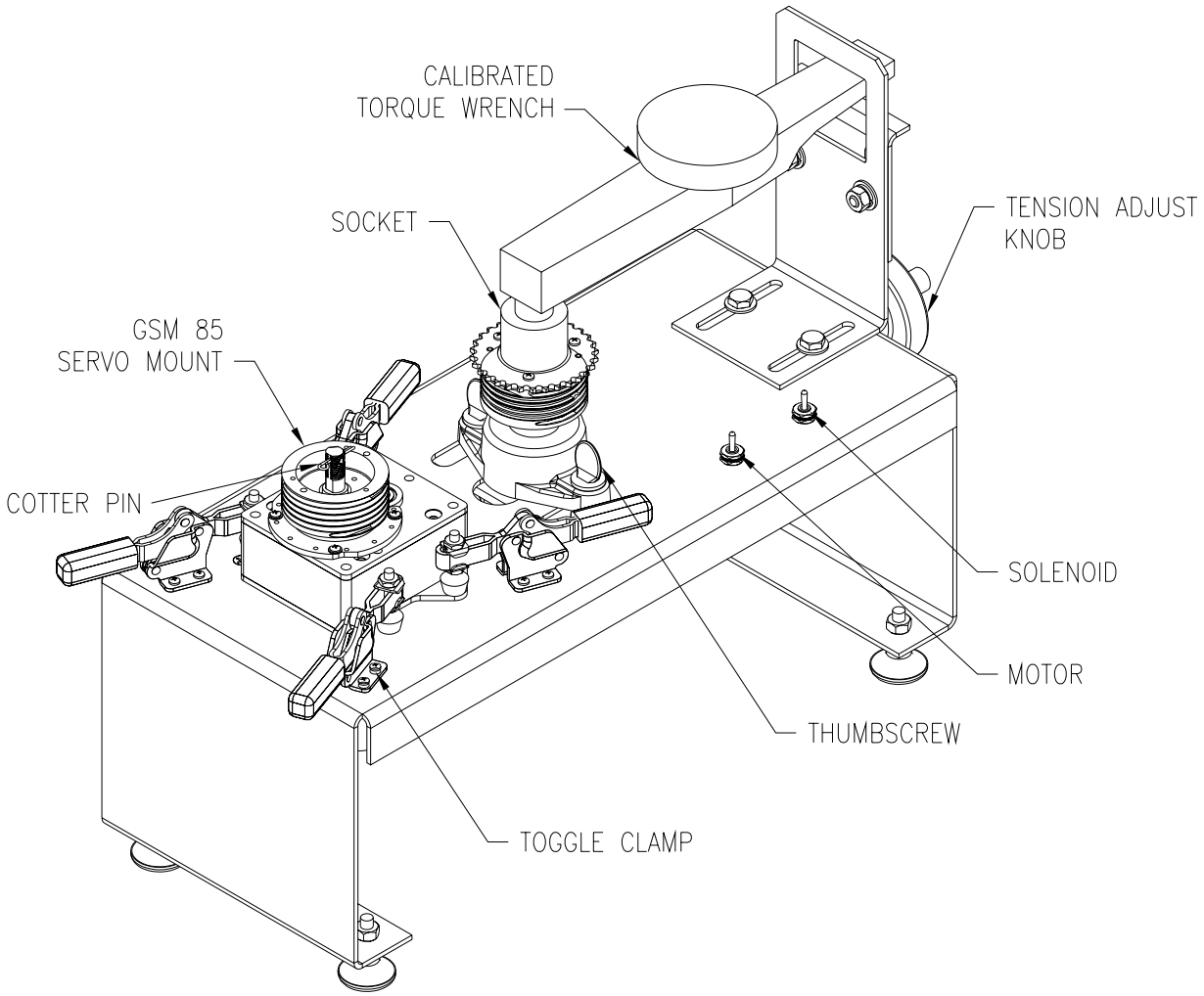

Figure 22 - 13 Garmin Servo Adjustment Fixture

22-22. GSA 81 AUTOPILOT PITCH SERVO (See Figure 22 - 14)

The GSA 81 is mated to the GSM 85 Servo Mount to form a single servo unit

The design of the servo assembly allows the servo portion (GSA 81) to be removed from the capstan (GSM 85) without the need to de-rig the aircraft control cables. The pitch servo is located beneath the aft baggage area.

a. **Removal**

1. Ensure aircraft power is off. Unplug any auxiliary power supplies.
2. Remove the baggage bulkhead access panel located under the hat rack per Chapter 25.
3. Disconnect servo connector P681.
4. Remove the two bolts holding the servo to the servo mount. See Figure 22 - 12.
5. Remove the unit.
6. Remove excess grease build-up from the single servo output gear using a lint-free cloth.

NOTE

It is not necessary to remove all of the grease from the output gear, only the excess grease. D0 NOT USE SOLVENTS TO CLEAN THE OUTPUT GEAR!

7. Using a brush or other applicator, apply a thin coat of grease to the servo output gear. Use Aeroshell 17 or equivalent (Synthetic Diester, Low Temp; Must meet MIL-G-21164D).

b. **Installation**

1. Set unit in place.
2. Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
3. Install the two bolts holding the servo to the servo mount.
4. Connect servo connector P681.
5. Configure and test the GSA 81 per the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.
6. Operate all controls through full travel and ensure that no binding or restriction occurs.
7. Reinstall the baggage bulkhead access panel and the hat rack per Chapter 25.

22-23. GSM 85 AUTOPILOT PITCH SERVO MOUNT

a. **Removal**

1. Remove the GSA 81 servo per paragraph 22-22.
2. Remove the two clamps holding the bridle cable to the main control cable.
3. Remove the four screws, two washers, and two self locking nuts holding the mount to the servo bracket.
4. Remove the unit with the bridle cable.

5. A tapered shim is attached to the face of the servo mount by two 10-32 x 17/32 screws and two 10-32 x 25/32 screws. Remove shim and retain for installation.

b. **Installation**

1. Prior to installation, adjust and verify the slip clutch torque is set to 50 ± 7 in-lb. per paragraph 22-21.b.1.
2. Install tapered shim.
3. Set the unit in place.
4. Install the four screws, two washers, and two self locking nuts holding the mount to the servo bracket. Apply Loctite 242 to the two screws without washers and nuts.
5. Install the bridle cable. Wrap the turnbuckle end approximately 270º over the capstan. Wrap the fork end approximately 450º over the capstan. Set the cable tension to 25 to 60 lbs. The bridle cable should make two complete revolutions around the capstan and the lock ball should be mated in its seat. System rigging to be accomplished with controls neutral.
6. Install the GSA 81 servo per paragraph 22-22.

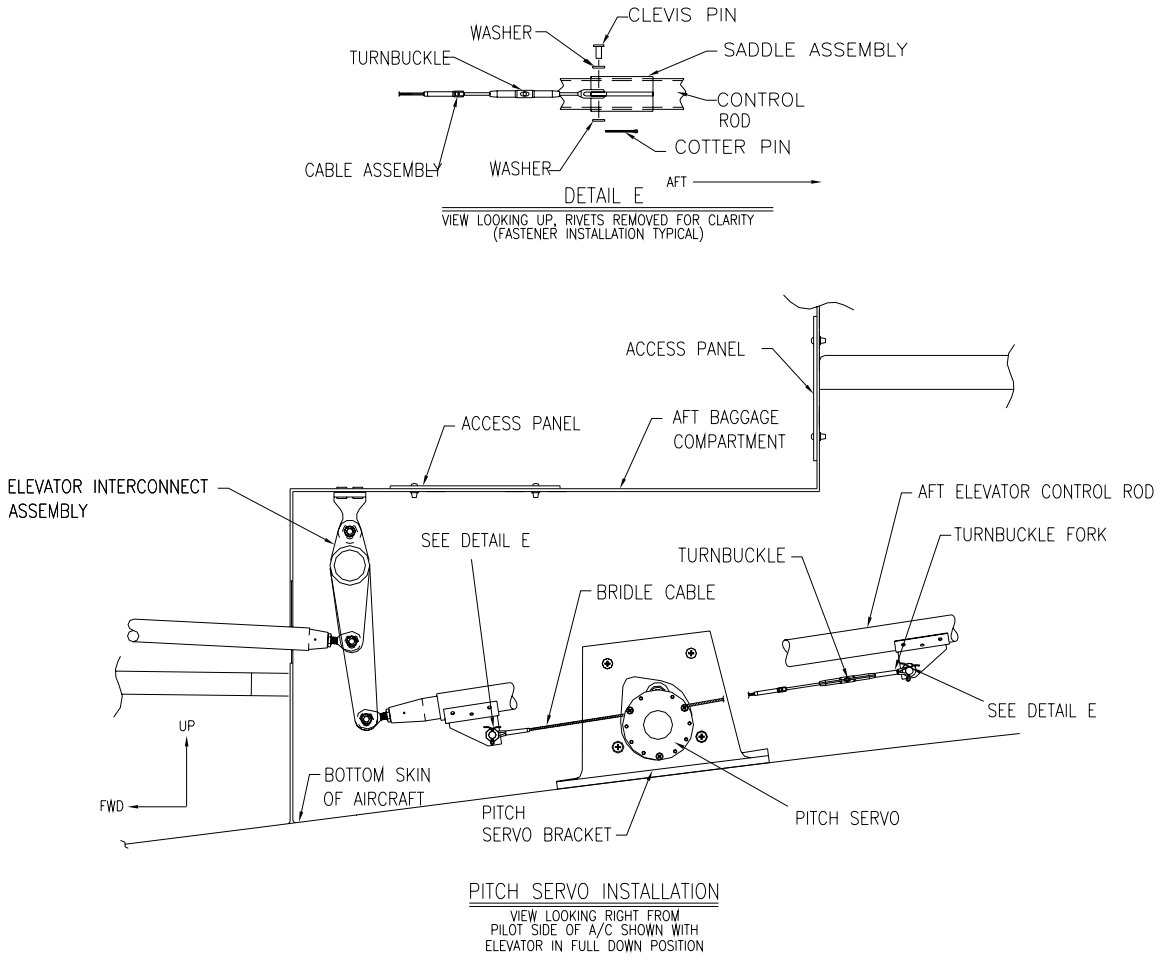

Figure 22 - 14 Pitch Servo

22-24. GTA 82 TRIM ADAPTER

The Garmin GTA 82 Trim Adapter is a remote mounted device that is used to allow the GFC 700 to drive a pitch trim actuator provided by the airframe manufacturer. The GTA 82 adapter is located under the instrument panel on the pilot side beneath the standby instruments. The trim adapter is powered by the Essential Bus and interfaces with two GIA 63/GIA 63W Integrated Avionics units through serial communication on separate RS-485 ports. See Figure 22 – 15.

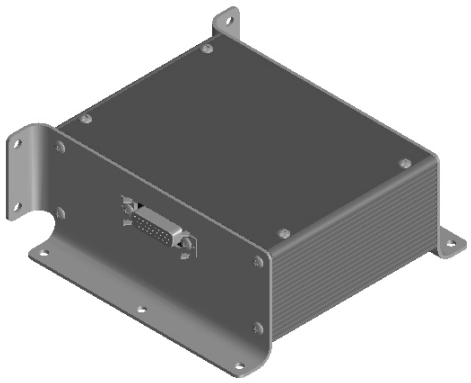

Figure 22 – 15 GTA 82 Pitch Trim Adapter

a. **Removal:**
   1. Remove four screws securing the GTA 82 to the panel. The GTA 82 is located under the instrument panel on the pilot side beneath the standby instruments.
   2. Disconnect connector.
   3. Remove unit.

b. **Replacement:**
   1. Set unit in place.
   2. Install four screws to secure unit to panel.
   3. Connect connector.
   4. After installation of unit, reload the software and the autopilot gain files.

22-25. INSPECTION

a. **Annual**
   1. Move the flight controls through their full range of motion. Check for any signs of binding or resistance.
   2. Engage the A/P, check the Pitch, Roll, and Trim servos for normal operation.
   3. Press the A/P disconnect switch on the pilot control stick and verify that the A/P disengages and a disconnect tone is played through the aircraft speaker. Repeat for the copilot stick.
   4. Visually inspect each Servo Mount (GSM 85), moving the control surface through its full range of motion. Ensure there is no binding in the control or bridle cables. Check the cables for fraying or corrosion, and check the mount brackets for cracks or wear.

5. Check that bridle cable tension is 25 to 60 lbs., and slip clutch settings are 50 ± 7 in.-lbs. for each capstan in accordance with Section 22-21.
6. Perform servo checks per Section 6.9.4 of the G1000 System Maintenance Manual, P/N 190-00577-03.

b. **500 Hour**

1. If the aircraft has been flown more than 500 hours since the last GSM 85 inspection, each servo mount clutch setting must be verified as outlined in Section 22-21.
2. Perform servo checks per Section 6.9.4 of the G1000 System Maintenance Manual, P/N 190-00577-03.

c. **1000 Hour –** Perform a visual inspection every 1000 airframe hours in conjunction with the next scheduled inspection and check for corrosion, damage, or other defects for each of the items listed below. Replace any damaged parts as required. Inspection may require the temporary removal of a unit or units to gain access to connectors.

3. **GSA 81 Servo Actuator** – Separate each GSA 81 from the GSM 85 Servo Mount (pitch, roll). Clean and re-grease the output gear in accordance with the Garmin G1000 System Maintenance Manual, P/N 190-00577-03. Check connector and wiring.
4. **GSM 85 Servo Mount** – Inspect the GSM 85 Mounts. Look for excessive wear in the capstan, and bridle cables.
5. **GTA 82 Trim Adapter** – Inspect the GTA 82 unit and connector for corrosion or other defects.

CHAPTER 23

COMMUNICATIONS

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Chapter 23

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23-1. GENERAL

a. The Cessna 350 (LC42-550FG) utilizes either the Garmin AT Apollo™ SL15, Garmin GMA 340, or Garmin GMA 1347 communications systems. There are variations of the aircraft communications and navigation systems (see Chapter 34 for navigation system description). Refer to Chapter 34 and the Garmin AT and Garmin manuals for specific information on the actual Nav/Comm hardware. This chapter describes the system of providing communications output to the occupants and protection from precipitation static and lightning. For a specific listing of installed equipment, refer to Section 6, Appendix A, of the POH/AFM.

1. **Description of Audio Panel –** Either the Apollo™ SL15MS Audio Selector Panel/Intercom System (ASPIS), Garmin GMA 340 audio panel, or Garmin GMA 1347 audio panel is installed in your aircraft. The primary purpose of the panel is to control communication and navigation audio selections, intercom functions, and the marker beacons. In addition, the unit has provisions for a stereo entertainment input. The audio panel is located in the top of the radio rack panel assembly.

   The Garmin GMA 1347 audio panel is installed in your aircraft with the Garmin G1000 Option. See Chapter 34 for description, removal, and installation.

2. **Description of Static Discharge System –** During use, the airplane is susceptible to a build-up of static electricity. An accumulated static electrical charge can render radio navigation and communication inoperative. The purpose of the static discharge system is to transfer the electrical charge off the airplane's surface and quietly discharge it into the air.
3. **Description of Lightning Protection System –** The composite construction of the Cessna 350 has a very high resistance to electricity and very little conductivity. Having a high conductivity is critical to lightning protection because it is important that all parts of the airplane have the same electrical potential. To achieve high conductivity, aluminum and copper mesh was integrated with the composite construction of the airplane. The mesh varies from being 10 to 30 thousandths of an inch below the surface of the paint and encompasses most surfaces of the airplane. Various parts of the airplane are interconnected through use of metal fasteners inserted through several plies of mesh, mesh overlaps, and bonding straps. See Chapter 20 for other lightning protection information.

CAUTION

Before any maintenance, removal, or installation of equipment is performed, make sure both the master and avionics switches are in the off position. Failure to do so could result in damage to these instruments.

23-2. SPEECH COMMUNICATIONS

a. All communication radios are integrated with some form of navigational receiver, and it is impracticable to segregate the discussion of each function in two separate chapters. Accordingly, air-to-ground speech communications are covered in Chapter 34.

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23-3. APOLLO SL15 AUDIO PANEL

a. The SL15 Audio Panel acts as the audio selector for the airplane. The SL15 can switch between three transceivers (Com 1, Com 2, and Com 3) and six receivers (Nav 1, Nav 2, ADF, DME, MKR, and AUX). The SL15 also contains a voice activated intercom system, which allows the pilot, copilot, and passengers of the airplane to communicate among themselves. See Figure 23 - 1 SL15 Audio Panel. The intercom select switch can be set to three modes.

1. In the "All" mode, the intercom is linked to all four-seat positions; everyone hears radio communications, and everyone hears music from input Entertainment No. 1.
2. In the "Crew" mode, the pilot and front seat passenger are linked together and can receive radio communications. The rear seat passengers are linked together but cannot hear radio communications or the pilot or front passenger.
3. In the "Isolate" mode, the pilot hears the radios, but is not connected to the intercom, while the front and rear passengers are on the same intercom loop but cannot hear radio communications.

b. All four headsets are hooked up to the intercom system. A volume knob on the left side of the SL15 controls volume for the headsets.

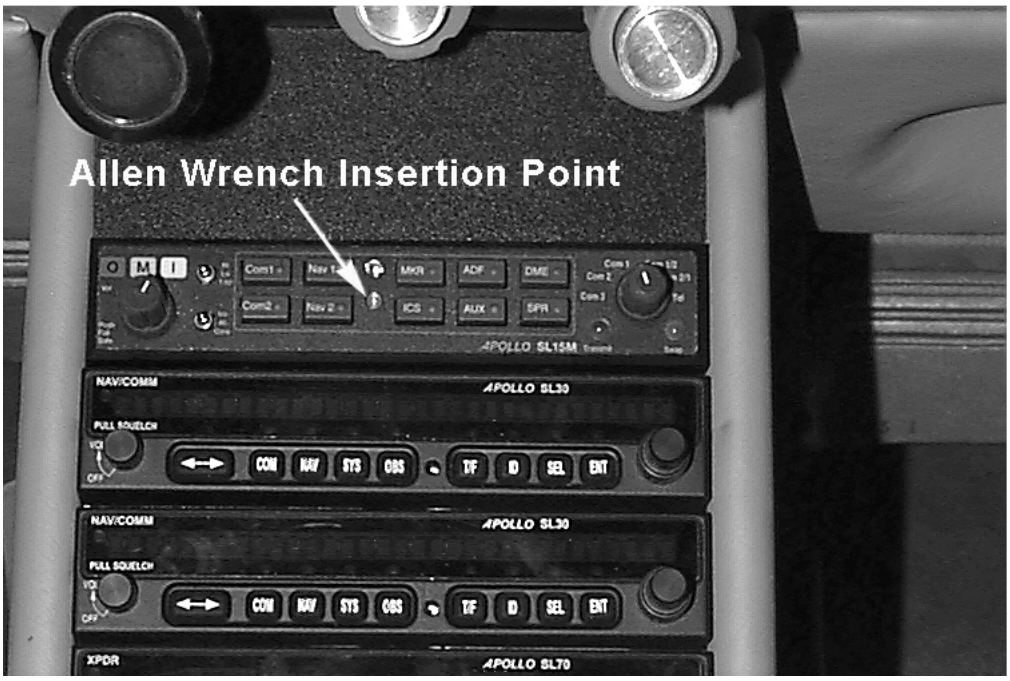

Figure 23 - 1 SL15 Audio Panel

c. **Removal of SL15 Audio Panel**

1. Insert a $^{3}/_{32}$ in. Allen wrench into the hole on the faceplate. See Figure 23 - 1.
2. Turn the Allen wrench counterclockwise.
3. The SL15 Audio Amplifier will slide out of the radio rack.

d. **Installation of SL15 Audio Panel**

1. Install fittings into the back of the SL15.
2. Slide the SL15 into its slot in the radio rack.

3. Insert a $^{3}/_{32}$ in. Allen wrench into the hole on the faceplate and turn clockwise. See Figure 23 - 1.

23-4. GARMIN GMA 340 AUDIO PANEL

a. The GMA 340 Audio Panel acts as the audio selector for the airplane. The GMA 340 is a VHF Communication Transceiver/VOR/ILS Receiver/GPS Receiver. The system is a fully integrated panel-mounted instrument which contains audio and microphone switching and amplification, a marker beacon receiver, and an intercom system. The GMA 340 also contains a voice activated intercom system, which allows the pilot, copilot, and passengers of the airplane to communicate among themselves. See Figure 23 - 2. The intercom select switch can be set to three modes.

1. In the "All" mode, the intercom is linked to all four-seat positions; everyone hears radio communications, and everyone hears music from input Music 1.
2. In the "Crew" mode, the pilot and front seat passenger are linked together and can receive radio communications. The rear seat passengers are linked together but cannot hear radio communications or the pilot or front passenger.
3. In the "Pilot" mode, the pilot hears the radios, but is not connected to the intercom, while the front and rear passengers are on the same intercom loop but cannot hear radio communications or communicate with the pilot.

Figure 23 - 2 Garmin GMA 340 Audio Panel

b. All four headsets are hooked up to the intercom system. There is a volume knob for the pilot and copilot's sides.

c. **Removal of GMA 340 Audio Panel**

1. Insert a $^{3}/_{32}$ in. Allen wrench into the jack screw on the faceplate. See Figure 23 - 1, which shows the SL15 panel, but is similar to the GMA 340 installation.
2. Turn the wrench counterclockwise until loose.
3. Slide out the GMA 340 unit, and disconnect the connectors.

d. **Installation of GMA 340 Audio Panel**

1. Install fittings into the back of the GMA 340.
2. Slide the unit into the rack until the jack screw makes contact with the receptacle located in the back plate.
3. Insert a 3/32 in. Allen wrench into the jackscrew access hole on the faceplate.
4. Turn the Allen wrench clockwise until the unit is secured in the rack. Continue turning until tight, but do not over-tighten.

23-5. SPEAKER

a. **Removal of Speaker**

1. Remove the overhead center panel per instructions in Chapter 25.

2. For Type A, remove the four screws connecting the stall warning buzzer mounting, speaker, loop clamp, and speaker mounting. For Type B, remove the four screws connecting the speaker to the adapter plate. See Figure 23 - 3.
3. Remove the stall warning buzzer mounting, as required.
4. Disconnect the wires.
5. Pull the speaker from the mounting.

b. **Installation of Speaker**

1. Put the speaker on the speaker mounting.
2. Connect the wires.
3. Place the stall warning buzzer mounting on top of the speaker, as required.
4. For Type A, tighten the four nuts connecting the stall warning buzzer mounting, speaker, loop clamp, and speaker mounting. For Type B, tighten the four nuts connecting the speaker to the adapter plate. See Figure 23 - 3.
5. Install the overhead center panel per instructions in Chapter 25.

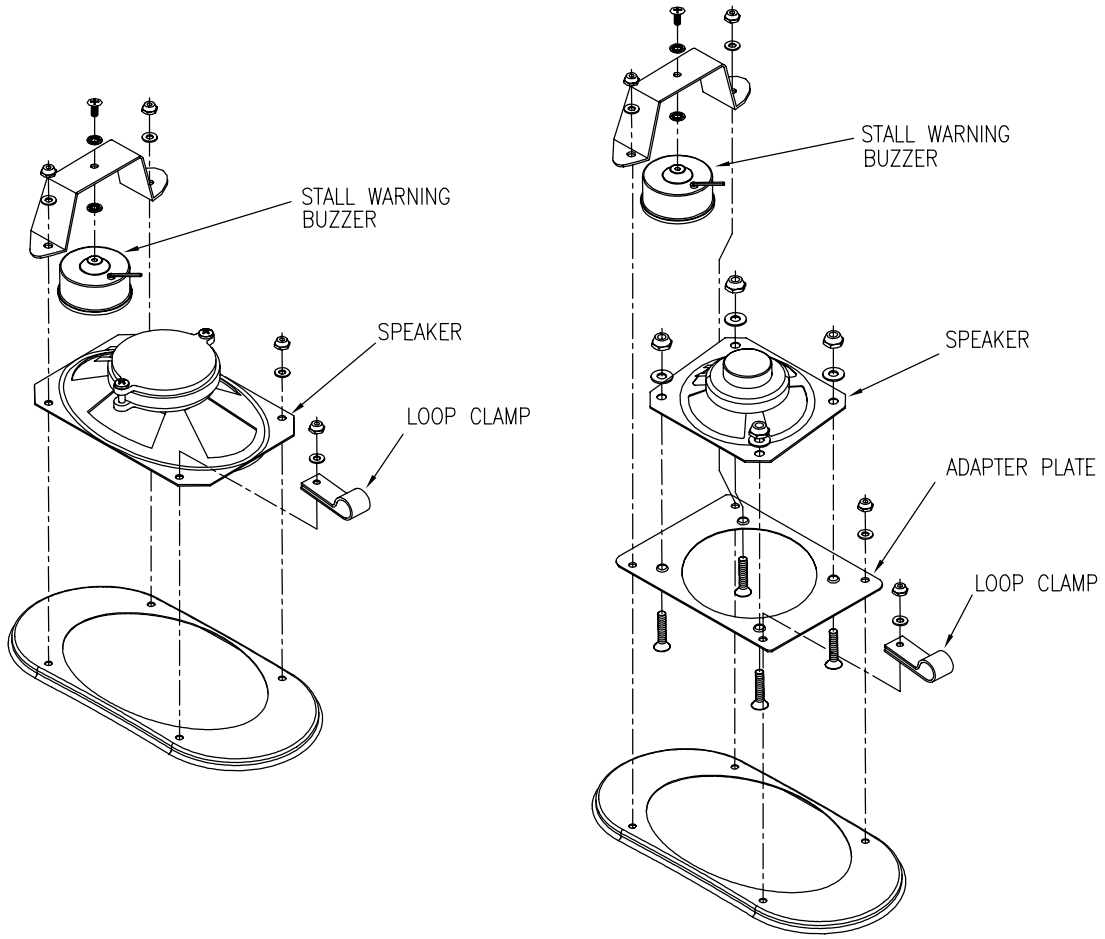

Figure 23 - 3 Overhead Speaker Installation

23-6. PUSH-TO-TALK SWITCH

a. **Location** – Each control stick has a push-to-talk switch located on the forward part of the stick where it is easily accessed by the index finger. See Figure 23 - 4. The switch is press-fit into the stick and connected by wire on the inside of the stick assembly.
b. **Description –** The push-to-talk switch is a single-button switch that when pressed, makes a connection required to energize the Comm unit by way of the Audio Panel.

23-7. TROUBLESHOOTING PUSH TO TALK FUNCTION

a. **Symptom** – Push-to-talk function not working properly.
Probable Cause – Broken wire or connection.
1. Locate the defective connection:
   a) Remove the interior panel covering the control stick.
   b) Locate P-46 (pilots side) and P-47 (copilot's side), and disconnect the plug.
2. Using a digital volt ohmmeter (DVOM), check for operation of the push-to-talk function switch. Place one probe on pin 1 and ground the other. The DVOM should show a closed circuit with the switch depressed (less than 2 ohms).
3. Check the jack side of the connection. Pin 2 should be ground and Pin 1 should be for the audio panel.
4. If the switch needs to be replaced, cut the tie-wraps at the base of the control stick to allow the switch to be gently pulled out of the wooden handle.
5. Solder in a new switch, and pull the wires tight at the base of the control stick tube.
6. Secure the wires with tie-wraps, and check for freedom of movement.

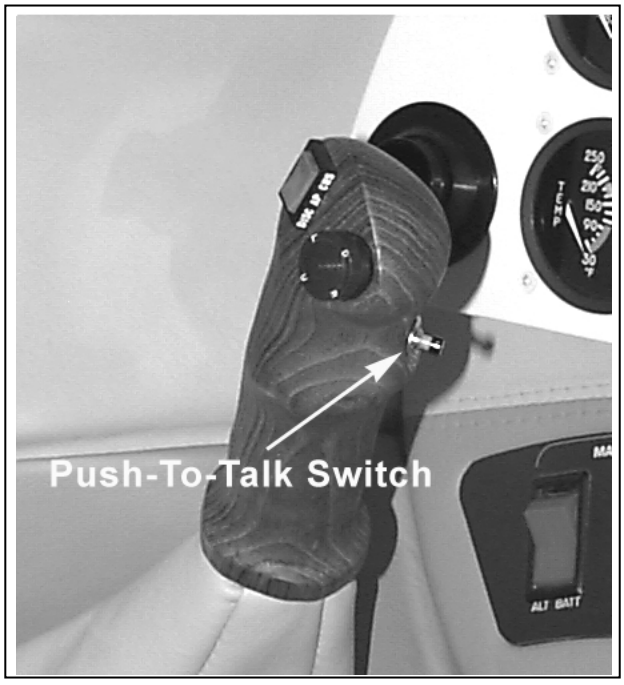

Figure 23 - 4 Push-To-Talk Switch Location

23-8. NAV/COM BYPASS SWITCH

a. **Description –** With the Garmin G1000 system the Nav/Com bypass switch is not installed. With the Avidyne option the aircraft is equipped with a Nav/Com bypass switch located above the attitude indicator next to the clock. The switch is designed to be used in situations where the engine is not running. The switch is an alternate action device that activates when depressed and needs to be depressed again to cycle off. When activated, the switch turns on the No. 1 GPS, No. 1 Com, and No. 1 Nav by connecting them directly to the right battery, bypassing the power grid, circuit breaker panel, and associated wiring. Communications is only through the pilot's headset and activated with the pilot's PTT switch.

b. **Removal of Switch**
   1. Remove the glare shield per instructions in Chapter 25 to gain access to the back side of the instrument panel.
   2. Disconnect the connector at the back of the switch.
   3. Pull the press-fit switch out of its housing at the back of the instrument panel as shown in Figure 23 - 5.

c. **Installation of Switch** – Installation is a direct reversal of removal.

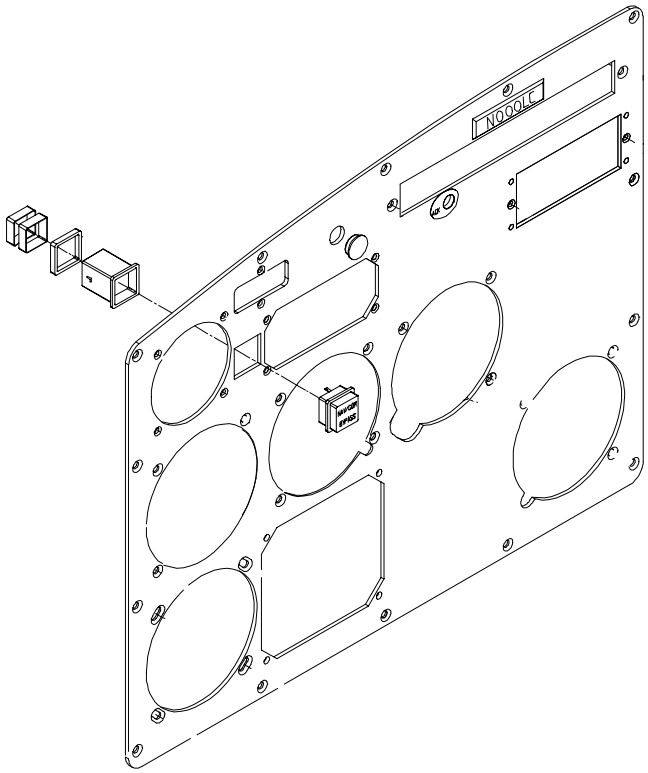

Figure 23 - 5 Nav/Com Bypass Switch

23-9. HEADPHONE JACKS/MICROPHONE PLUGS

a. **Description –** The Cessna 350 (LC42-550FG) is designed to handle both standard headphone/microphone jacks as well as Bose™ Stereo headphone jacks.

b. **Location –** Each seat has a jack plug receptacle.
   1. **Front Seats –** The location of the front seat receptacles are at the lower part of the center console for the Avidyne option and at the front of the center console for the Garmin G1000 option. See Figure 23 - 6 for Avidyne option and see Chapter 25 for Garmin G1000 option.

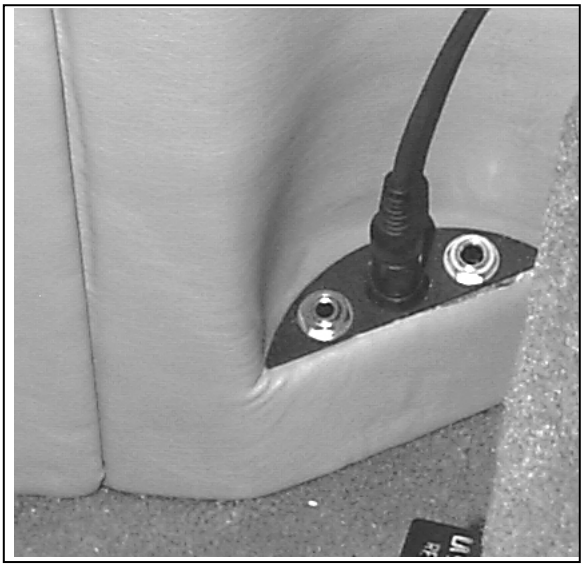

Figure 23 - 6 Front Seat Headphone Jack Receptacle

   2. **Rear Seats** – The location of the rear seat receptacles are on the rear seat side panels. See Figure 23 - 7.

Figure 23 - 7 Rear Seat Headphone Jack Receptacle

CAUTION

Bose™ headphones require power, which is supplied by the Bose™ jack. Before removing the intercom jack, ensure the system master switch is in the off position.

c. **Removal of Rear Passenger Intercom Jacks**
   1. To gain access to the backside of the jacks, remove the interior sidewall panel per instructions in Chapter 25.
   2. Unscrew and remove the Bose jack cover. Loosen and remove the jam nuts securing the jacks to the interior panel. See Figure 23 - 8.
   3. Label all wires according to where they were connected.
   4. Cut wire near the solder point at each jack.

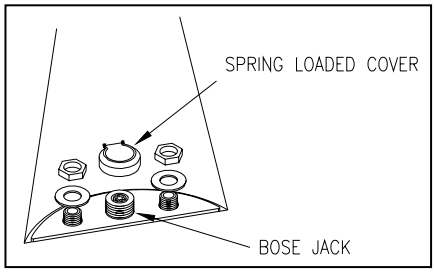

Figure 23 - 8 Headphone/Microphone Jack Installation

d. **Installation of Rear Passenger Intercom Jack**s
   1. Remove all traces of old solder from the jacks.
   2. Reconnect and solder the wires to the jacks. Ensure all wires are soldered to the correct pins on the jacks.
   3. Secure the jacks to the sidewall panel using jam nuts.
   4. Install the Bose jack cover.
   5. Install the sidewall panel.

CAUTION

Bose™ headphones require power, which is supplied by the Bose™ jack. Before removing the intercom jack ensure the system master switch is in the off position.

e. **Removal of Pilot/Front Passenger Intercom Jacks/Microphone Plug**
   1. To gain access to the jacks, remove both sides of the center console panels. Only one panel in Garmin G1000 option.
   2. Unscrew and remove the Bose jack cover. Loosen and remove the jam nuts securing the jacks to the interior panel. See Figure 23 - 8.
   3. Label all wires according to where they were connected.
   4. Cut wire near the solder point at each jack.

f. **Installation of Pilot/Front Passenger Intercom Jacks/Microphone Plug**
   1. Remove all traces of old solder from jacks.
   2. Reconnect and solder wires to the jacks. Ensure all wires are soldered to the correct pins on the jacks.
   3. Secure the jacks to the sidewall panel using jam nuts.
   4. Install the Bose jack cover.

5. Install the sidewall panel.

23-10. STEREO AUDIO INPUT JACK, OPTIONAL

a. **Avidyne Option**

1. **Removal of Stereo Audio Input Jack**
   a) To gain access to the jack, remove the center console access panel.
   b) Lift up the armrest and locate the jack at the forward end of the pencil box.
   c) Loosen and remove the jam nut securing the jack to the pencil box. See Figure 23 - 9.
   d) Label all wires according to where they were connected.
   e) Cut the wire near the solder point at each jack.
2. **Installation of Stereo Audio Input Jack**
   a) Remove all traces of old solder from the jack.
   b) Reconnect and solder wires to the jack. Ensure all wires are soldered to the correct pins on the jack.
   c) Secure the jack to the pencil box using the jam nut.
   d) Install the center console access panel.

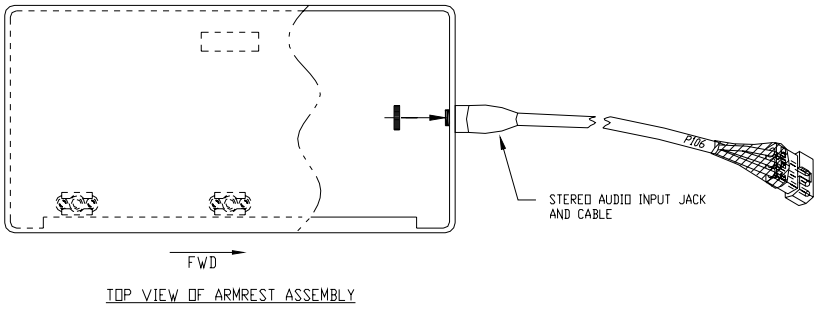

Figure 23 - 9 Stereo Audio Input Jack

b. **Garmin G1000 Option –** The stereo audio input jack is located under the instrument panel to the right of the tower on the co-pilot's side.

1. **Removal of Stereo Audio Input Jack**
   a) Loosen and remove the jam nut securing the jack to the instrument panel.
   b) Reach up behind the instrument panel and remove the jack.
   c) Label all wires according to where they were connected.
   d) Cut the wire near the solder point at each jack.
2. **Installation of Stereo Audio Input Jack**
   a) Remove all traces of old solder from the jack.
   b) Reconnect and solder wires to the jack. Ensure all wires are soldered to the correct pins on the jack.
   c) Secure the jack to the instrument panel using the jam nut. Apply one drop of Loctite 243 to the threads and tighten snug (no looseness plus 1/4 turn).

23-11. STATIC DISCHARGE WICKS

a. **Detection of a Broken Static Wick –** To check the wick's integrity, hold the trailing edge between the thumb and forefinger, and gently move it left and right approximately 2 in. If the unit flexes at point A, as shown in Figure 23 - 10, the wick is broken and should be replaced.

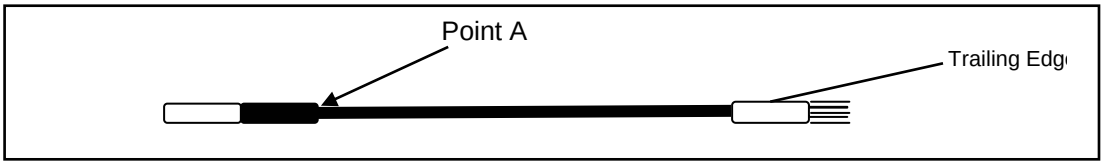

Figure 23 - 10 Static Wick

b. **Removal of Static Wick** – (See Chapter 20 for repair details)
   1. Unscrew the screw connecting the static wick to its housing.
   2. Remove the static wick from the housing.

CAUTION

When removing and handling a good static wick, do not bend it. The static wick can be broken without obvious exterior indications.

WARNING

If the wicks are removed and not reinstalled before flight, problems with navigation and communication equipment can occur.

NOTE

If the static wick does not slide out after removing the screw, use a pair of pliers and gently pull the static wick from its housing.

c. **Installation of Static Wick** – (See Chapter 20 for repair details)

CAUTION

Before installing the static wick, perform the broken static wick test described below. A static wick can be broken without any obvious exterior indications.

   1. To install a static wick place the wick in its housing.
   2. Screw the static wick and its housing together.

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CHAPTER 24

ELECTRICAL POWER

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Chapter 24

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24-1. ELECTRICAL SYSTEM - GENERAL

a. **Electrical Power System** – The electrical system in this aircraft consists of two independent buses, which are referred to as the left bus and right bus. Two alternators provide charging power for the batteries, as well as system power. The batteries will also provide additional power in the event of an over demand situation where the requirements on the system are greater than what can be provided by the alternator. The left and right buses in turn feed the avionics and essential buses. A summary of the busses and equipment powered by each bus is shown in Figure 24 - 1 or Figure 24 - 2. A functional diagram of the electrical system is shown in Figure 24 - 3 or Figure 24 - 4.

b. Five current limiters protect the alternators and bus outputs. In addition, the left and right buses are physically isolated inside a covered area mounted to the firewall. Left and right bus controls, grounds, and outputs are routed through separate holes, connectors, and cable runs so any failure on one bus will not affect the operation of the other bus.

c. Control of the buses is via the master switch panel located on the lower left portion of the instrument panel (Basic or Avidyne) or located in the cockpit overhead (Garmin G1000). There is also a crosstie switch on this panel, which will restore power in the event of failure of the right or left systems. For example, if the alternator or some other component on the left side should fail, the crosstie switch will restore power to the electrical items on the left bus by connecting the left bus to the right bus.

d. As its name may suggest, power to the essential bus is never affected, provided power from at least one bus (left or right) is available. The essential bus is diode fed, i.e., current will only flow in one direction, from both the right bus and the left bus allowing the essential equipment to have two sources of power.

e. **Avionics Bus**
   1. **Basic or Avidyne Option –** The avionics bus provides power to the Audio/Voice, GPS 1, GPS 2, Nav/Com 1, Com 2, Transponder/Encoder/Equipment Fan, HSI or Traffic, Autopilot, Map or MFD, and Weather.
   2. **Garmin G1000 Option –** The avionics bus provides power to the Audio/MKR, Integrated, Avionics #2, Com #2, Transponder, Avionics Fan, Traffic, Autopilot, MFD, and Weather.

f. **Left Bus**
   1. **Basic or Avidyne Option –** The left bus provides power for the Aileron Trim, Pitot Heat, SpeedBrakes, Engine Instruments, Rudder Limiter, Carbon Monoxide Detector, Oxygen, Position Lights, Landing Light, Left Voltage Regulator, Clock, Cabin Fan, and PFD.
   2. **Garmin G1000 Option –** The left bus provides power for the Aileron Trim, Pitot Heat, SpeedBrakes, Rudder Limiter, Position Lights, Landing Light, Left Voltage Regulator, and Fan.

g. **Right Bus**
   1. **Basic or Avidyne Option –** The right bus provides power for the Strobe Lights, Taxi Light, Right Voltage Regulator, Door Seal, Power Point, PFD, and Elevator Trim.
   2. **Garmin G1000 Option –** The right bus provides power for the Strobe Lights, Taxi Light, Right Voltage Regulator, Door Seal/Power Point, Carbon Monoxide Detector, Oxygen, Display Keypad, and Air Conditioning.

h. **Essential Bus**

1. **Basic or Avidyne Option –** The essential bus is diode fed from either the right or the left bus and provides power for the Attitude Horizon, Turn Coordinator, Panel Lights, Annunciators, Left Bus Relays, Fuel Pump, Stall Warning, Flaps, and the Right Bus Relays.
2. **Garmin G1000 Option –** The essential bus is diode fed from either the right or the left bus and provides power for the PFD, AHRS, Elevator Trim, Panel Lights, Standby Instrument Lighting, Air Data Computer, Engine Airframe, Integrated Avionics #1, Com #1 Left Bus Relays, Fuel Pump, Stall Warning, Flaps, Standby ADI, and the Right Bus Relays.

i. **Battery Bus**

1. **Basic or Avidyne Option –** Four items are connected to the battery bus. These items will operate even if the left and right buses are turned off since they are directly connected to either the left or right battery. The Nav/Com bypass switch, Hobbs Meter, and ELT, are connected to the right battery and the courtesy lights/flip lights are connected to the left battery. The Hobbs Meter, ELT, and courtesy lights/flip lights are each protected by a 3 amp fuse which is not accessible from the cockpit.
2. **Garmin G1000 Option –** Three items are connected to the battery bus. These items are directly wired to the battery, but the term battery bus is used for consistency. These items will operate even if the left and right buses are turned off since they are directly connected to the right or left battery. The items directly wired to the right battery include the Hobbs Meter, and ELT. The courtesy lights/flip lights are directly wired to the left battery. A 3-amp fuse protects each component and is not accessible from the cockpit.

CAUTION

For extended service life, disconnect the batteries to prevent them from being continuously charged by external power when the aircraft is in maintenance.

CAUTION

Do not replace the sealed lead acid batteries with wet lead acid batteries.

CAUTION

Never "jump-start" an aircraft that has a "dead" or discharged battery. It takes approximately three hours to recharge a fully discharged battery with the aircraft generating system or external power.

3. **Nav/Com Bypass Circuit –** The Nav/Com bypass switch is an alternate action device that activates when depressed and needs to be depressed again to cycle off. Activation of the Nav/Com bypass switch turns on the No. 1 GPS, No. 1 Com, and No. 1 Nav by connecting them directly to the right battery, bypassing the power grid, circuit breaker panel, and associated wiring. The Nav/Com bypass switch is protected by two 5 amp fuses.

SUMMARY OF BUSES, BASIC OR AVIDYNE OPTION		
Bus	**Bus Component**	**Circuit Breaker**
AVIONICS BUS	• Audio/Voice • GPS 1 • GPS 2 • Nav/Com1 • Com 2 • Transponder/Encoder and Equipment Fan • HSI (5 amp) or Traffic (4 amp) • Autopilot • Map or MFD • Weather	5 amp 5 amp 5 amp 10 amp 10 amp 5 amp 5 amp or 4 amp 5 amp 10 amp 5 amp
LEFT BUS	• Aileron Trim • Pitot Heat (fuse amperage depends on Pitot installed) • SpeedBrakes • Engine Instruments • Rudder Limiter • Carbon Monoxide Detector • Oxygen • Position Lights • Landing Light • Left Voltage Regulator • Clock, and Cabin Fan • PFD Power • Air Conditioning	1 amp 7.5 amp or 10 amp 3 amp 3 amp 5 amp 2 amp 3 amp 10 amp 4 amp 5 amp 7.5 amp 10 amp 20 amp
RIGHT BUS	• Strobe Lights • Taxi Light • Right Voltage Regulator • Door Seal/Power Point • PFD Power • Elevator Trim • Air Conditioning	10 amp 4 amp 5 amp 5 amp 10 amp 1 amp 15 amp
ESSENTIAL BUS	• Attitude Horizon • Turn Coordinator • Panel Lights • Annunciators • Left Bus Relays • Fuel Pump • Stall Warning • Flaps • Right Bus Relays.	3 amp 3 amp 7.5 amp 3 amp 5 amp 10 amp 5 amp 10 amp 5 amp
BATTERY BUS	• Nav/Com Bypass Switch • ELT • Courtesy Lights/Flip Lights • Hobbs Meter	Two 5 amp 3 amp 3 amp 3 amp

Figure 24 - 1 Summary of Buses and Equipment (Basic or Avidyne Option)

SUMMARY OF BUSES FOR GARMIN G1000		
Bus	**Bus Component**	**Circuit Breaker**
AVIONICS BUS	• Audio/MKR • Integrated Avionics #2 • Com #2 • Transponder • Avionics Fan • Traffic • Autopilot • MFD • Weather	5 amp 5 amp 5 amp 5 amp 3 amp 3 amp 5 amp 5 amp 3 amp
LEFT BUS	• Aileron Trim • Pitot Heat • SpeedBrakes • Rudder Limiter • Position Lights • Landing Light • Left Voltage Regulator • Fan	2 amp 7.5 amp 3 amp 5 amp 5 amp 5 amp 5 amp 5 amp
RIGHT BUS	• Strobe Lights • Taxi Light • Right Voltage Regulator • Door Seal/Power Point • Carbon Monoxide Detector • Oxygen • Display Keypad • Air Conditioning	5 amp 2 amp* 5 amp 5 amp 2 amp 3 amp 2 amp 15 amp
ESSENTIAL BUS	• PFD • AHRS • Elevator Trim • Panel Lights • Air Data Computer • Engine Airframe • Integrated Avionics #1 • Com #1 • Left Bus Relays • Fuel Pump • Stall Warning • Flaps • Standby Attitude Horizon • Right Bus Relays	5 amp 5 amp 2 amp 7.5 amp 5 amp 5 amp 5 amp 5 amp 1 amp 5 amp 2 amp 10 amp 3 amp 1 amp
BATTERY BUS	• Hobbs Meter • ELT • Courtesy Light	3 amp 3 amp 3 amp

* 5 amp for xenon taxi lights, S/N 42502 and on

Figure 24 - 2 Summary of Buses and Equipment (Garmin G1000 Option)

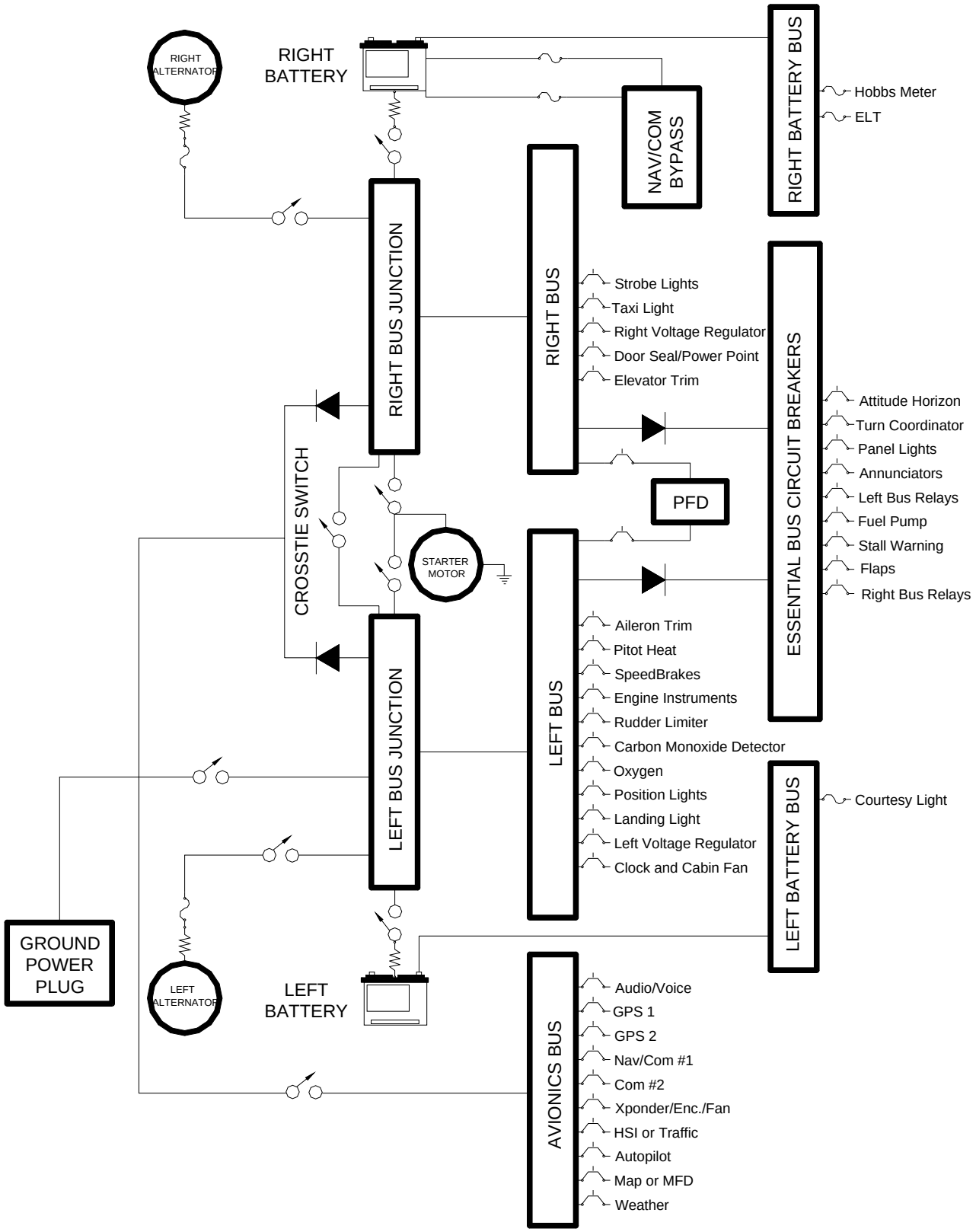

Figure 24 - 3 Electrical System (Basic or Avidyne Option)

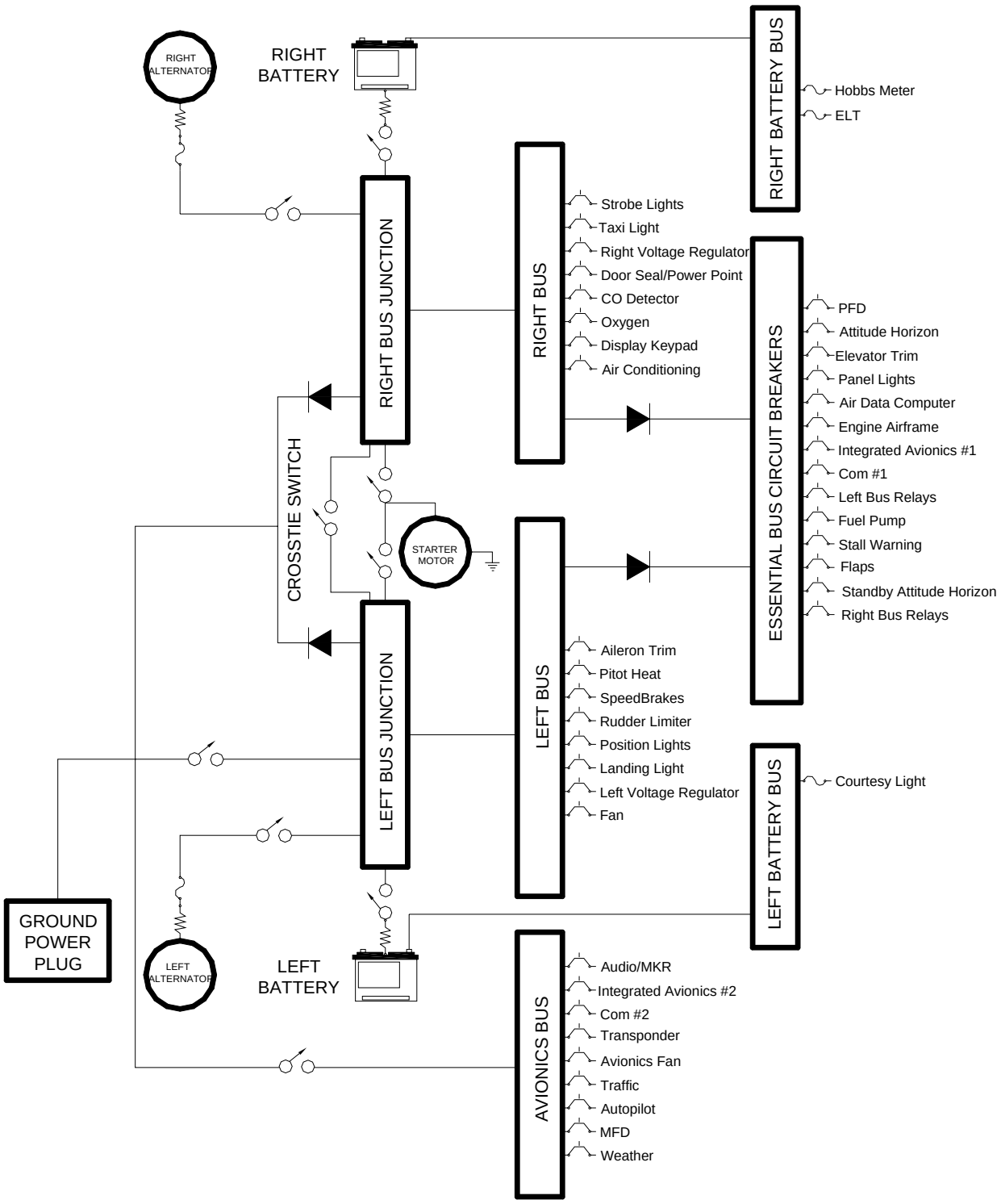

Figure 24 - 4 Electrical System (Garmin G1000 Option)

24-2. TROUBLESHOOTING PROCEDURES FOR THE ELECTRICAL SYSTEM

a. **Symptom** – System does not charge or discharge showing on ammeter.

b. **Probable Cause** – Bad alternator, voltage regulator, or system short

c. **Action** – Charging system problems associated with an alternator should be analyzed in the manner of: Pre-Voltage Regulator Checks, Voltage Regulator Checks, and Post Voltage Regulator Checks.

d. **Pre-Voltage Regulator Checks –** Check the affected alternator switch, voltage regulator circuit breaker, wires, and terminals for contact resistance build-up. Resistance should be less than 0.1 ohms.

e. **Voltage Regulator Check –** Check that there is power coming into and going out of the voltage regulator. The fault/function indicator on the unit is designed to alert the user to the condition of the alternator/voltage regulator.

   1. A red light on the unit, with the affected master switch on means there is a ground short in the alternator field circuit or the voltage regulator's field control transistor is shorted.
   2. A green steady light on the unit with the engine running means there is power coming out of the voltage regulator, but the alternator field or field wire to the voltage regulator is open.
   3. A green fast flickering light on the unit with the engine running means power input devices (like circuit breaker, switch, etc.), and the field systems are functioning correctly.
   4. A green slow flickering light on the unit with the engine running means the power-input devices (like switch, etc.) are defective or have higher than normal resistance.
   5. No light on the unit with the affected master switch on means that one power input device (like switch, circuit breaker, or wiring) or the voltage regulator is defective.

f. **Post-Voltage Regulator Checks**

   1. Check the affected alternator by measuring the resistance of the field. Resistance from field to ground on the alternator should be from 3.5 to 5.0 ohms. Check the resistance of the meter leads before measuring field.
   2. Check for "flying" short and other intermittent problems by slowly rotating the affected alternator while measuring the field resistance. A drop below 3.0 ohms could indicate a bad alternator that might damage the voltage regulator.
   3. Check the condition of the alternator's field, terminals, and wires connecting it to the voltage regulator and the aircraft charging system.
   4. Check the batteries' relays for proper operation and connections.
   5. Check the affected alternator relay for proper operation and connections.

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24-3. DC GENERATION

The Cessna 350 electrical system requires two alternators for the operation of electrical equipment. The No. 1 alternator comes with the engine, is located at the front, right of the engine, and runs the right bus. The No. 2 alternator is the added alternator, is located on the front, left of the engine, and runs the left bus.

A third alternator may be installed on the aircraft, as required, for optional equipment. This alternator is located at the right, rear of the engine.

a. **No. 1 (Right) Alternator –** See Figure 24 - 5

1. **Removal**
   a) Disconnect the two connectors attached to the alternator.
   b) Remove the plain washer, lock washer, and nut from the four mounting studs.
   c) Discard the lock washers.
   d) Remove the alternator from the mounting studs.
   e) Remove the gasket from the mounting studs.
2. **Installation**
   a) Install a new gasket on the alternator.
   b) Install the alternator assembly on the mounting studs with four plain washers, four new lock washers, new gasket, and four plain nuts.
   c) Torque nuts 180 to 220 in.- lbs.
   d) Connect two connectors to the alternator.

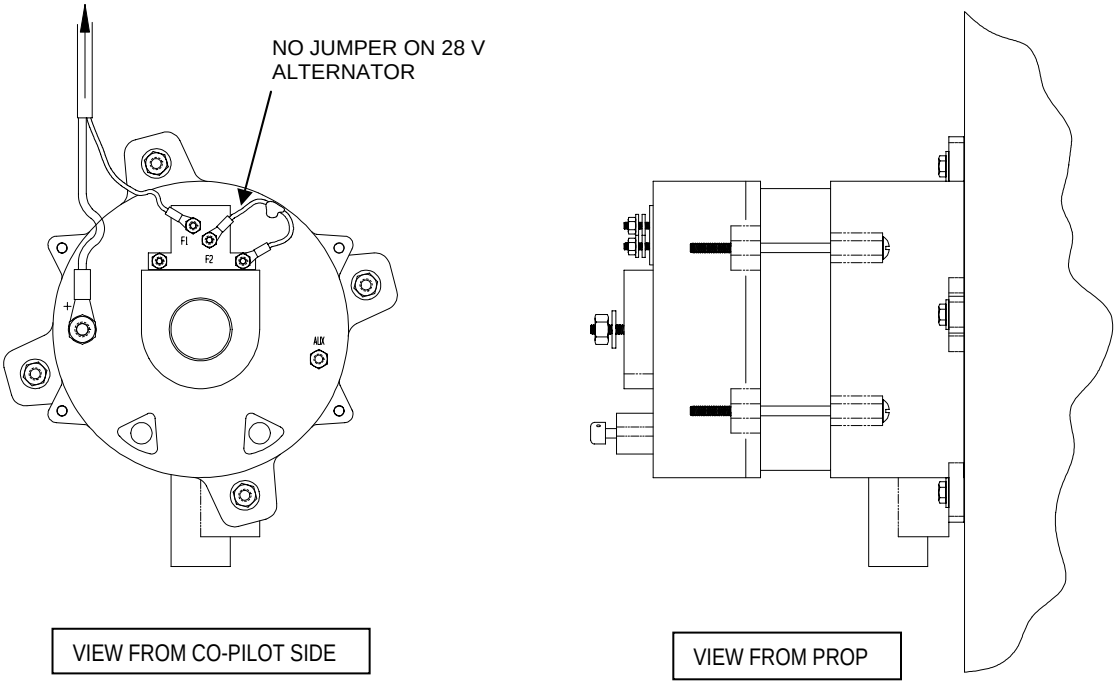


Figure 24 - 5 No. 1 (Right) Alternator

3. **Maintenance Procedures**
   a) The torque required to slip a new elastomer coupling must be a minimum of 180 in.-lbs. (15 ft.-lbs.) torque measured after 45 degrees of revolution at a rate of 1 to 2 degrees per second. If you can hand-turn the gear relative to the black housing, it is below the requirements, and the coupling assembly must be replaced. See Figure 24 - 6.
   b) Slippage must occur at the outside diameter of the elastomer with no damage to the elastomer.
   c) On couplings that have been in service for more than 25 hours, slippage torque must not be less than 140 in.-lbs. (11.7 ft.-lbs.) torque. If you can hand-turn the gear relative to the black housing, it is below the requirements, and the drive hub assembly must be replaced. See Figure 24 - 6.

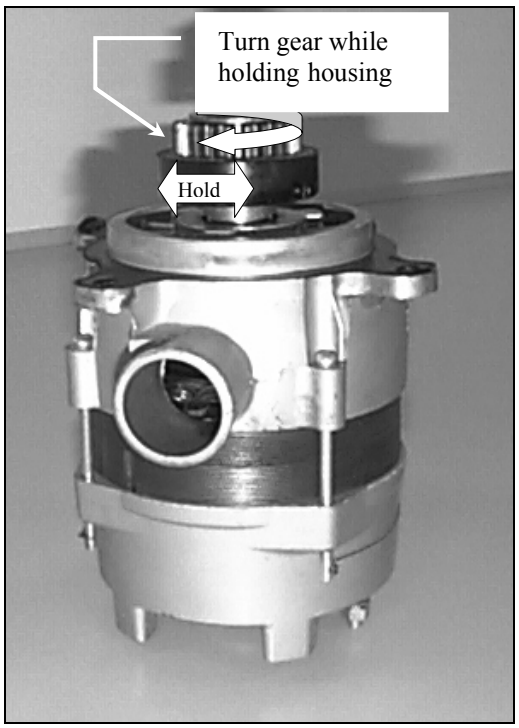


Figure 24 - 6 Alternator Drive Hub

4. **Alternator Brushes** – Every 1000 hours inspect the alternator brushes. The alternator must be replaced if the brushes are worn more than 75 %.
   a) Disconnect the battery wires connected to the alternator.
   b) Remove the two brush block retainer screws and washers.
   c) Remove the brush block. The brushes are spring loaded and may fly out of the brush block.
   d) Inspect the brushes.
   e) Reinsert the brushes into the brush block. Depress the brushes, thus loading the springs, and insert a rod or wire through the hole above the brush seats to hold the brushes in place.
   f) Install the brush block into the alternator.
   g) Withdraw the rod or wire and listen for a “click” to indicate the brush has properly seated. There should be one click per brush.

h) Repeat steps v through vii until the brushes seat properly.
i) Replace the retaining screws and washers and torque per alternator manufacturer's specifications.
j) Connect the battery wires.

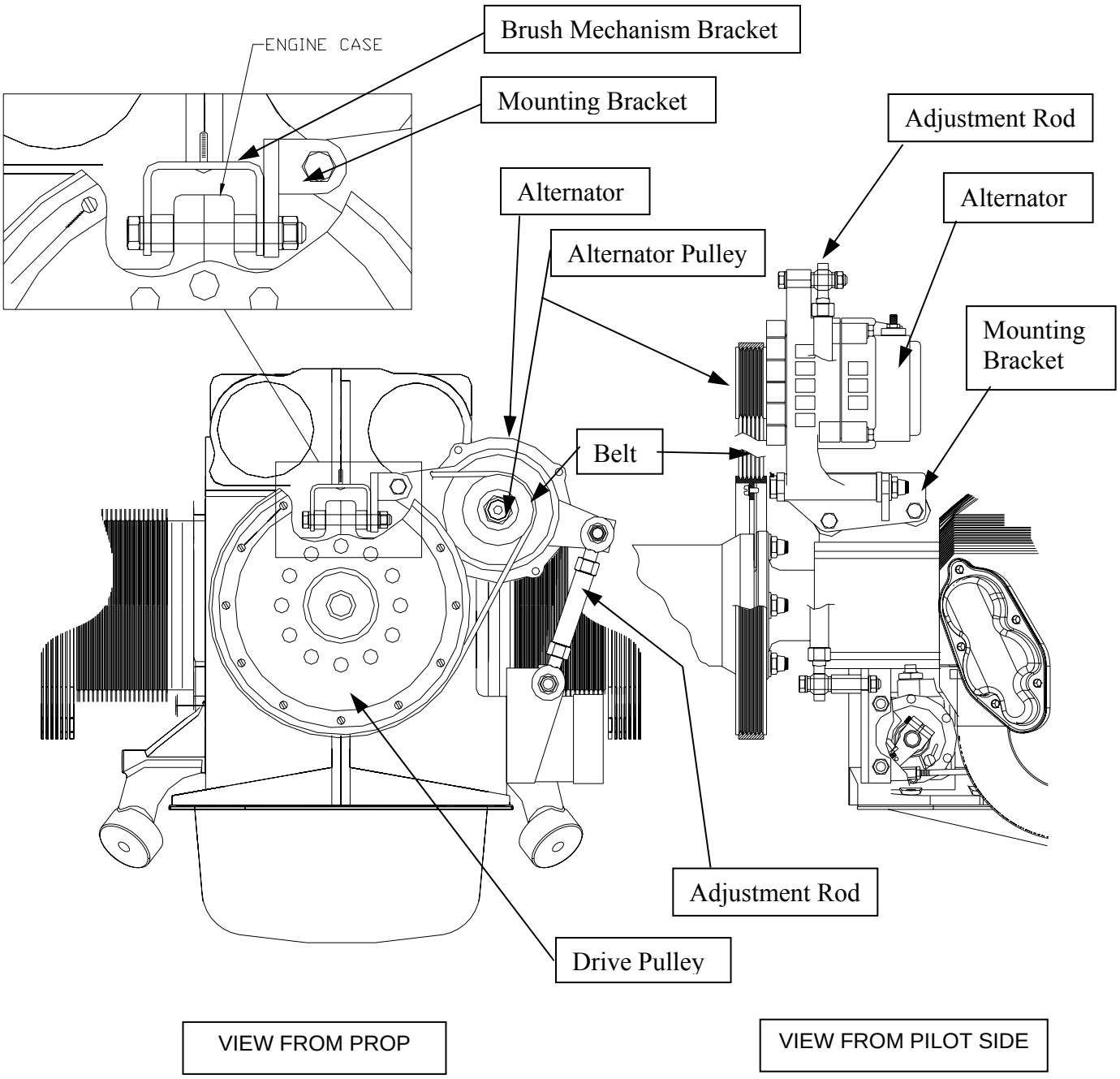


Figure 24 - 7 No. 2 (Left) Alternator with Adjustment Rod

b. **No. 2 (Left) Alternator with Adjustment Rod**– See Figure 24 - 7.

1. **Removal**
   a) Disconnect the battery wires attached to the alternator.
   b) Loosen the adjustment rod and remove the belt from the alternator pulley. The adjustment rod ends are threaded such that rotating the rod itself will loosen or tighten the ends equally.
   c) Remove the self locking nut, bolt, and washers connecting the adjustment rod to the alternator.
   d) Remove the self locking nuts, bolts, washers and spacers connecting the alternator mounting bracket and the brush mechanism bracket to the engine.
   e) Remove the self locking nut, bolt, washer, and spacer connecting the alternator to the mounting bracket.
   f) Retain all hardware for use in reinstallation.
2. **Installation**
   a) Attach the alternator to the mounting bracket.
   b) Attach the alternator mounting bracket and the brush mechanism bracket to the engine. Torque 180 to 220 lb.-in. Ensure that a minimum of 2 threads are exposed through the self-locking nut after final installation.
   c) Align belt onto drive pulley and alternator pulley.
   d) Ensuring that the adjustment rod ends are assembled into the rod equal distance attach the adjustment rod to the alternator.
   e) Connect a torque wrench to the alternator pulley retention nut. With an assistant holding the propeller, to prevent the engine crank shaft from rotating, tighten the adjustment rod until the belt no longer slips on the alternator pulley and force applied to the torque wrench is 13 to 17 ft.-lbs. Do not over tighten belt. After final belt adjustment has been made torque jam nut on both ends of adjustment rod and nut attaching the adjustment rod to the alternator 15 to 20 ft.-lbs.
   f) After the belt is tensioned torque the nut attaching the alternator to the mounting bracket 40 to 45 ft.-lbs.

WARNING

Ensure that there is a minimum clearance of 1/2 in. between the upper cowling skin and the alternator case, fan, belt, and pulley.

   g) Connect the battery wires to the alternator.
3. **Alternator Pulley**
   a) **Removal**
      (a) Disconnect the battery wires attached to the alternator.
      (b) Loosen the adjustment bracket and remove the belt from the alternator pulley.
      (c) Remove the pulley retention nut and washer from the alternator drive shaft.
      (d) Remove the pulley.

b) **Installation**
   (a) Installation of the pulley is the reverse of removal.
   (b) Torque the pulley retention nut 35 to 45 ft.-lbs.
   (c) Connect the battery wires to the alternator.
   (d) Reinstall and tension the alternator belt per paragraphs 24-3.b.2.e) and 24-3.b.2.f) above.

4. **Drive Pulley Removal and Installation**
   a) Drive pulley removal and installation is as described in Chapter 61.

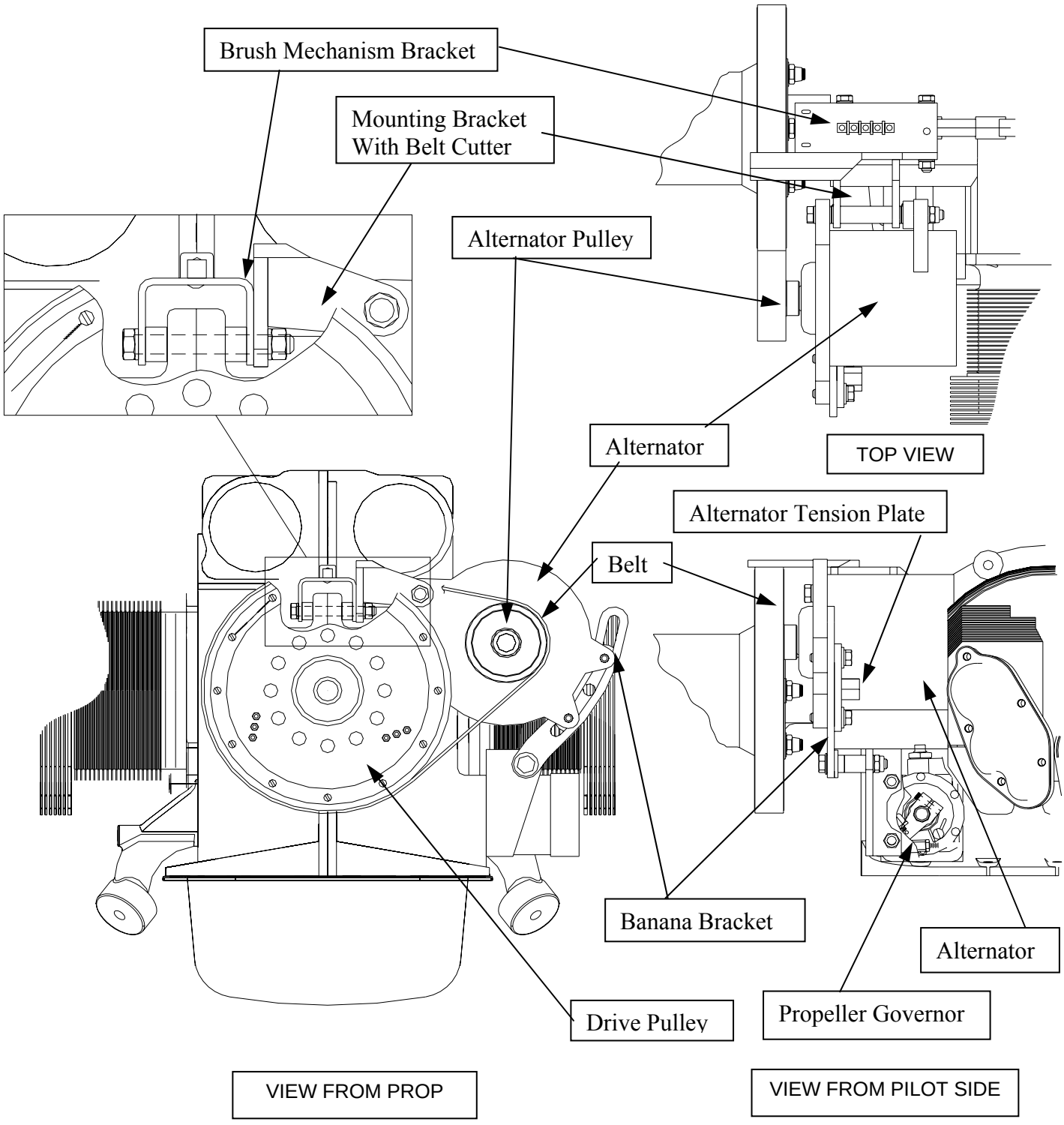


Figure 24 - 8 No. 2 (Left) Alternator with Banana Bracket

c. **No. 2 (Left) Alternator with Banana Bracket** – See Figure 24 - 8.

1. **Removal**
   a) Disconnect the wires attached to the alternator.
   b) Remove the two bolts, washers, and alternator tension plate attaching the alternator to the banana bracket.
   c) Remove the belt from the alternator pulley.
   d) Remove the self locking nut, bolt, and washers connecting the alternator to the mounting bracket. Note the position of the washers for reinstallation.
   e) Remove the alternator.
   f) Retain all hardware for use in reinstallation.
2. **Installation**
   a) Attach the alternator to the mounting bracket. Ensure the washers are installed so that the edge of the belt is parallel to the face of the drive pulley by ± 1.0 degree.
   b) Install the alternator belt.
   c) Connect the wires to the alternator and torque the attachment nuts 10 to 12 in.-lbs.
   d) Insert the alternator tension plate into the banana bracket and hand tighten the hardware attaching the assembly to the alternator.
   e) Connect a torque wrench to the alternator pulley retention nut. With an assistant holding the propeller, to prevent the engine crank shaft from rotating, tighten the alternator tension plate until the belt no longer slips on the alternator pulley and force applied to the torque wrench is 13 to 17 ft.-lbs. Do not over tighten the belt. Torque the alternator tension plate bolts 12 to 15 ft.-lbs. and ensure that a minimum of two threads extend past the alternator body.
   f) After the belt tension has been fully adjusted, torque the nut attaching the banana bracket to the propeller governor bracket and the nut attaching the alternator to the mounting bracket 15 to 20 ft.-lbs.

WARNING

Ensure that there is a minimum clearance of 1/2 in. between the upper cowling skin and the alternator case, fan, belt, and pulley after final assembly.

3. **Alternator Pulley**
   a) **Removal**
      (a) Disconnect the battery wires attached to the alternator.
      (b) Remove the bolts attaching the alternator tension plate to the banana bracket and remove the belt from the alternator pulley.
      (c) Remove the pulley retention nut and washer from the alternator drive shaft.
      (d) Remove the pulley.
   b) **Installation**
      (a) Installation of the pulley is the reverse of removal.
      (b) Torque the pulley retention nut 35 to 45 ft.-lbs.

(c) Connect the battery wires to the alternator.
(d) Reinstall and tension the alternator belt per paragraphs 24-3.c.2.e). and 24-3.c.2.f). above.

4. **Drive Pulley Removal and Installation**
   a) Drive pulley removal and installation is as described in Chapter 61.

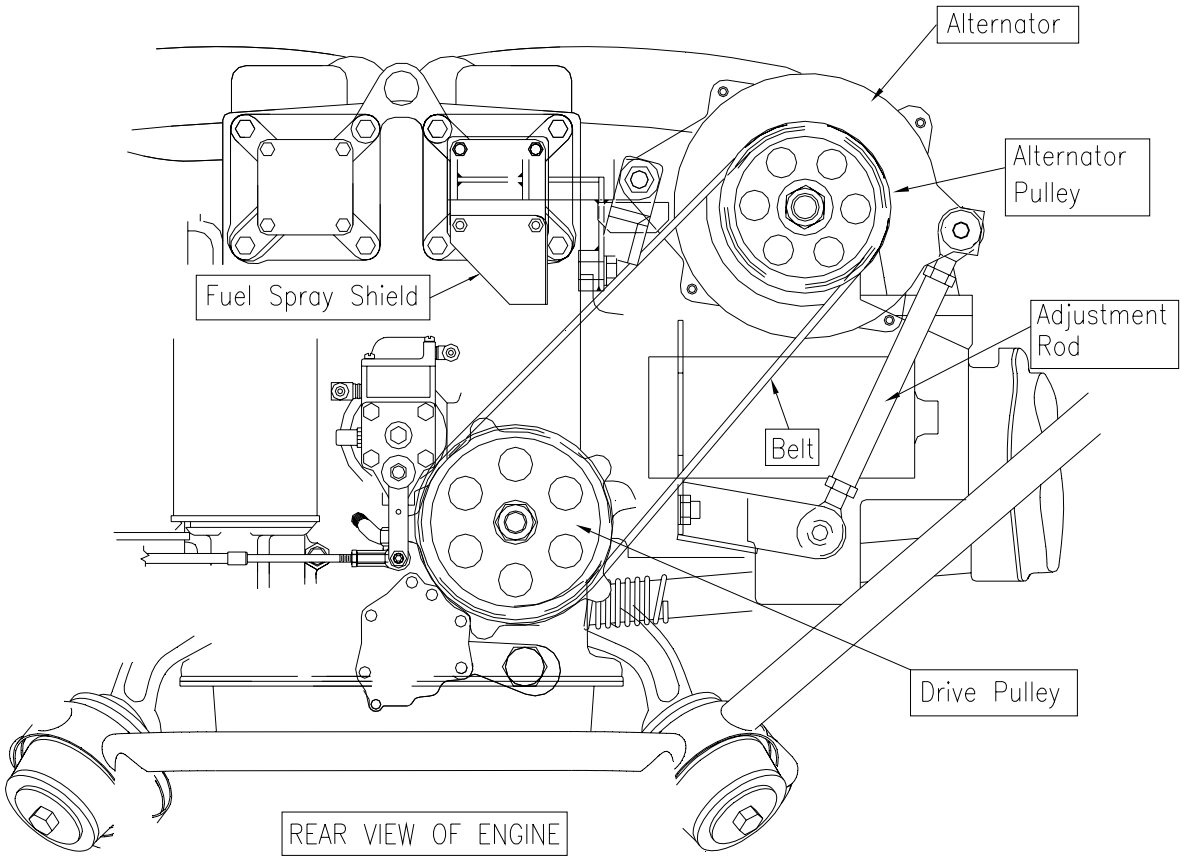


Figure 24 - 9 Accessories Alternator

d. **Accessories Alternator –** See Figure 24 - 9.
   1. **Removal**
      a) Disconnect the battery wires attached to the alternator.
      b) Loosen the adjustment rod and remove the belt from the alternator pulley. The adjustment rod ends are threaded such that rotating the rod itself will loosen or tighten the ends equally.
      c) Remove the bolt and washer connecting the adjustment rod to the alternator.
      d) Remove the nut, bolt, and washer connecting the alternator to the mounting bracket. Remove the belt cutter.
      e) Retain all hardware for use in reinstallation.
   2. **Installation**
      a) Ensure the alternator cooling fan kit, spacer, and alternator drive pulley are installed on the alternator. Verify the nut is torqued 350 to 400 in-lbs.

b) Place the alternator into the alternator bracket on the rear of the engine. Insert the bolt through the alternator pivot. Place the belt cutter, washer, and nut onto the pivot bolt. Do not tighten at this time.
c) Align the belt onto drive pulley and alternator pulley.
d) Ensuring that the adjustment rod ends are assembled into the rod equal distance, attach the adjustment rod to the alternator. Tighten the bolts. For the bolt attaching the alternator to the adjustment rod, protrusion of two bolt threads is not required and no visible female threads on the alternator are allowed; lockwire the bolt.
e) Tighten, or loosen, the adjustment rod until the breakaway (slip) torque at the alternator drive pulley is 440 to 480 in.-lbs. on a new belt. After the belt has been run for a few hours the minimum belt tension should not be below 300 in.-lbs. After final belt adjustment has been made the torque jam nuts on the adjustment rod to hold the proper adjustment.

CAUTION

Do not use the alternator drive nut to check belt tension. Use Kelly Aerospace tool to check alternator belt breakaway torque.

f) Connect the battery wires to the alternator.

3. **Alternator Pulley**
   a) **Removal**
      (a) Disconnect the battery wires attached to the alternator.
      (b) Loosen the adjustment bracket and remove the belt from the alternator pulley.
      (c) Remove the pulley retention nut and washer from the alternator drive shaft.
      (d) Remove the pulley.
   b) **Installation**
      (a) Installation of the pulley is the reverse of removal.
      (b) Torque the pulley retention nut 350 to 400 in.-lbs.
      (c) Connect the battery wires to the alternator.
      (d) Reinstall and tension the alternator belt per paragraph 24-3.d.2 above.
4. **Drive Pulley Removal and Installation**
   a) Remove the nut and lock washer from the starter adaptor drive shaft.
   b) Remove the alternator drive pulley.
   c) Installation is the reverse of removal. Torque the nut 700 to 720 in.-lbs. Ensure a minimum clearance or .050 in. between the drive pulley and the end of the starter adapter stud.

e. **Pulley Maintenance Procedures**
   1. The pulleys should be free from large nicks, scratches and excessive warpage. If warpage exceeds .050" the pulley must be replaced. No shimming is permissible. Small scratches and nicks are acceptable as long as they do not present a hazard to the belt. Blending and burnishing of small scratches and nicks is permissible.

2. The drive pulley for the No. 2 alternator must be replaced if elongated screw holes are present. The maximum elongation permissible is 1/4 diameter where run out is within .050" maximum (.025" either way).
3. If there is excessive black, sooty residue on the pulleys or surrounding area, immediately remove the belt from service and assure that the pulleys are properly aligned.
4. The belt must be replaced if there is any evidence of fraying or dryrot.
5. Do not apply "belt treatments" spray or otherwise to the belt. Any spillage of ester based oils, turbine oils or MIL-H-5606 hydraulic fluid on the belt must be immediately removed with hot soapy water (household dishwashing soap or equivalent).

f. **Alternator Brushes** – Every 1000 hours inspect the alternator brushes. The alternator must be replaced if the brushes are worn more than 75 %.
1. Disconnect the battery wires connected to the alternator.
2. Remove the bolts attaching the alternator tension plate to the banana bracket or loosen the adjustment rod and remove the belt from the alternator pulley.
3. Loosen the self locking nut connecting the alternator to the mounting bracket and rotate the alternator upward allowing access to the alternator brush block.
4. Remove the two brush block retainer screws and washers.
5. Remove the brush block. The brushes are spring loaded and may fly out of the brush block.
6. Inspect the brushes.
7. Reinsert the brushes into the brush block. Depress the brushes, thus loading the springs, and insert a rod or wire through the hole above the brush seats to hold the brushes in place.
8. Install the brush block into the alternator.
9. Withdraw the rod or wire and listen for a "click" to indicate the brush has properly seated. There should be one click per brush.
10. Repeat steps 7 through 9 until the brushes seat properly.
11. Replace the retaining screws and washers and torque per alternator manufacturer's specifications.
12. Connect the battery wires to the alternator.
13. Reinstall and tension the alternator belt per paragraphs 24-3.c.2.e). and 24-3.c.2.f), or 24-3.d.2.e) above.

24-4. NORMAL BATTERY SYSTEM (S/N 42001 to 42500)

a. **Applicability –** This procedure is based on the Concorde™ RG-1215 batteries. Refer to battery manufacturer's instructions for batteries other than the one specified.

b. **Removing Battery**. See Figure 24 - 10 and Figure 24 - 12.

CAUTION

To reduce the chance of personal injury and possible equipment damage, remove the negative wire before removing the positive wire.

1. For S/N 42001 to 42022, loosen the three bolts securing the battery box lid to the battery box, and open the lid. For S/N 42023 and on, remove the four bolts securing the battery box cover to the battery box and remove the cover.
2. Disconnect the negative connector, then the positive connector for each battery. See detail of battery cable attachment in Figure 24 - 11.
3. Remove the battery.

c. **Installing Battery**

1. Place the battery in the battery box.

WARNING

Failure to reconnect the battery connectors properly could result in reversing the polarity of the battery.

2. Reconnect positive connector, then connect the negative connector for each battery. Torque bolts 36 to 40 in.-lbs. See detail of battery cable attachment in Figure 24 - 11.
3. For S/N 42001 to 42022, close the battery box cover over the batteries, and install the three bolts. Torque 30 to 36 in.-lbs. For S/N 42023 and on, replace the battery box cover and install the four bolts. Hand tighten the bolts until snug.

CAUTION

Do not replace the sealed lead acid batteries with wet lead acid batteries

NOTE

Do not use any type of material or sealant to seal holes or gaps in the battery box, or the open area formed where battery wiring passes through a grommet. These spaces must remain clear to ensure ventilation of the batteries.

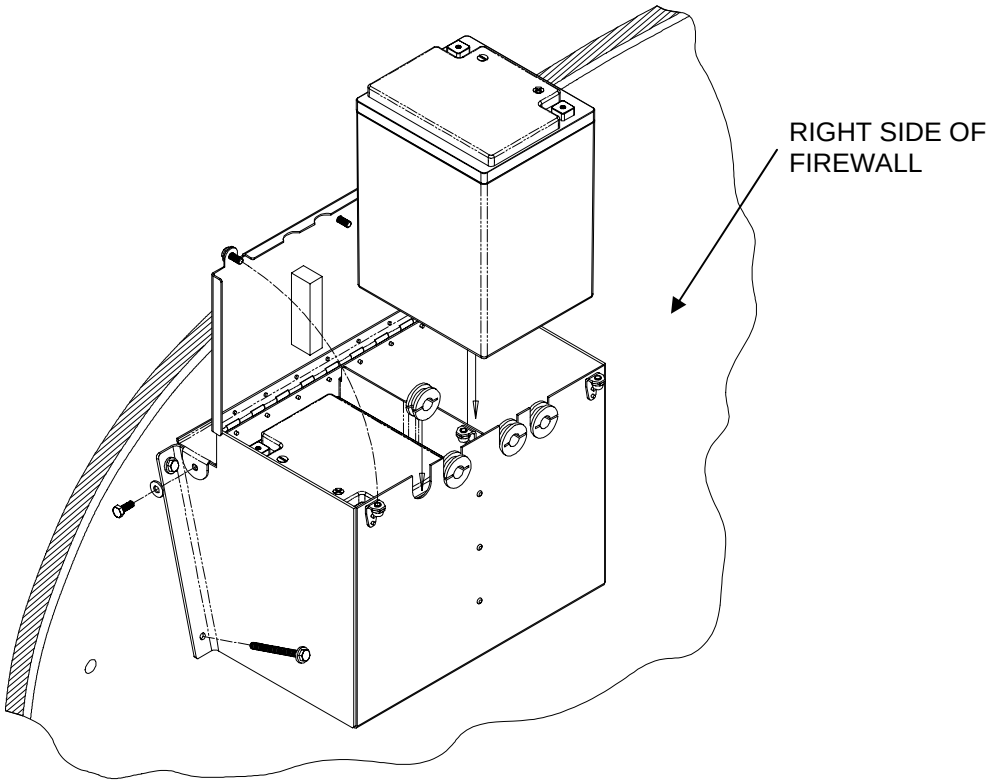


Figure 24 - 10 Battery Installation (S/N42001 to 42022)
(Left Battery Shown)

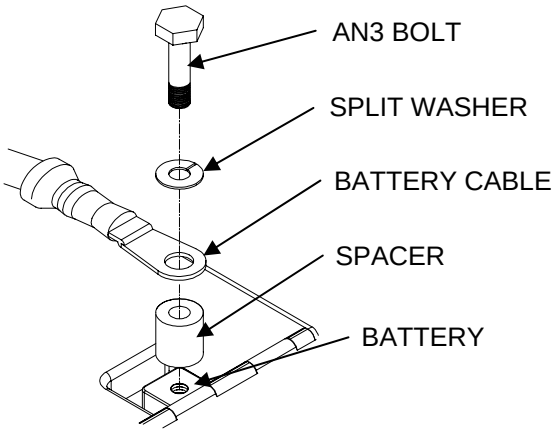


Figure 24 - 11 Battery Cable Attachment (S/N 42001 to 42500)

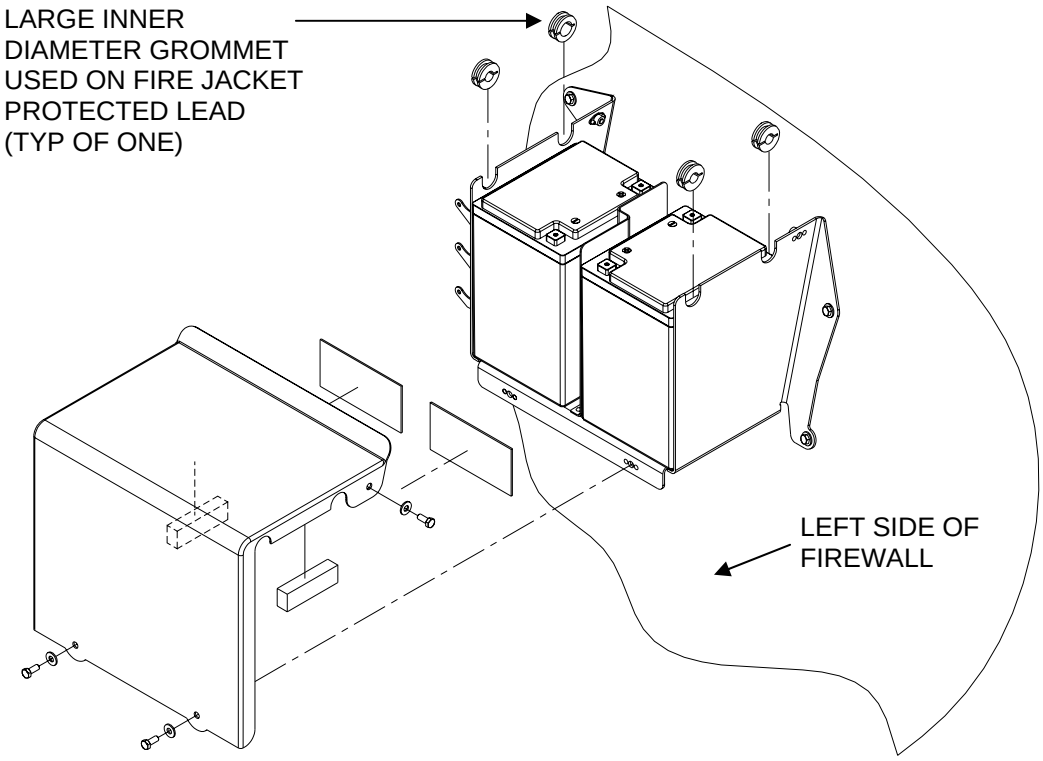


Figure 24 - 12 Battery Installation (S/N42023 to 42500)

d. **Test Procedure for Reserve or Emergency Capacity**
   1. Make sure the battery is charged.
   2. With the battery temperature above 59°F, discharge the battery at the 80-85% test rate [reference FAR 23.1309, 23.1351, 23.1353(h)].
   3. The minimum end point after one hour of the time established for the airframe essential power requirement of discharge must be 9 volts. If the battery fails to deliver the above rated ampere-hour capacity, it has reached its end of life. The battery is no longer considered airworthy and must be replaced.
   4. Allow the battery to cool to room temperature before recharging.

e. **Maintenance Procedures for Battery**
   1. The batteries must be boost charged every 90 days when in storage.
   2. Batteries that have not been recharged every 90 days are to be conditioned. First discharge the battery at the test rate. Recharge with a constant current charge at the C/10 rate (1/10 of the C rate amperes) for eighteen hours or until the battery voltage reaches 2.5 volts per cell. Allow the battery to rest for at least one hour and re-test per paragraph 3, above.
   3. The battery should be recharged when the open circuit voltage is below 2.08 volts per cell.

4. The battery should be recharged at $^1/_2$ of the C rate with a constant potential (CP) or constant voltage charger regulated at 2.35 volts per cell. The battery is charged when the charge current stabilizes for one hour.
5. It is recommended the emergency capacity test procedure be performed after twelve months or six hundred hours of operation, whichever comes first. To ensure continued airworthiness, another emergency capacity test should be performed every twelve months or every two hundred hours.

24-5. NORMAL BATTERY SYSTEM (S/N 42501 and on)

a. **Applicability –** This procedure is based on batteries approved by Cessna. Refer to battery manufacturer's instructions for batteries other than those approved by Cessna.

A battery charging circuit has been added to the power grid on aircraft built on or after mid 2007, retrofit available S/N 42501 and on, that will energize the battery relays and allow ground power charging of flat or discharged batteries without removing the battery(ies) from the aircraft. Batteries will start charging when the ground power unit is connected and the master and crosstie switches are ON. The master and crosstie switches must be turned off before removing the ground power plug. Flat or discharged batteries on aircraft without the battery charging circuit must be removed from the aircraft before charging. The circuit does not affect the operation of the master and crosstie switches.

NOTE

Batteries that are suspected to be unserviceable must be removed from the aircraft and serviced or replaced.

b. **Removing Battery**. See Figure 24 - 13.

CAUTION

To reduce the chance of personal injury and possible equipment damage, remove the negative wires before removing the positive wires.

1. Remove the seven bolts and washers securing the battery box cover to the battery box and remove the cover.
2. Disconnect the negative connector, then the positive connector for each battery. See detail of battery cable attachment in Figure 24 - 14.
3. For each battery, remove the bolt and washer attaching the battery hold down and remove the hold down.
4. Remove the battery.

c. **Installing Battery**

1. Place the battery in the battery box.

WARNING

Failure to reconnect the battery connectors properly could result in reversing the polarity of the battery.

2. Reinstall the battery hold down and tighten the bolt snug.

3. Reconnect positive connector, then connect the negative connector for each battery. Torque nuts or bolts, as applicable, 36 to 40 in–lbs. See detail of battery cable attachment in Figure 24 - 14.
4. Replace the battery box cover and secure with bolts and washers. Tighten bolts snug.

CAUTION

Do not replace the sealed lead acid batteries with wet lead acid batteries.

CAUTION

The battery cables must be tied to tie blocks to keep them from chafing against the cowling.

NOTE

Do not use any type of material or sealant to seal holes or gaps in the battery box or the open area formed where battery wiring passes through a grommet. These spaces must remain clear to ensure ventilation of the batteries.

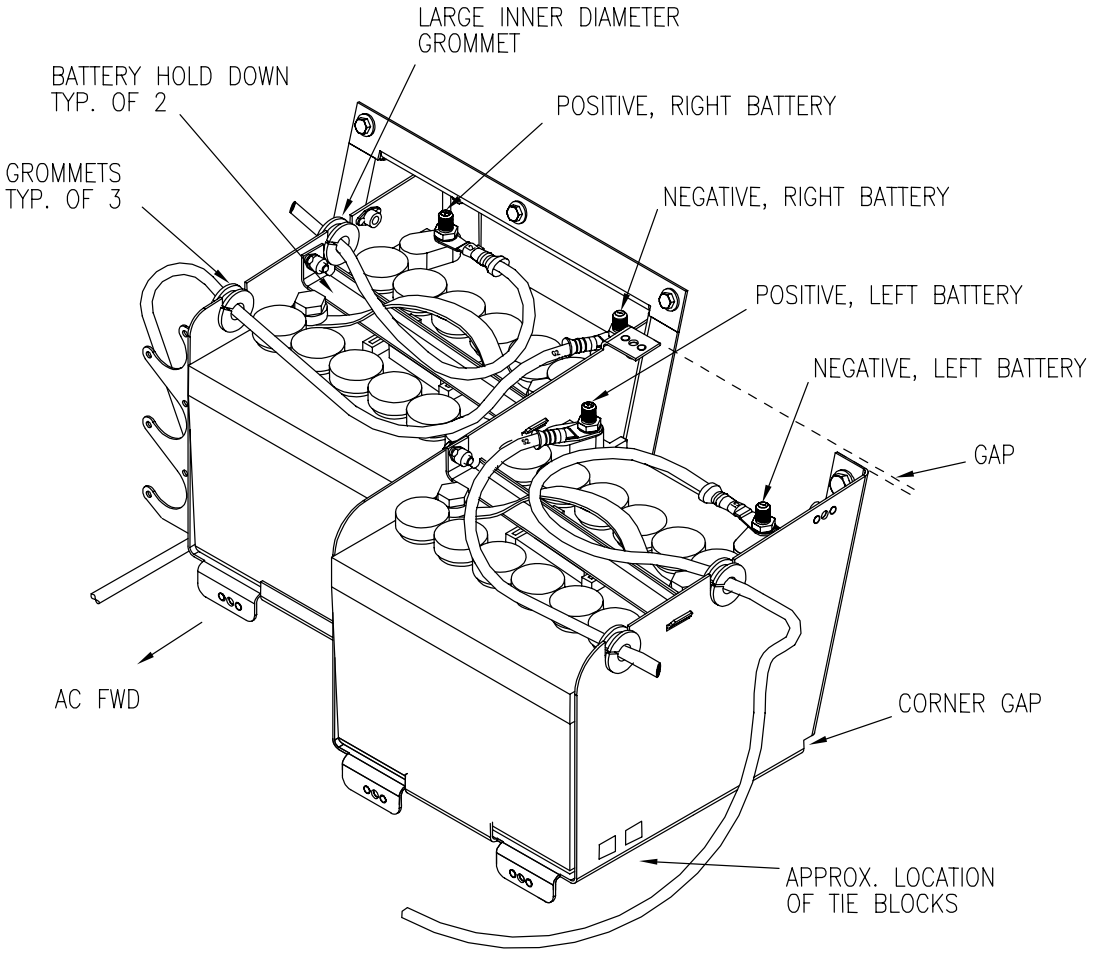
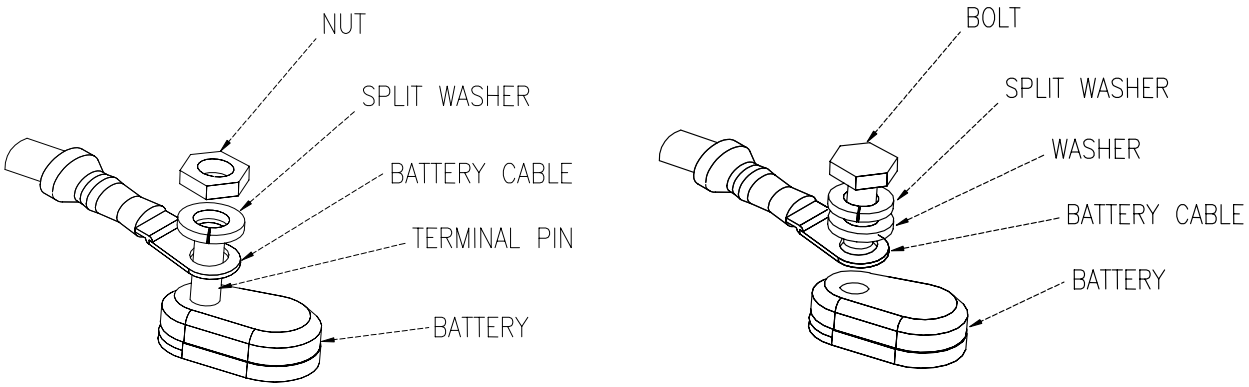


Figure 24 - 13 Battery Installation (S/N 42501 and on (Cover Not Shown)



Gill Battery **Concorde Battery**

Figure 24 - 14 Battery Cable Attachment (S/N 42501 and on)

d. **Battery Charging –** Use this procedure if the aircraft is not equipped with the battery charging circuit, is equipped with the circuit but ground power is not available, or the battery requires bench charging.

1. Remove the battery from the aircraft.
2. Using an AA Portable Power Corp. Smart Charger 24V 1.5 Amp. constant current/constant voltage 3 stage battery charger Model No. CH-LA2415, or equivalent, connect the charger to the battery ensuring correct polarity.
3. Connect the charger to AC power. The following will result:

   - The green LED will turn orange.
   - When the LED is orange the battery is charging.

CAUTION

Ensure the battery does not overheat. The battery temperature, measured at the case surface, should not exceed 100º F.

4. When the LED turns green the battery is fully charged.
5. The Smart Charger may be left on the battery after the battery is full charged.
6. After the battery is fully charged, turn off the Smart Charger by removing the AC power plug.
7. Disconnect the battery.
8. Allow the battery to cool down to room temperature.

e. **Battery Discharge Test**

1. Fully bench charge the battery.
2. Prior to connecting to the battery, ensure that the load tester is off.
3. The battery should be discharged at the 1 hour rate (check the battery amphour rating) to an end voltage of 1.67 volts per cell or 20 volts. Measure the time. The battery must achieve at least 80% of the rated time (or 48 minutes at the 1 hour rate). If the first discharge time is less than 48 minutes, repeat the cycle one more time.

4. If the second discharge fails to deliver at least 48 minutes, the battery should be rejected. Return the battery to the vendor for servicing or replacement if the warranty is still valid. If not, scrap the battery.
5. Once the battery has passed all required inspections and after it is fully recharged, the battery is ready for installation.

f. **Battery Maintenance**

1. After initial installation, a capacity check of the battery is required at 800 hours or 11 ± 1 months, whichever comes first with subsequent capacity checks performed every 400 hours or 6 ± 1 months.

WARNING

The battery must be removed from the installation and serviced in a well ventilated designated area. During servicing, the battery will generate oxygen and hydrogen gases, which can be explosive under the right conditions.

2. Visually inspect the battery for any signs of cracks, corrosion, unusual terminal pin wear or discoloration on the pins.

WARNING

These batteries contain sulfuric acid which is highly corrosive and which can cause serious physical injury if it comes in contact with skin or if inhaled. It can also cause serious eye injury or blindness if it comes into contact with the eyes.

Caution must be exercised to avoid damage to the exterior case which could allow the contents to escape or come in physical contact with external materials or personnel.

If a battery case is found to be damaged, handle the battery with care and avoid contact with the skin. Inspect all areas adjacent to the battery for evidence of corrosion.

3. Before reconnecting to the aircraft, ensure the terminals have not worn or become loose.

g. **Battery Reconditioning –** Delaying the recharge of even a partially discharged battery can make it resist the charger for some time. When put on a charger the displays may indicate that the battery is fully charged, but when you attempt to use it again it acts as if completely discharged.

Recharging immediately after use will significantly increase the service life of the battery. It is recommended that the battery be recharged within 24 hours of being discharged. The amount of capacity regained with each cycle will eventually level out and this stabilization is an accurate indication of the battery's useful capacity.

1. Charge the battery for 24 hours to ensure that it has gone through a complete charging cycle.

2. Perform a discharge test.
3. If the battery fails the discharge test, charge the battery again for 24 hours.
4. Perform a discharge test.
5. Repeat this charge/discharge process 4 or 5 times. The capacity of the battery should increase appreciably with four to five charge/discharge cycles. If the battery still fails to measure up to capacity, return the battery to the vendor for servicing or replacement if the warranty is till valid. If not, scrap the battery.

24-6. DC INDICATING INSTRUMENTS – The following information applies to aircraft equipped with the Basic or Avidyne avionics option only. DC monitoring for aircraft equipped with the Garmin G1000 option is displayed on the MFD.

a. **Dual Ammeter/Loadmeter Description of Operation** – The ammeter is in the engine instrument panel in the center-left position. The ammeter is a dual presentation gauge that measures the condition of both batteries and alternators in terms of charging or discharging.

The ammeter is selectable to show either the condition of the batteries or the alternators. When power is first applied, the indicator defaults to the "BATT" indicator. When the button on the lower left side of the instrument is pressed, the indicator switches to the "ALT" indicator. Pressing the button will toggle between each indication. When the battery mode is selected, "BATT" is illuminated by a white light. When the alternator mode is selected, "ALT" is illuminated by a white light.

The range of the indications run from a + 60 amps to – 100 amps in 30 amp increments. While there is no placarded operating range, under most conditions the instrument should indicate a positive charging state. The master switches for the left and right bus systems must be on for the ammeter to operate.

b. **Removal of Dual Ammeter/Loadmeter**
   1. Remove the engine instrument panel per instructions in Chapter 25.
   2. Unplug the cable going into the back of the ammeter.
   3. Unscrew the four screws surrounding the ammeter.

c. **Installation of Dual Ammeter/Loadmeter**
   1. Put the ammeter in its slot, and screw in the four screws attaching the instrument to the engine instrument panel.
   2. Plug the cable into the back of the ammeter.
   3. Install the engine panel per instructions in Chapter 25.

d. **Voltmeter Description of Operation (S/N 42001 to 42500)** – The voltmeter is located in the top left corner of the flight instrument panel. This instrument contains three separate indications: a voltage reading, an outside air temperature provided by the top window, and a multi function timepiece in the lower window. The voltmeter displays the left system's bus voltage only, and under normal conditions should indicate about 14.2 volts. At 16 volts, the voltage regulator will take the left alternator off-line, and at approximately 8 volts, electrical equipment connected to the left bus will cease to operate or will operate erratically. The voltmeter is electrically powered and will not operate if the left master switch is turned off. See Chapter 31 for a drawing of the voltmeter. This voltmeter is not present on aircraft S/N 42501 and on.

e. **Removal of Voltmeter**
   1. Remove the glare shield, and remove the connector installed into the back of the unit.
   2. Unscrew the four screws surrounding the voltmeter, and remove it.

3. Slide the unit out of the flight instrument panel.

f. **Installation of Voltmeter**

1. Install the connector into the back of the voltmeter.
2. Slide the unit into its slot in the flight instrument panel, and screw in the four screws.
3. Install the glare shield.

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24-7. GROUND POWER PLUG

a. **Removal and Replacement of the Ground Plug Relay**

1. Remove the cover from the power grid by removing the two 8-32 pan head screws and washers.
2. Verify that the aircraft power is off.
3. Remove the large power cables and mark the location.
4. Remove the relay.
5. Install a new transorb (surge suppressor) on the new relay.
6. Move the reverse current diode to the new relay.
7. Reassemble in the reverse order.

b. **Removal and Replacement of the Power Plug Receptacle (S/N 42001 to 42500)**

1. Turn the aircraft master switch on and fully extend the flaps.
2. Turn the aircraft master switch off and placard “Do Not Turn On Power”.
3. Remove the four 8-32 screws attaching the step cover, and remove the cover.
4. Remove two ¼-28 screws, washers, and self-locking nuts attaching the receptacle (see Figure 24 - 15).
5. Pull the power plug receptacle down to gain access to the solder joint.
6. Unsolder with a propane torch adjusted to low.
7. Re-solder a new plug, and slide the boot over the solder joint.
8. Reassemble in reverse order. Torque attachment screws 140 to 180 in.-lbs.

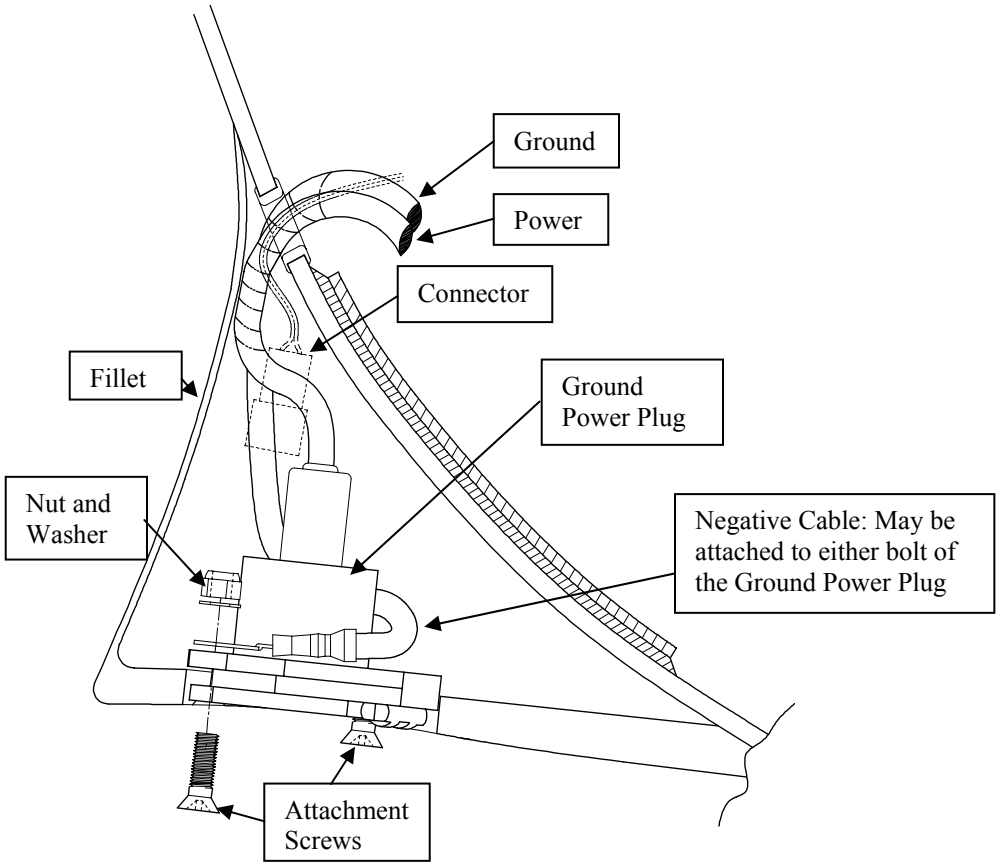

Figure 24 - 15 Ground Power Plug Installation (S/N 42001 to 42500)

c. **Removal and Replacement of the Power Plug Receptacle (S/N 42501 and on))**

1. Turn the aircraft master switch off and placard "Do Not Turn On Power".
2. Remove the hat rack carpet per instructions in Chapter 25.
3. Remove the screws securing the baggage bulkhead access panel and remove the panel.
4. Remove the nuts, lock washers, and washers attaching the cables to the receptacle(see Figure 24 - 16).
5. Remove the four 6-32 pan head screws and washers, attaching the receptacle (see Figure 24 - 16). Note the location of the washers for reinstallation. The purpose of the washers is to align the receptacle outer door flush with the fuselage.
6. Reassemble in reverse order. Torque 6-32 screws 10 in.-lbs.

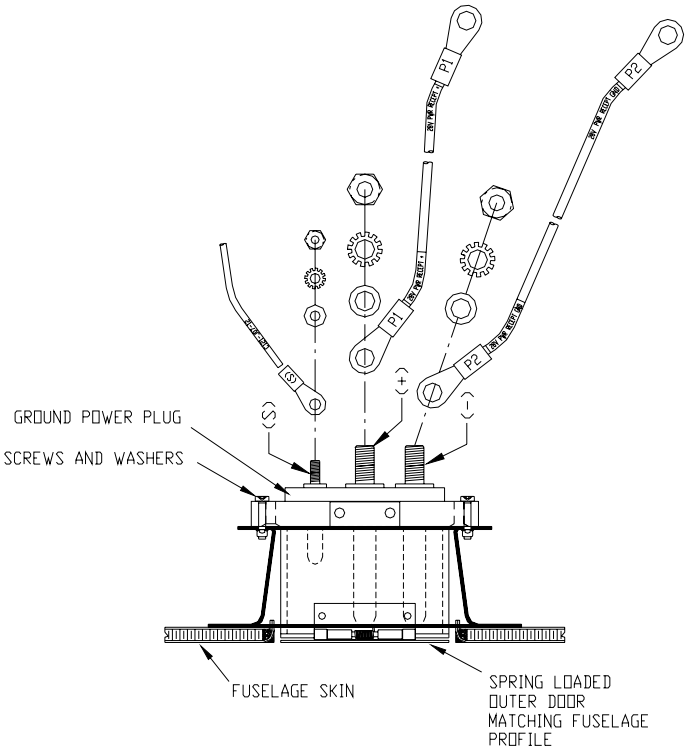

Figure 24 - 16 Ground Power Plug Installation (S/N 42501 and on)

d. **Ground Power Plug Troubleshooting**

1. If the ground power plug relay will not engage:
   a) The reverse current diode mounted on the relay may be open or have loose connections. Replace the relay or tighten the connections.
   b) The transorb (surge suppressor) is shorted. Replace transorb.
   c) Ground at the power plug or at the engine block is disconnected or loose. Reconnect or tighten the ground.
   d) Contamination in the ground power plug does not allow the connector to go in all the way. Replace plug.
   e) Ground Power relay windings are open. Replace relay.
2. Ground power relay will not disengage.
   (a) Relay contacts are welded together internally. Replace relay.
   (b) BATT switches are turned on. Turn off BATT switches.

24-8. ELECTRICAL LOAD DISTRIBUTION

a. **Removal of Voltage Regulator**

1. The voltage regulators are either attached to each other and a metal plate connected to the inside firewall on the copilot's side or attached under the instrument panel on the copilot's side. Remove the glare shield as necessary for access to the regulators.
2. Unhook the wires connected to the voltage regulators.
3. For the regulators attached to the firewall, remove the three pan-head screws attaching the top voltage regulator to the bottom voltage regulator, and remove the top voltage regulator. For each regulator attached under the instrument panel, remove the three pan-head screws and washers and remove the regulator; skip step 4 below.
4. Remove the three standoffs securing the bottom voltage regulator to the metal plate, and remove the voltage regulator from its mounting. See Figure 24 - 17.

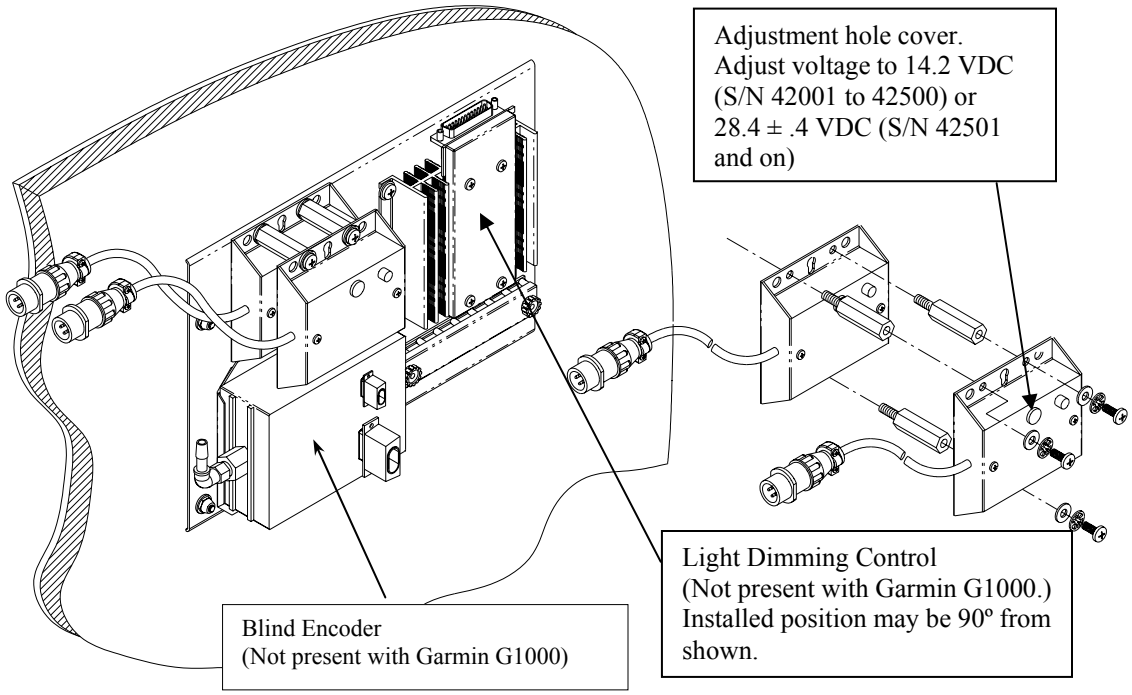

Figure 24 - 17 Voltage Regulator Installation

b. **Installation of Voltage Regulator**

1. For the voltage regulators located under the instrument panel installation is the reverse of removal. For voltage regulators located on the firewall see steps 2 to 4 below.
2. Place the bottom voltage regulator in its mounting.
3. Install the three standoffs to secure the bottom voltage regulator to the plate. The standoffs must protrude through the nutplate flush or better.
4. Place the top voltage regulator over the standoffs and install the three pan-head screws and six washers.
5. Reconnect the wires to the voltage regulator.

c. **Removal of Rocker and Master Switch Panels**
   1. Steps for the removal of the rocker and master switch panels can be found in Chapter 25.

d. **Installation of Rocker and Master Switch Panels**
   1. Steps for the installation of the rocker and master switch panels can be found in Chapter 25.

24-9. CIRCUIT BREAKER PANEL

Steps for the removal and installation of the circuit breaker panel can be found in Chapter 25.

24-10. POWER GRID

The power grid is locate on the firewall in the engine compartment on the pilot's side. The power grid does not require any service or maintenance. However, it should be visually inspected every 1000 hours for frayed or chafed wiring and for evidence of corrosion and/or overheating,

24-11. ELECTRIC A/C INTERLOCK ASSEMBLY

The Electric A/C Interlock Assembly, located under the left side of the baggage shelf, is a component of the air conditioning system (ACCS) when the system has the optional electrically driven compressor. Refer to Chapter 21 for description of the ACCS.

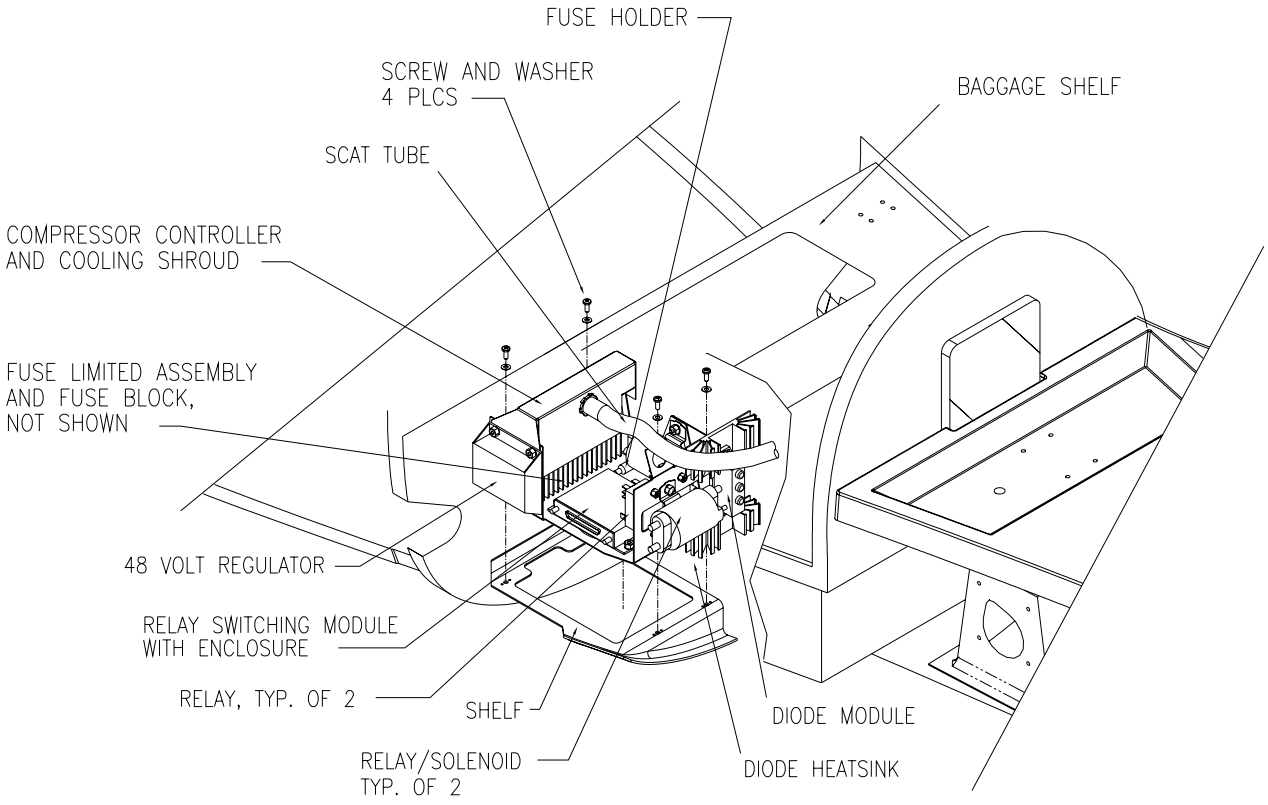

Figure 24 - 18 Electric A/C Interlock Assembly

a. Removal
   1. Remove the baggage shelf carpet per instructions in Chapter 25.
   2. Remove the Elevator Interconnect Access Panel and/or the Electronics Bay Access Panel per instructions in Chapter 25.
   3. Disconnect all electrical connections to the interlock assembly.
   4. Disconnect the 3/4 in. interlock cooling SCAT tube attached to the interlock assembly.
   5. Remove 4 screws and washers attaching the interlock assembly bracket to the shelf.
   6. Remove the interlock assembly.
b. Installation – Installation is the reverse of removal.
c. Maintenance – The Electric A/C Interlock Assembly does not require maintenance.

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CHAPTER 25

EQUIPMENT FURNISHINGS

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Chapter 25

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25-1. GENERAL

a. **Front Seats –** Two individual, adjustable, tubular frame seats provide the front seating for the pilot and passenger. The base of the tubular seat frame is covered with sheet aluminum, and the seat cushions are attached to the aluminum through a series of Velcro strips. The seat backs on the front seats fold forward to permit access to the aft seating area.

The seat cushions and seat backs are foam filled and covered with natural leather and ultra-leather. For added protection, both the front and rear seats incorporate a special rigid, energy absorbing foam near the bottom of the cushion. The cushion is designed for the loads applied by a seated passenger, and it is possible to damage the seat if concentrated loads are applied. Care must be taken to avoid stepping on the seats or placing heavy objects on the seat. Care and maintenance of the seats is discussed in Chapter 12.

The front seats are adjustable fore and aft through a range of approximately 7 in. The adjustment control for the front seats is located below the seat cushion on the left side. To adjust the position of either seat, move the control lever towards the middle until the seat unlocks from the seat track, and adjust the seat to the desired position.

WARNING

Seat cushions are designed to crush to absorb loads in a crash situation. In the event of any incident or accident with a downward force component that results in damage to the aircraft, the seat cushions should be replaced.

WARNING

Do not place anything under the seat pans. The hollow area under the seat is designed for the crush of the seat to absorb energy in the event of a crash. Placing items in this area will prevent the proper operation of the seats.

b. **Rear Seats –** The rear seats are a split bench-type design and are nonadjustable. The bench seat frame is composite construction and bolted to the interior of the fuselage. The foam-filled seat and seat back cushions are covered with natural leather and ultra-leather and attached to the seat bench with Velcro fasteners. There are two non-locking headrests inserted into the top of the seat back. The headrest can be removed by grasping it on each side and pushing up.

c. **Seat Belts and Shoulder Harnesses –** The seat belts and shoulder harnesses are an integrated three-point restraint type of design. With this type of restraint, the lap belt and diagonal harness are one continuous piece of belt webbing. The webbing is anchored on each side of the seat for the lap belt restraint and then on the side of the fuselage for the harness restraint.

d. **Interior Panels –** Most of the interior panels are covered with ultra-leather and are secured with screws, PEM (Penn Engineering & Manufacturing) studs, Shuler clips, Velcro, or standard clips. When all the fasteners are removed, care must be taken when removing the panel so as not to damage the finish. Care and maintenance of the panels is discussed in Chapter 12. The panels are classified according to their location, i.e.,

forward, middle, aft, center, upper, lower, etc. The following is a brief discussion of each panel and the accompanying abbreviation. See Figure 25 – 1.

1. **Forward and Aft Center Overhead Light Panel (FCOLP and ACOLP)** – This panel is located in the center portion of the main cabin area and contains the electrical lights and speaker.
2. **Forward Overhead Side panels Left and Right (FOSPL and FOSPR)** – The forward panels to the left and right of the COLP are referred to as the left and right overhead side panels.
3. **Aft Overhead Side panels Left and Right (AOSPL and AOSPR) –** These side panels are just aft of the FOSPL and FOSPR.
4. **Forward Lower Side Panels Left and Right (FLSPL and FLSPR)** – These panels are located on the left and right side of the airplane and cover each side of the airplane from about the rudder pedals aft to a location near the backrests of the front seats.
5. **Middle Lower Side Panels Left and Right (MLSPL and MLSPR)** – These panels are located on each side of the airplane, between the front and rear seatbacks.
6. **Baggage Lower Side Panel Left and Right (BLSPL and BLSPR)** – These panels are located on the left and right side of the baggage compartment, near the floor, and are also referred to as the left and right push-pull tube channel covers.

e. **Engine, Flight Instrument, and Electrical Panels**

1. **Basic or Avidyne Option** – The flight instrument panel is located directly in front of the pilot, and the engine instrument panel is to the left of the flight instrument panel and is canted at about 30º. On the right side of the airplane there is a kidney panel located in an approximate mirrored location in relation to the engine instrument panel. Below the instrument panel are a series of electrical panels, which contain the master switch, rocker switch, trim, flap, and ECS panels. Below the electrical panels are the left and right knee bolsters. Above the instrument panels is the glare shield. Figure 25 – 2 and Figure 25 – 3 show the location of each panel.
2. **Garmin G1000 Option –** The instrument panel is located directly in front of the pilot. Integral to the instrument panel are the 5 Pack switches, backup airspeed indicator, backup attitude indicator, backup altimeter, PFD, audio panel, and MFD. Below the instrument panel are the flap panel, engine controls, and ECS or ACCS panels. Above the instrument panel is the glare shield. See.

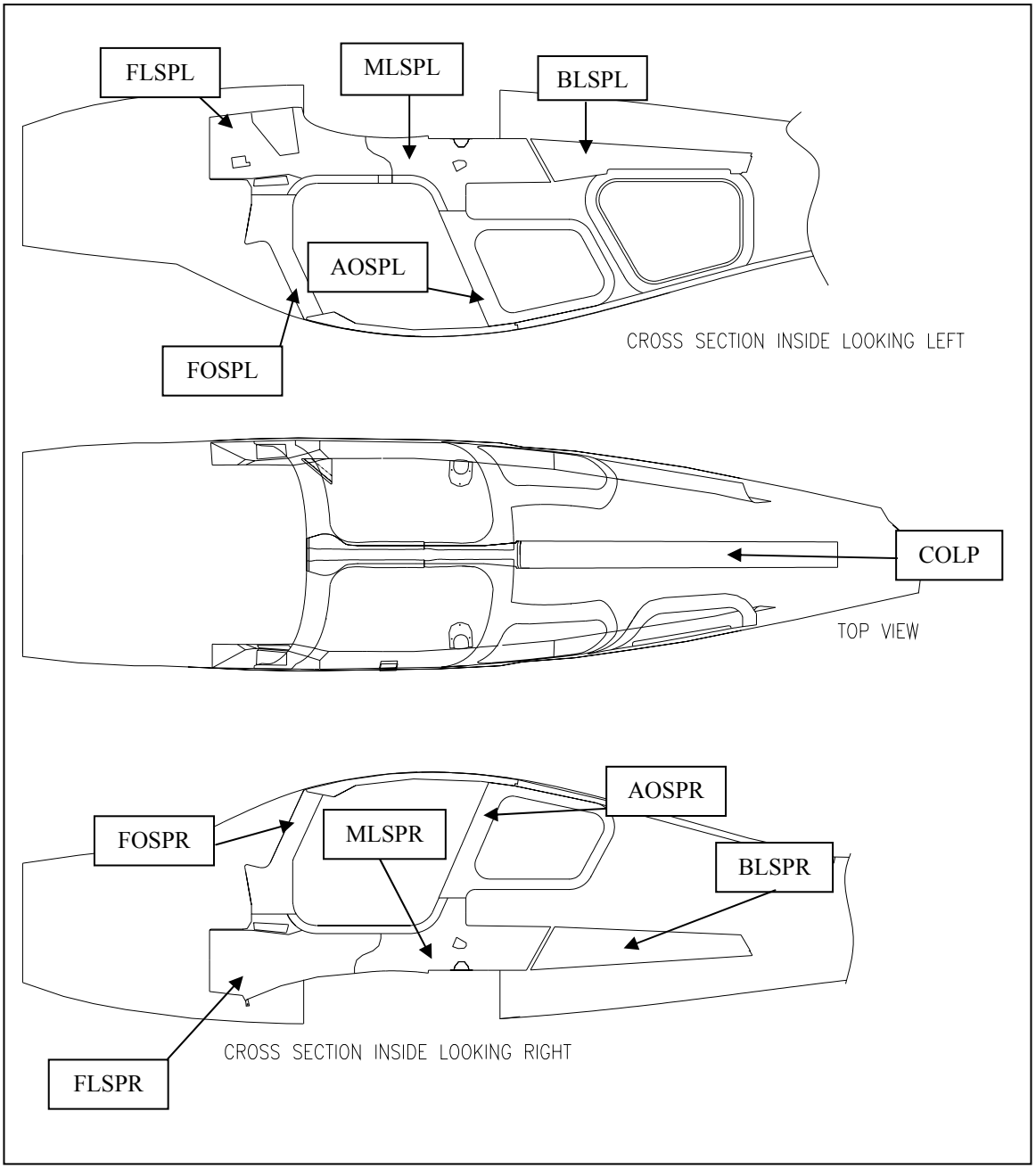


Figure 25 – 1 Interior Panel Locations

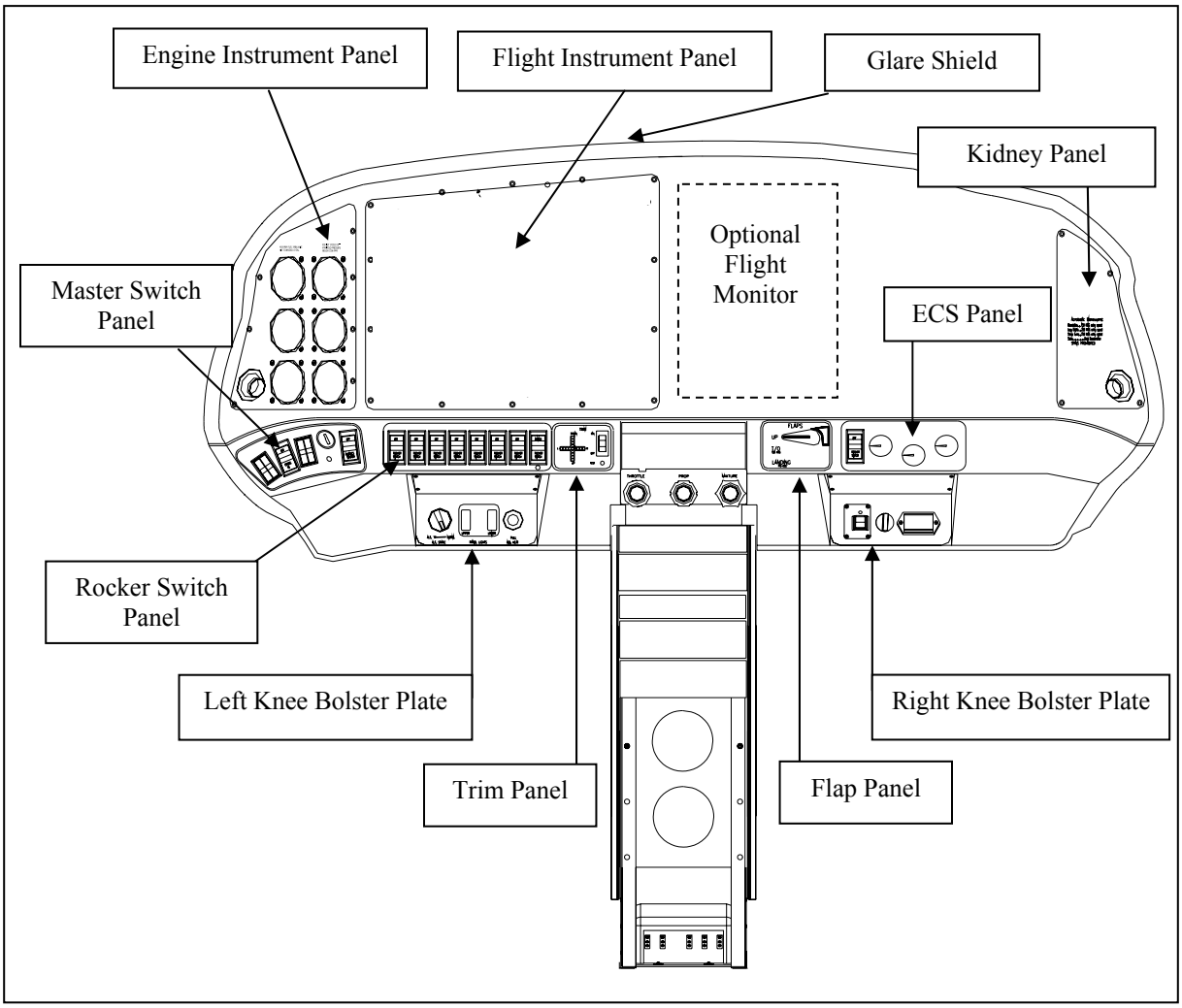


Figure 25 – 2 Panel Location (Basic Option)

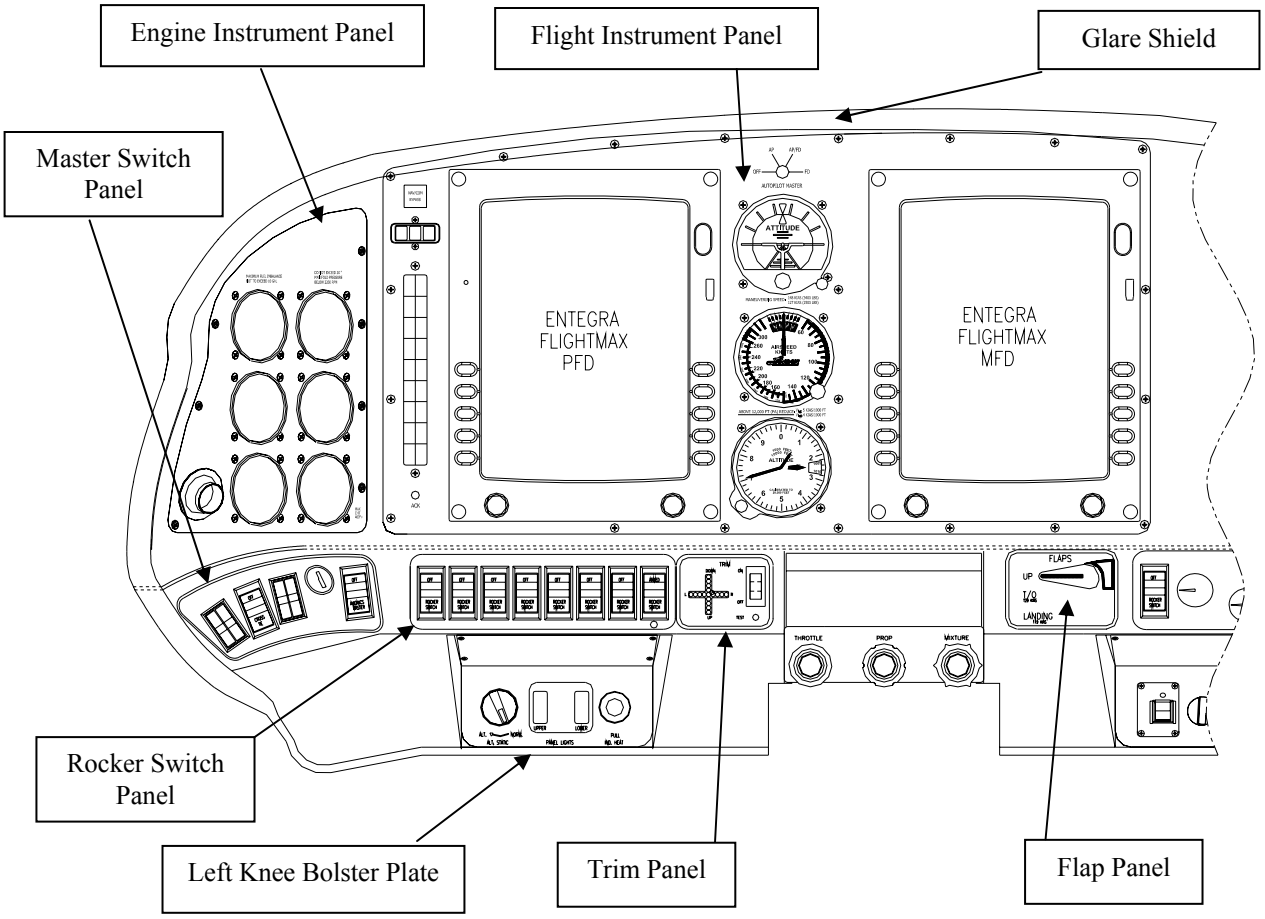
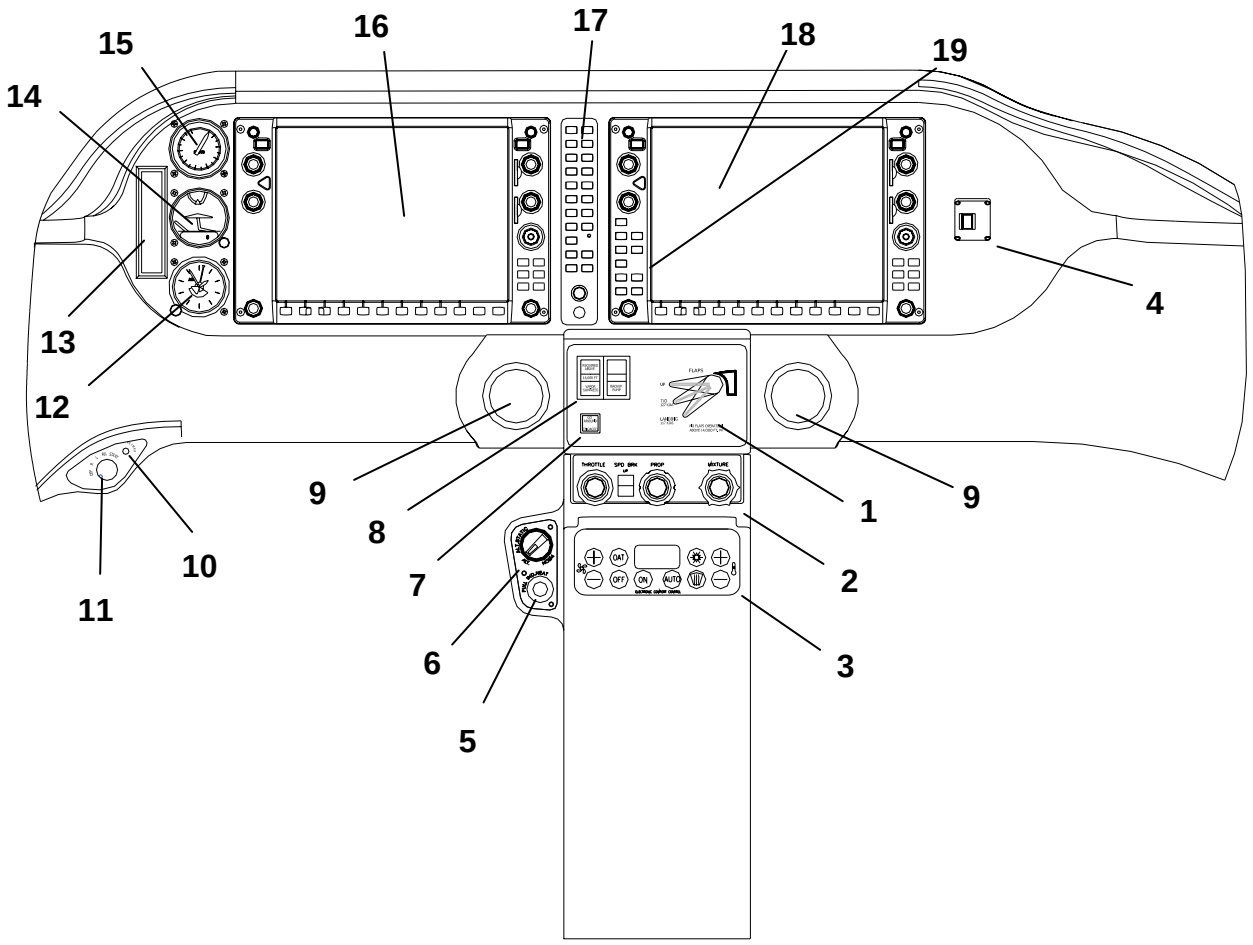


Figure 25 – 3 Panel Location (Avidyne Option)



Instrument Panel and Cockpit

1. Flap Panel – Flap Switch and Annunciator
2. Engine Controls
3. Environmental Control System (ECS) Panel or Automatic Climate Control System (ACCS) Panel
4. ELT Remote Switch
5. Heated Induction Air
6. Alternate Static Air
7. Go Around Switch
8. Rocker Switches: Backup Boost Pump and Vapor Suppression
9. Air Vents
10. Primer Switch
11. Ignition Switch
12. Altimeter
13. Pitot Heat, Door Seals, and Optional Switches
14. Attitude Indicator
15. Airspeed Indicator
16. Primary Flight Display (PFD)
17. Audio Panel
18. Multi-Function Display (MFD)
19. Autopilot Controls

Figure 25 – 4 Panel Location (Garmin G1000 Option)

25-2. GENERAL MAINTENANCE

a. Performing maintenance on the various components and systems in the airplane will frequently involve the removal of interior panels. Special tools are not needed. Most work can be performed with screw drivers, wrenches, sockets, torque wrenches, and adapters. When using an adapter on a torque wrench, follow the formula in Chapter 20-00-00 to avoid over-torquing.

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25-3. REMOVAL, INSTALLATION, AND ADJUSTMENT OF THE FRONT SEATS

a. **Removal** – The front seat cushions and seatbacks are interchangeable, i.e., the cushions or seatbacks are identical on the pilot's and copilot's side. The following procedure is used to remove the front seats. See Figure 25 – 5.

1. **Cushion** – The front seat bottom cushions are attached to the seat pan with Velcro and are removed by grasping the cushion near its forward base and pulling up on it.
2. **Seatback** – Removal of the front seatback is accomplished by removing the cotter pin and washer on each side of the seat pan. The clevis pin is captured inside the seat pan and cannot be removed, however, it can be rotated. Use the following steps for removal of the seatback:
   a) Move the seat to the forward position. This is necessary to access the outboard cotter pins.
   b) Remove the left and right cotter pins from their clevises.
   c) Simultaneously, apply an outward pressure (away from the seat pan) on each seatback attachment bracket until the hole in each bracket is free of the clevis.
   d) When each bracket's holes are free from their clevis, gently pull up on the seatback until both brackets are clear of the clevis pins.

NOTE

Once a cotter pin is removed, it must be discarded. Do not reuse a cotter pin on the seat mechanism or on any other part in the airplane.

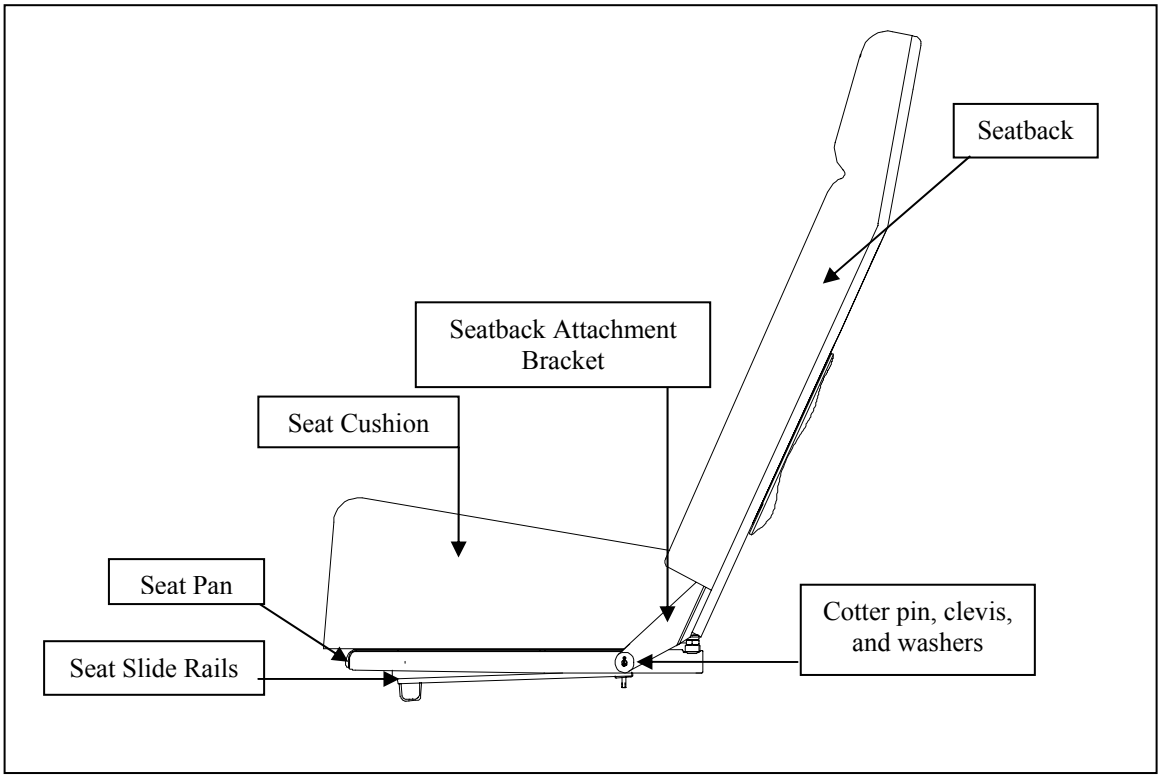

Figure 25 – 5 Front Seat

3. **Seat Pan** – The seat pan is removed using the following steps.
   a) Slide the seat fully back, and remove the two socket head screws holding the rails to the cabin floor.
   b) Slide the seat fully forward, and remove the two socket head screws by inserting an allen wrench through each of two small holes in the seat pan.
4. **Alternate Seatback Removal Technique** – It is also permissible to remove the seatback by first removing the four bolts on the seat pan and then removing the seat assembly from the airplane. When the seatback is attached to the seat pan, it is a bit more awkward to maneuver, and care must be take not to damage the interior. However, access to the outboard cotter pins is facilitated.
5. **Adjusting the Seatback** – The tilt of the seatbacks can be adjusted fore and aft, depending on the position of two course-threaded $^{3}/_{8}$ in. bolts located in the aft portion of the seat pan. The bolts are held in place by two jam nuts. To adjust the seat, use the following procedure:
   a) Remove the seat cushion, and move the seatback to the full forward position. **See Caution statement below.**
   b) Loosen both jam nuts enough to allow for the anticipated adjustment.
   c) Raise or lower both course-threaded $^{3}/_{8}$ in. bolts, depending on whether the seatback is to be adjusted fore or aft. Ensure that the bolts on each side are adjusted an equal number of turns. **See Warning statement below.**
   d) At this point the seat cushion can be installed, and the person for whom the adjustment is being made can test the position of the seatback.
   e) When the seatback is in the desired position, tighten the jam nuts against the base of the seat pan and torque 72 to 84 in.-lbs.

CAUTION

When the seat cushions are installed, they limit the forward travel of the seatback. However, with the cushion removed, the seatback will fold forward until its headrest touches the instrument panel or glare shield. When moving the seatback with the seat cushion removed, hold on to the seatback until it is touching the instrument panel. Do not move the seatback past its center of balance and then release it, because this can cause damage to the seat, glare shield, flight instruments, or flight instrument panel. In addition, when the seatback is resting on the instrument panel or glare shield, ensure no additional pressures are applied to the seatback.

WARNING

There is a limitation on the forward tilt of the seat. When adjusting the tilt of the seatback, at least 6½ threads of the course-threaded $^{3}/_{8}$ in. bolts must be inserted into the seat pan.

b. **Installation** – Installation of the front seats and their related parts is performed by reversing steps 1 and 4 of paragraph a above. During the installation process, remember the following points:
   1. The seatback attachment brackets must be temporarily flexed outward (widened slightly) by applying an outward pressure as they are slipped on to the clevis.

2. There are four washers for each seatback, two for each seatback attachment bracket. The washers should be positioned on each side of the bracket. To make the bracket sandwich, use the following steps.
   a) Insert one washer over each clevis.
   b) Install the seatback.
   c) Install the second washer over each clevis.
   d) Install the cotter pin.
   e) The bolt on the seat pan should be torqued 30 to 36 in.-lbs.

c. **Seat Slide Rails** – The slide rails, under normal conditions, should last for many years, and the likelihood of replacement is remote. Moreover, access to the removal hardware is difficult and requires significant disassembly. If problems do occur with the slide rails that necessitate removal, contact the manufacturer for specific instructions.

25-4. REMOVAL AND INSTALLATION OF THE REAR SEATS AND CROSSBAR

a. **Removal** – The rear seat cushions are not interchangeable because of the outboard cutout in each seat. Technically, the rear seatbacks are interchangeable; however, the holes for the quick-release pins may not match perfectly from left to right, and it is recommended that the location of each seatback be tagged before removal. To remove the seats, use the following steps. See Figure 25 – 6.

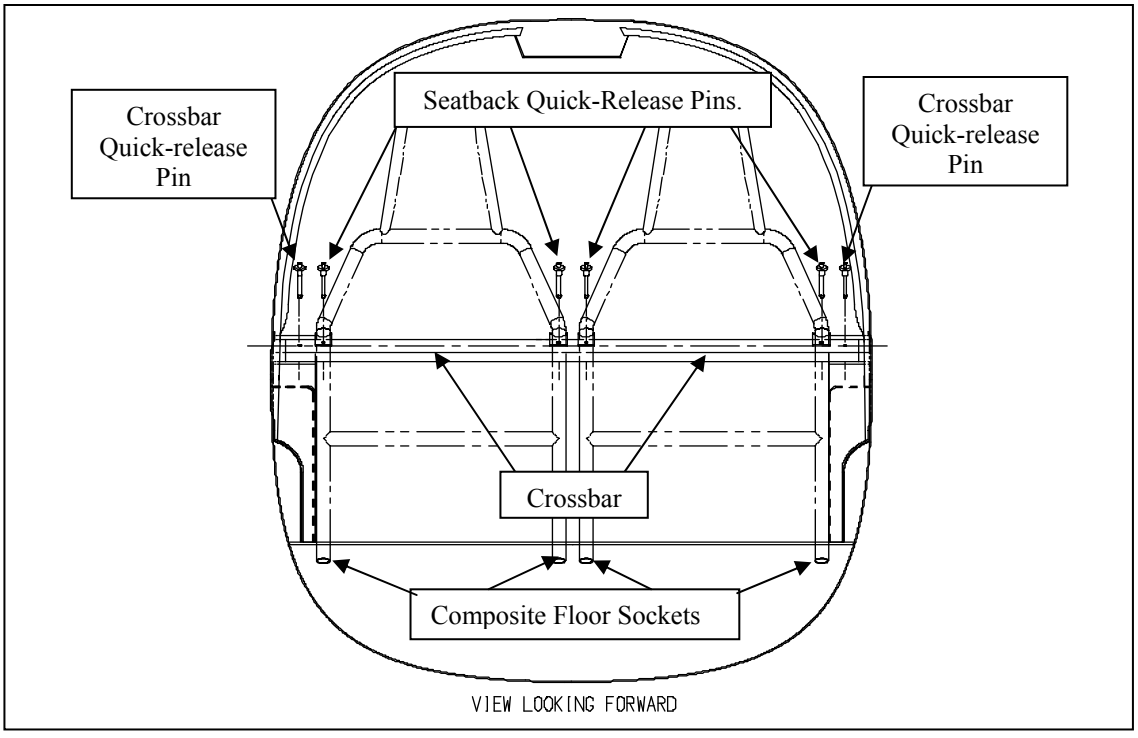

Figure 25 – 6 Rear Seats

1. The rear seat bottom cushions are attached to the composite seat bench with Velcro and are removed by grasping the cushion near its forward base and pulling up.
2. Remove the four quick-release pins from the seatbacks.
3. Pull the seat forward until the seatback pin bracket is clear of the crossbar, and lift the seatback out of the composite sockets in the floor.

4. To remove the crossbar, remove the two outboard quick-release pins. If the retention clip affixed to the pin bracket becomes dislodged, reinstall it with the smaller hole on the bottom side of the bracket, the open end of the clip facing outboard, and the body of the clip facing inboard.

b. **Installation** – Reverse steps 1 through 4 of paragraph a for reinstallation of the rear seat seatbacks, cushions, and crossbar.

25-5. INSTRUMENT PANELS AND ELECTRICAL PANELS

a. **Flight Items** – This section covers procedures for removal of the following (see Figure 25 – 2 and Figure 25 – 3):
   1. Glare shield and glare shield extension
   2. Engine instrument panel
   3. Flight instrument panel
   4. Left and right knee bolsters
   5. Kidney panel

b. **Electrical Panels** – In addition, removal of certain electrical panels are covered in this section. These include:
   1. Circuit breaker panel
   2. Master switch panel
   3. Rocker switch panel
   4. Trim panel
   5. ECS Panel
   6. Flap Panel

CAUTION

Use a non-magnetic screwdriver when working on or around the instrument or electrical panels including the glare shield.

25-6. GLARE SHIELD

a. **Basic or Avidyne Option**

   1. **Glare Shield Extension** – The removal of the glare shield extension is accomplished by pulling aft on the curved lip of the extension shield until it stops. Using both hands, simultaneously bend the outer edges down on both sides to clear the stops, and then pull free. Installation is the reverse of the above.
   2. **Glare Shield Removal and Installation** – Use the following steps to remove and install the glare shield.
      a) Remove the three flat head, 8-32 screws and washers from the underside of the panel.
      b) Tilt the glare shield up to gain access to the wiring for light. Disconnect this wiring.
      c) Slide the glare shield aft out of the clips bonded in the defrost channels at the base of the windshield.
      d) Installation is the reverse of the above. Tighten glare shield screws until snug.

b. **Garmin G1000 Option**

   1. **Glare Shield Removal**
      a) Remove the Garmin GDU 1040 PFD and GDU 1042/GDU 1044 MFD per Chapter 34.
      b) Remove the six washer head, 8-32 screws from the underside of the panel.
      c) Tilt the glare shield up to gain access to the wiring for light. Disconnect this wiring.
      d) Slide the glare shield aft out of the clips bonded in the defrost channels at the base of the windshield. Take care not to contact the windshield.

2. **Glare Shield Installation –** Installation is the reverse of removal. Tighten glare shield screws until snug.

25-7. INSTRUMENT PANEL

a. **Removal and Installation of Instrument Panel (Basic or Avidyne Option)** – Removal of the instrument panel is a major task and is not likely to ever be needed as access to areas behind the instrument panel can be gained from removing the panels within the instrument panel. The instructions for removal are only for removing the instrument panel when the same panel is to be replaced. If a new instrument panel is required, contact the manufacturer for further instructions. The instrument panel and radio rack should be removed and installed as a unit.

1. Remove or disconnect all instruments in the instrument panel.
2. Remove the center console, center overhead console, upper and lower forward side panels, and glare shield per instructions in Chapter 25. Disconnect the ventilation tubing to the eyeball vents and main heater control valve per instructions in Chapter 21.
3. Disconnect the control cables at the throttle quadrant per instructions in Chapter 76.

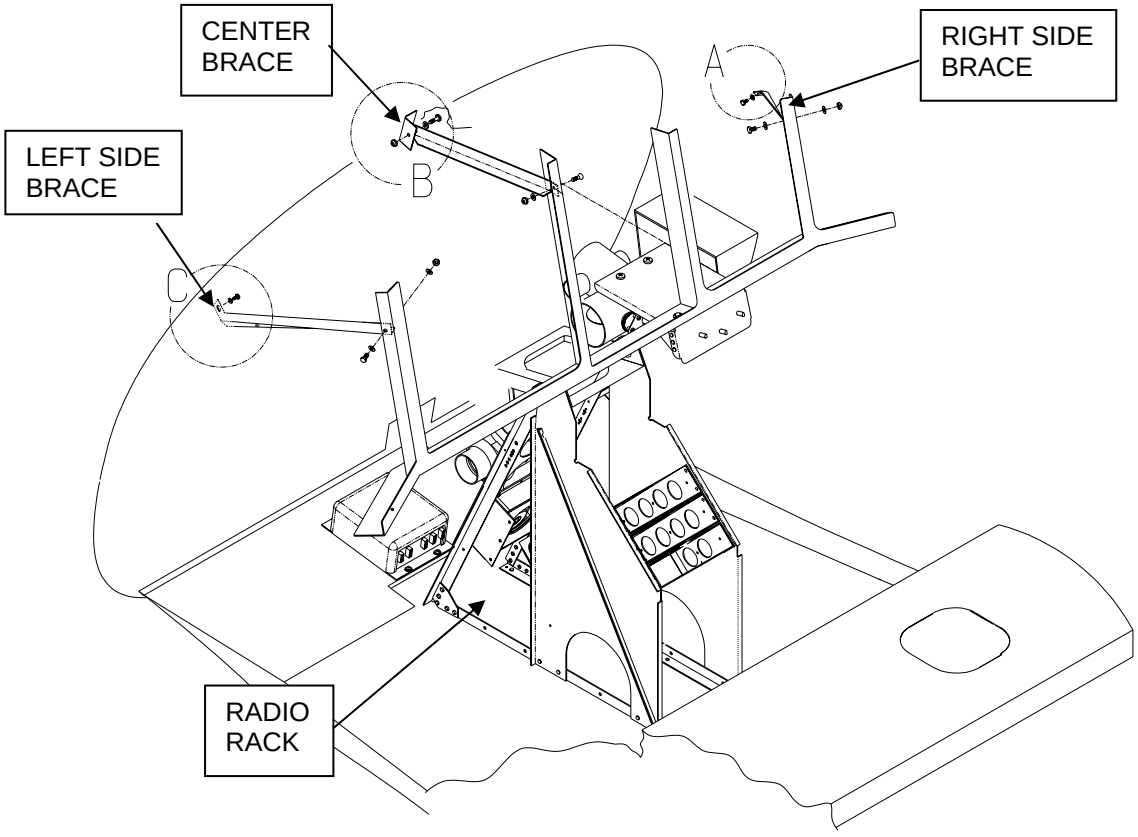

Figure 25 – 7 Instrument Panel Brace Locations

4. Remove the hex head bolt, washer, and self-locking nut that attaches the center brace at the top of the instrument panel to the firewall shown in Figure 25 – 7 and Figure 25 – 8.

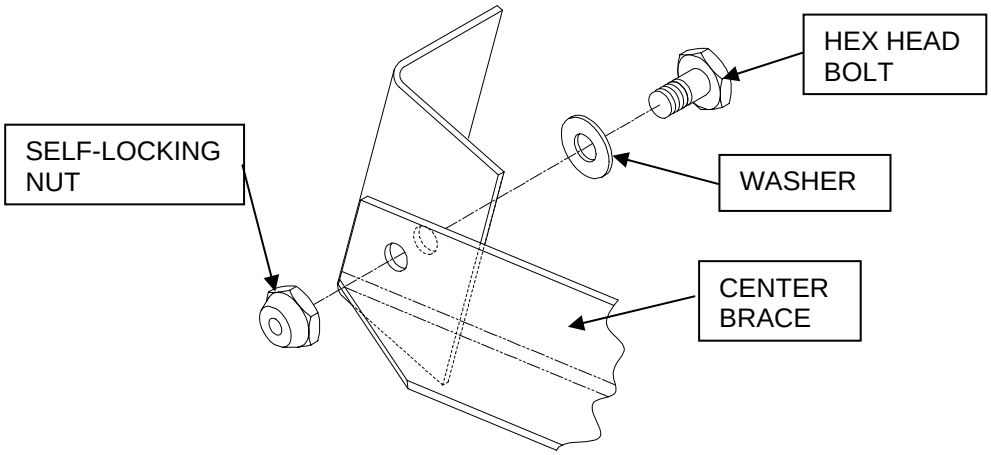

Figure 25 – 8 Instrument Panel Center Brace (Detail B)

5. Remove the 8-32 pan head bolt and washer from the threaded insert that attaches the left side brace of the instrument panel to the fuselage as shown in Figure 25 – 7 and Figure 25 – 9.

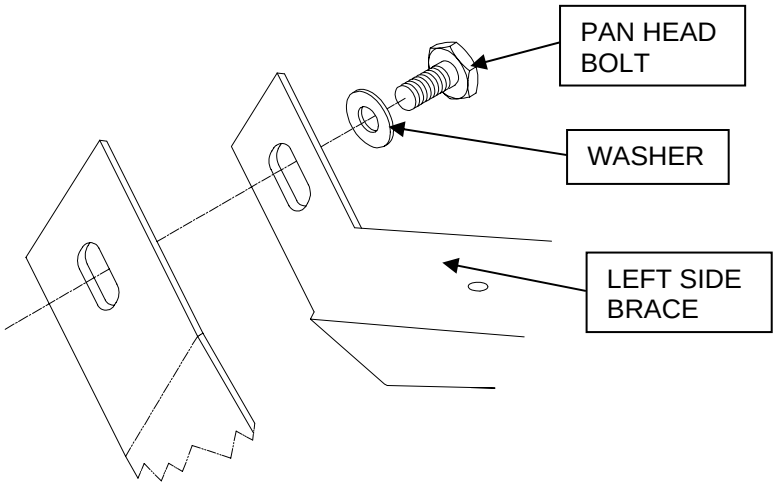

Figure 25 – 9 View of Left Side Brace (Detail C)

6. Remove the 8-32 pan head bolt and washer from the threaded insert that attaches the right side brace of the instrument panel to the fuselage as shown in Figure 25 – 7 and Figure 25 – 10.

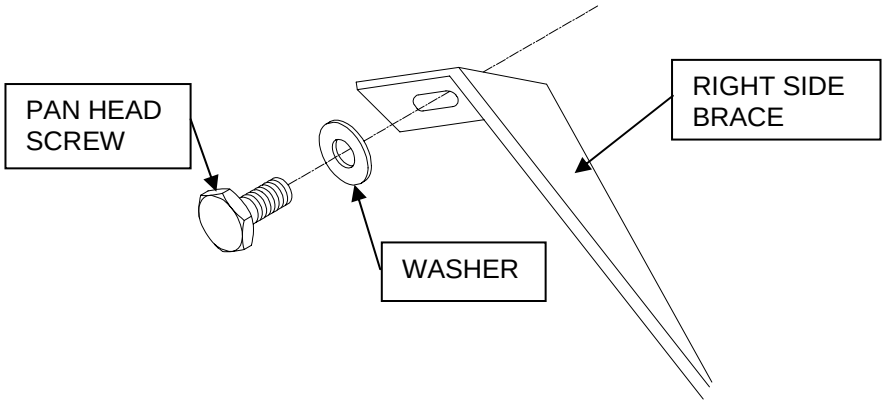

Figure 25 – 10 View of Right Side Brace (Detail A)

7. Remove the two 8-32 pan head screws, washers, and nuts attaching the instrument panel to the fuselage mount on both the left and right side. One screw on each side goes through an additional grounding strap and washer.

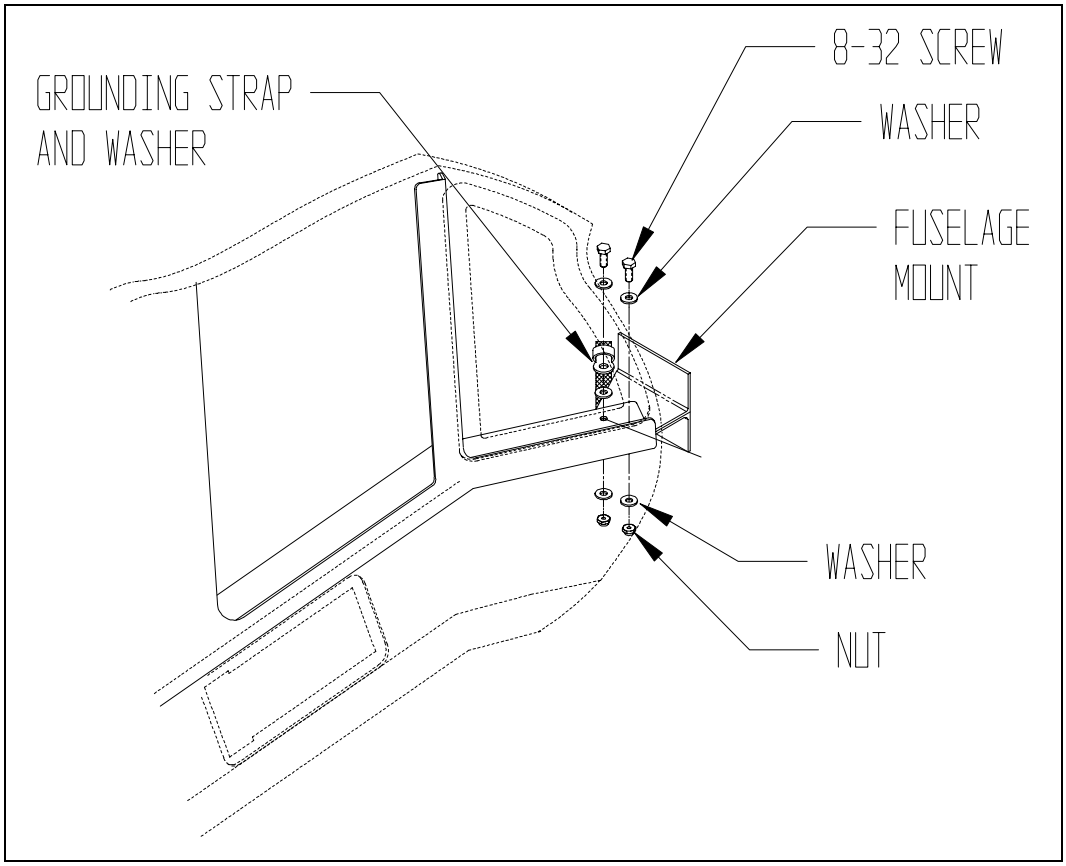

Figure 25 – 11 Instrument Panel Fuselage Mount

8. Remove the 10, 10-32 hex head bolts and washers attaching the radio rack to the floor, and remove the instrument panel and radio rack from the airplane.
9. Installation is a direct reversal of removal. Torque fasteners per section 20-3.

b. **Removal and installation of Instrument Panel (Garmin G1000 Option)** – Removal of the instrument panel is a major task and is not likely to ever be needed as access to areas behind the instrument panel can be gained by removing the Garmin GDU 1040 PFD, GDU 1042/GDU 1044 MFD, and the LRUs within the instrument panel. The instructions for removal are only for removing the instrument panel when the same panel is to be replaced. If a new instrument panel is required, contact the manufacturer for further instructions.
   1. Remove or disconnect all instruments in the instrument panel.
   2. Remove the glare shield, lower instrument panel closeout, left and right instrument panels, and upper and lower forward side panels per instructions in Chapter 25.
   3. Remove the flap panel per instructions in Chapter 25.
   4. Disconnect the ventilation tubing to the eyeball vents per instructions in Chapter 21.
   5. Disconnect the control cables at the throttle quadrant per instructions in Chapter 76.
   6. Remove the four screws at the top of the tower attaching it to the instrument panel. See Figure 25 – 12.
   7. Remove the two bolts, washers, and nuts attaching the instrument panel to the fuselage mount on both the left and right side. See Figure 25 – 12. One bolt on each side goes through an additional grounding strap and washer.
   8. Installation is a direct reversal of removal. Torque fasteners per section 20-3.

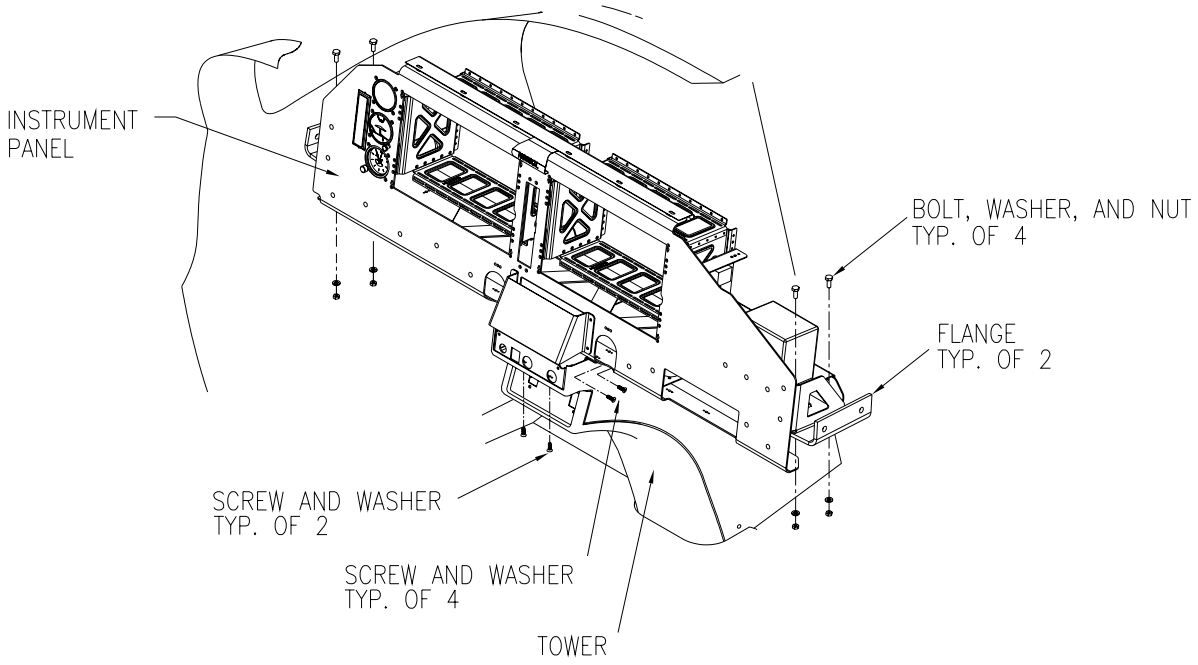

Figure 25 – 12 Instrument Panel Fuselage Mount

25-8. ENGINE INSTRUMENT PANEL (Basic or Avidyne Option)

a. **Removal and Installation** – The engine instrument panel assembly is secured to the instrument panel with six flat head 6-32 screws along the perimeter of the panel. Removal of these screws allows the panel to be tilted forward for access to the instruments. Refer to the appropriate chapter for removal and installation of a specific

instrument, control, switch, or light. When installing the engine instrument panel screws, torque 10 to 12 in.-lbs.

25-9. FLIGHT INSTRUMENT PANEL (Basic or Avidyne Option)

a. **Removal and Installation**

1. **Without Entegra Option–** The flight instrument panel without Entegra option is secured to the instrument panel with 14 flat head 6-32 screws along the perimeter of the panel. Removal of these screws allows the panel to be tilted forward for access to the instruments. Refer to the appropriate chapter for removal and installation of a specific instrument, control, switch, or light. When installing the flight instrument panel screws, tighten until the metal panel is tight against the fabric.
2. **With Entegra Option –** The flight instrument panel with the Entegra option is secured to the instrument panel with 19 flat head 6-32 screws along the perimeter of the panel and two on the interior. The Entegra Primary Flight Display and Multi-Function Display must be removed before removal of the flight instrument panel. Refer to the appropriate chapter for removal and installation of a specific instrument, control, switch, or light. When installing the flight instrument panel screws, tighten until the metal panel is tight against the fabric.

25-10. LEFT AND RIGHT KNEE BOLSTER (Basic or Avidyne Option)

a. **Removal and Installation** – The knee bolsters are attached to the instrument panel at the top of the bolster by four flat-head screws and to the side with either two self-locking nuts and washers or two nuts with integral toothed washer that attach to the studs. See Figure 25 – 13. Torque hardware per section 20-3.

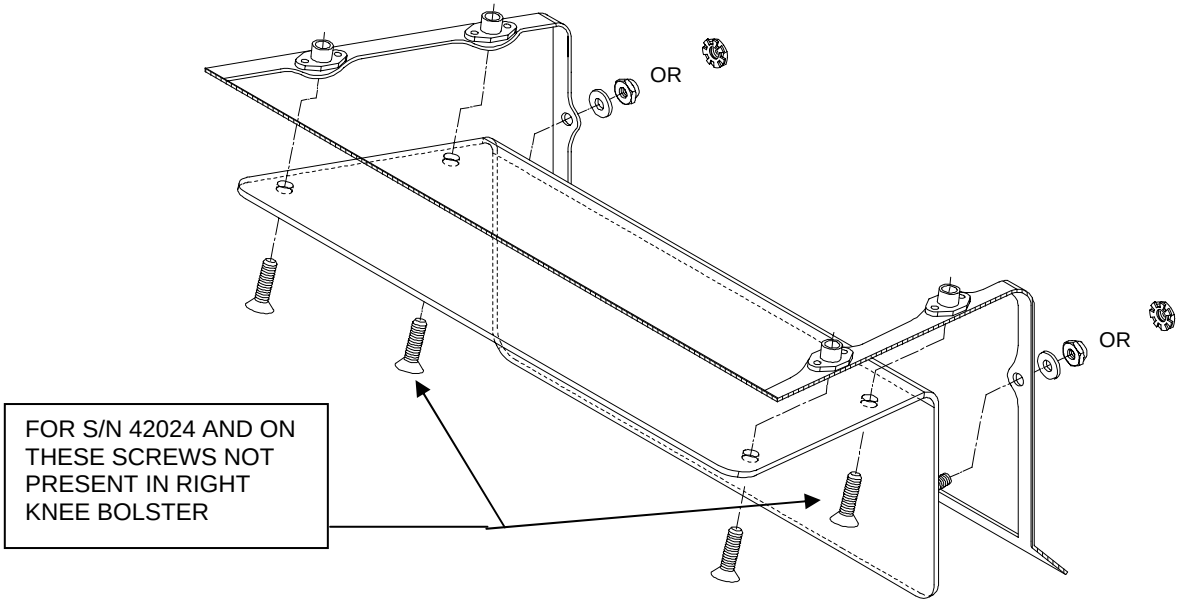

Figure 25 – 13 Knee Bolster to Instrument Panel Attachment

25-11. KIDNEY PANEL (Basic or Avidyne Option)

a. **Removal and Installation** – The kidney panel is secured to the instrument panel with four flat head 6-32 screws along the perimeter of the panel. Removal of these screws allows the panel to be tilted forward for access behind it. When installing the kidney panel screws, torque 10 to 12 in.-lbs.

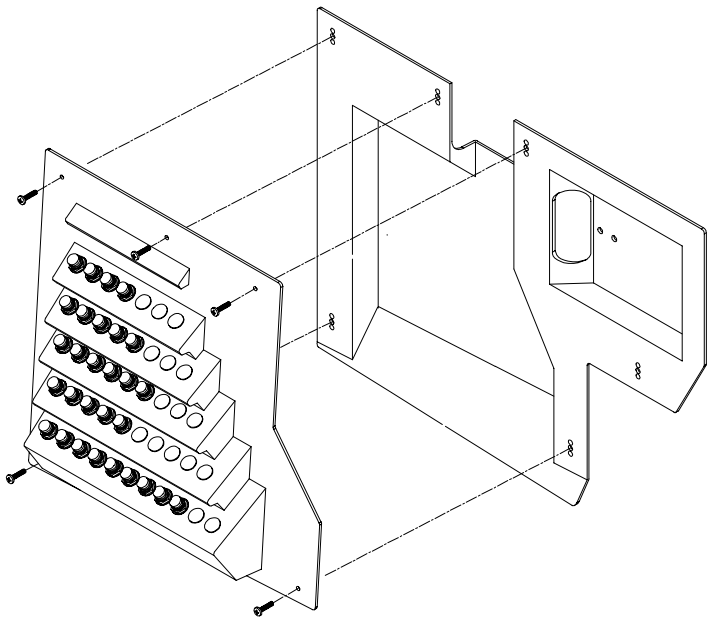

Figure 25 – 14 Circuit Breaker Panel

25-12. CIRCUIT BREAKER PANEL

a. **Removal and Installation** – The circuit breaker panel assembly is secured by five pan head 6-32 screws around the perimeter of the panel (see Figure 25 – 14). To gain access to these screws, remove the left forward lower side panel first (see paragraph 25-23). Removal of these screws allows the panel to be rotated out for access to the circuit breakers and wiring. Refer to the appropriate chapter for removal and installation of a specific circuit breaker, switch, or light. When installing the circuit breaker panel screws, torque 12 to 15 in.-lbs.

25-13. MASTER SWITCH PANEL

a. **Removal and Installation (Basic or Avidyne Option**)– The master switch panel faceplate has four threaded 6-32 studs that protrude through mating holes into the backside of the instrument panel (see Figure 25 – 15). Four self-locking 6-32 nuts with washers (or nuts with integral toothed washer), accessible from the backside of the instrument panel, screw onto these studs to secure the master switch panel to the instrument panel. Removal of the nuts with washers allows the master switch panel to slide out for access to the switches. Refer to the appropriate chapter for removal and installation of a specific switch or light. Removal of the left knee bolster (see paragraph

25-10) makes accessing the nuts securing the master switch panel easier. When installing the master switch panel nuts, torque 7 to 10 in.-lbs.

b. **Removal and Installation (Garmin G1000 Option)** – The master switch panel faceplate has four threaded 6-32 studs that protrude through mating holes into the backside of the FCOLP. Four self-locking 6-32 nuts with washers (or nuts with integral toothed washer), screw onto these studs to secure the master switch panel to the FCOLP. Remove the FCOLP per paragraph 25-26. Removal of the nuts with washers allows the master switch panel to slide out for access to the switches. Refer to the appropriate chapter for removal and installation of a specific switch or light.

25-14. ROCKER SWITCH PANEL (Basic or Avidyne Option)

a. **Removal and Installation** – The rocker switch panel faceplate has four threaded 6-32 studs that protrude through mating holes into the backside of the instrument panel (see Figure 25 – 15). Four self-locking 6-32 nuts with washers (or nuts with integral toothed washer), accessible from the backside of the instrument panel, screw onto these studs to secure the rocker switch panel to the instrument panel. Removal of the nuts with washers allows the rocker switch panel to slide out for access to the switches. Refer to the appropriate chapter for removal and installation of a specific switch or light. Removal of the left knee bolster (see paragraph 25-10) makes accessing the nuts securing the rocker switch panel easier. When installing the rocker switch panel nuts, torque 7 to 10 in.-lbs.

25-15. TRIM PANEL (Basic or Avidyne Option)

a. **Removal and Installation** – The trim panel faceplate has four threaded 6-32 studs that protrude through mating holes into the backside of the instrument panel (see Figure 25 – 15). Four self-locking 6-32 nuts with washers (or nuts with integral toothed washer), accessible from the backside of the instrument panel, screw onto these studs to secure the trim panel to the instrument panel. Removal of the nuts with washers allows the trim panel to slide out for access to the trim controls and indicator. Refer to the appropriate chapter for removal, service, and installation of the trim control and indicator. Removal of the left knee bolster (see paragraph 25-10) makes accessing the nuts securing the trim panel easier. When installing the trim panel nuts, torque 7 to 10 in.-lbs.

25-16. ECS PANEL

a. **Removal and Installation** – Refer to Chapter 21 for removal and installation instructions.

25-17. FLAP PANEL

a. **Removal and Installation** (**Basic or Avidyne Option)** – The flap panel faceplate has four threaded 6-32 studs that protrude through mating holes into the backside of the instrument panel (see Figure 25 – 15). Four self-locking 6-32 nuts with washers (or nuts with integral toothed washer), accessible from the backside of the instrument panel, screw onto these studs to secure the flap panel to the instrument panel. Removal of the nuts with washers allows the flap panel to slide out for access to the flap controls and indicator. Refer to the appropriate chapter for removal, service, and installation of the flap control and indicator. Removal of the right knee bolster (see paragraph 25-10) makes

accessing the nuts securing the flap panel easier. When installing the flap panel nuts, torque 7 to 10 in.-lbs.

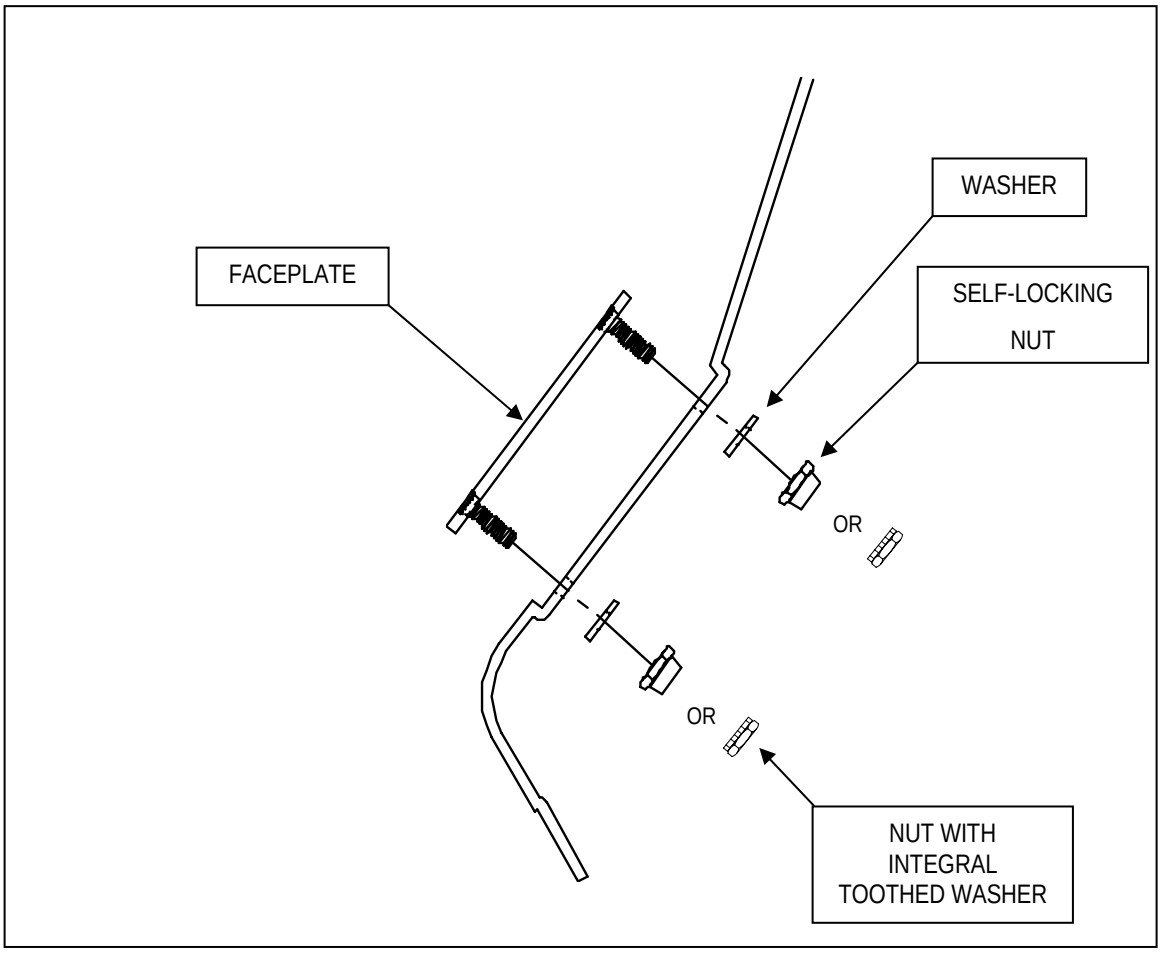

Figure 25 – 15 Panel and Faceplate

b. **Removal and Installation** (**Garmin G1000 Option)** – The flap panel faceplate has three 6-32 ball studs that fit into mating ball stud speed clips mounted on the instrument panel. To remove the flap panel firmly grasp the edges of the panel and pull aft. Removal of the flap panel allows access to the flap controls and indicator, backup pump switch, vapor suppression switch, and go around switch. Refer to the appropriate chapter for removal, service, and installation of the flap control and indicator, backup pump switch, vapor suppression switch, and go around switch. Installation of the flap panel is the reverse of removal.

25-18. ANNUNCIATOR PANEL (Basic or Avidyne Option)

a. **Removal and Installation –** The annunciator panel is secured by two flat head 6-32 screws. Two washers and self-locking nuts, accessible from the backside of the instrument panel, attach onto the screws. To remove the panel, disconnect the electrical connector from the backside of the panel, remove the screws, nuts, and washers, and slide out the panel. Installation is the reverse of removal. Tighten screws snug.

25-19. AURAL WARNING SYSTEM (Basic or Avidyne Option)

a. **Removal and Installation of the Acknowledge Switch or Acknowledge/Traffic Switch (with Ryan TCAD) –** The acknowledge switch is located directly below the annunciator panel.

1. Acknowledge Switch without Avidyne Primary Flight Display and Ryan TCAD
   a) Remove the glare shield per instructions in 25-6.
   b) Disconnect the two-pin plug at the back of the switch.
   c) Loosen the nut around the switch in front of the flight instrument panel, and remove the nut, washer, and placard. The nut on the rear of the flight instrument panel around the switch does not need to be removed in order to remove the switch.
   d) Remove the switch.
   e) Replace with the new switch, and reassemble in reverse order.
2. Acknowledge/Traffic Switch with Ryan TCAD
   a) Remove the Avidyne Primary Flight Display per instructions in Chapter 34.
   b) Disconnect the two plugs at the back of the switch.
   c) Pull the press-fit switch out of its housing at the back of the instrument panel.
   d) Remove the switch.
   e) Replace with the new switch, and reassemble in reverse order

b. **Removal and Installation of the Aural Warning Box (S/N 42001 and 42002)**

1. Remove the center console side panels per instructions in 25-23.
2. Remove three truss head screws securing the mounting plate to the radio rack. See Figure 25 – 16.
3. Remove the one truss head screw, self-locking nut, and washer that secures the clamp and mounting plate to the radio rack.
4. Push in the mounting plate, turn, and push out to gain access to the aural warning box.
5. Loosen the connector screws, and remove the connector.
6. Remove the four pan head screws, washers, and self-locking nuts securing the aural warning box to the mounting plate.
7. Replace the aural warning box, and reassemble in reverse order. Torque all screws 10 to 12 in.-lbs.

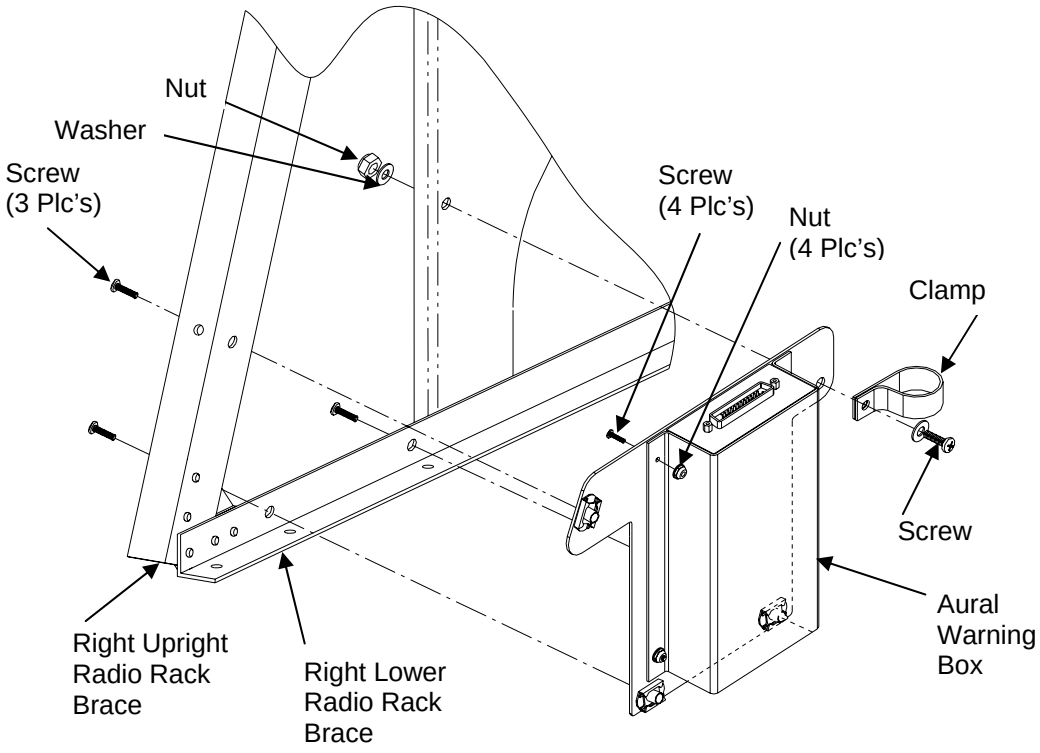

Figure 25 – 16 Aural Warning Box Installation (S/N 42001 and 42002)

c. **Removal and Installation of the Aural Warning Box (S/N 42003 and on)**
   1. Remove the center console side panels per instructions in 25-23.
   2. Remove the four 6-32 pan head screws, then remove the box.
   3. Replace the aural warning box, and reassemble in reverse order. Torque per section 20-3.

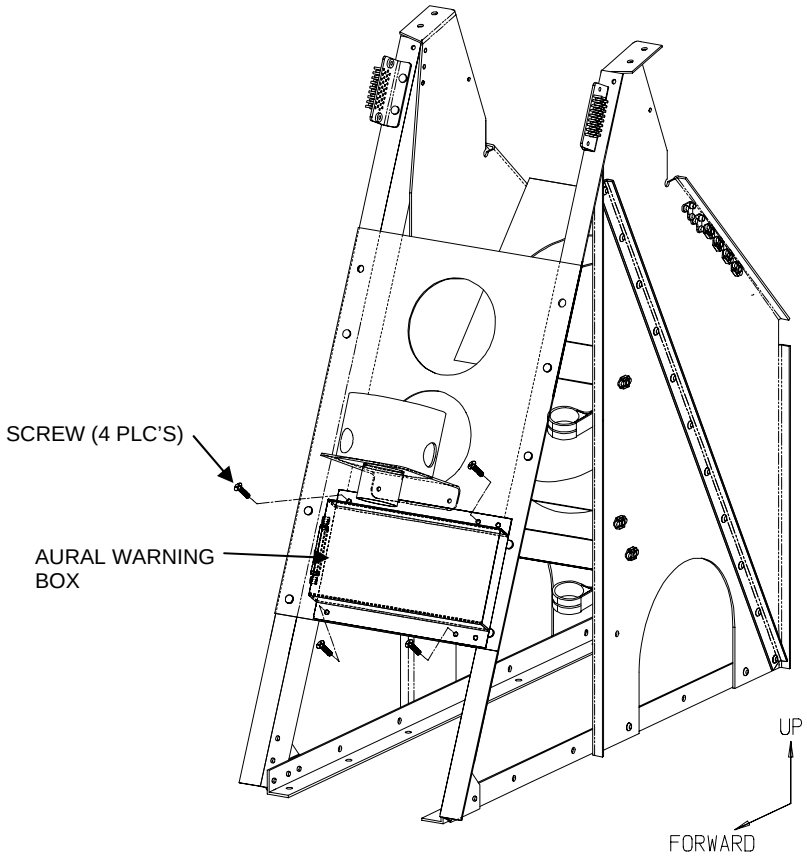

Figure 25 – 17 Aural Warning Installation (S/N 42003 and on)

d. **Aural Warning Troubleshooting**

1. No Audio Output
   a) Check that the audio panel and the avionics master switch is turned on.
   b) Check that the speaker is connected and is not defective.
   c) Check that the audio panel circuit breaker is not defective.
   d) Check that the connector on the remote aural warning box is not loose.
   e) Check that the ground for the aural warning box is being activated by the oil pressure switch on the engine.
   f) Check that the audio adjustment is turned up high enough on the aural warning box.
2. No Audio Output from One of the Six Warnings (Except "Door is Open")
   a) Check for a broken wire or a pushed-back pin.
   b) Temporarily ground the input pin for the inoperative channel, and test the aural warning box. You should hear the audio if the oil pressure switch is activated or jumpered.
3. No Audio Output from the "Door is Open" Warning
   a) If the tachometer is working, check that the wire going to the rear of the tachometer is not broken. Check that the connector does not have a pushed-back pin.

b) If the tachometer is not working, check the tachometer system per instructions in 77-2.
c) Check that the RPM adjustment is within tolerance.
d) Check for a broken wire or pushed back pin on the aural warning box.

4. Alarm Will Not Cancel
   a) Check that the acknowledge switch is connected.
   b) The acknowledge switch may be defective.
   c) Check for a broken wire or pushed back pin at the connector.

25-20. CARBON MONOXIDE DETECTOR (OPTIONAL)

The carbon monoxide detector is designed to detect, measure, and provide a visual and audible alert before the level of carbon monoxide (CO) reaches a critical level. The CO detector installation consists of the CO detector, a test/reset button (Basic or Avidyne option only), an annunciator panel light (Basic or Avidyne option only), and an interface to the aural warning system.

a. **Removal and Installation of the CO Detector Test/Reset Switch –** The test/reset switch is located on the left side of the engine instrument panel.
   1. Remove the engine instrument panel per instructions in paragraph 25-8.
   2. Disconnect the two-pin plug at the back of the switch.
   3. Loosen the nut around the switch in front of the flight instrument panel, and remove the nut, and washer. The nut on the rear of the flight instrument panel around the switch does not need to be removed in order to remove the switch.
   4. Remove the switch.
   5. Replace with the new switch, and reassemble in reverse order.

b. **Removal and Installation of the CO Detector –** The carbon monoxide detector is mounted to a bracket affixed to a gusset on the firewall behind the instrument panel. See Figure 25 – 18. On some Garmin G1000 aircraft, the CO detector is mounted on a shelf inside the tower.
   1. For the Basic or Avidyne option remove the engine instrument panel per section 25-9 or the Primary Flight Display per instructions in Chapter 34 depending upon which assembly is installed in the aircraft. For the Garmin G1000 option access the CO detector from under the instrument panel or remove the tower per section 25-21, as applicable.
   2. Disconnect the plug at the back of the detector.
   3. Remove the two screws and washers holding the detector to the mounting bracket.
   4. Remove the detector.
   5. Replace the detector, and reassemble in reverse order. Torque per section 20-3.
   6. For the Basic or Avidyne option reinstall the engine instrument panel per section 25-9 or the Primary Flight Display per instructions in Chapter 34.

c. **CO Detector Testing**
   1. **Basic or Avidyne Option –** When power is applied to the CO detector, a self-test routine begins. The test checks for functionality of the CO sensor, temperature sensor, and the integrity of the total CO detector system

WARNING

If the CO detect light in the annunciator panel flashes every four seconds,

there is a failure of the CO sensor, temperature sensor or the micro-controller. If the failure condition continues after resetting of the CO detector, the detector must be removed and returned to CO Guardian LLC for repair or replacement. No field repair or service is allowable other than to the wiring harness, test/reset switch , and circuit breaker.

a) Self-Test – The CO detect light in the annunciator panel will flash twice and then will remain off until there is another CO alert or until a failure of the unit occurs.
b) Test/Reset Switch – Operational check the CO detector by depressing the test/reset switch and verifying the CO detect annunciator panel light operation.

2. **Garmin G1000 Option –** See Garmin G1000 Cockpit Reference Guide.

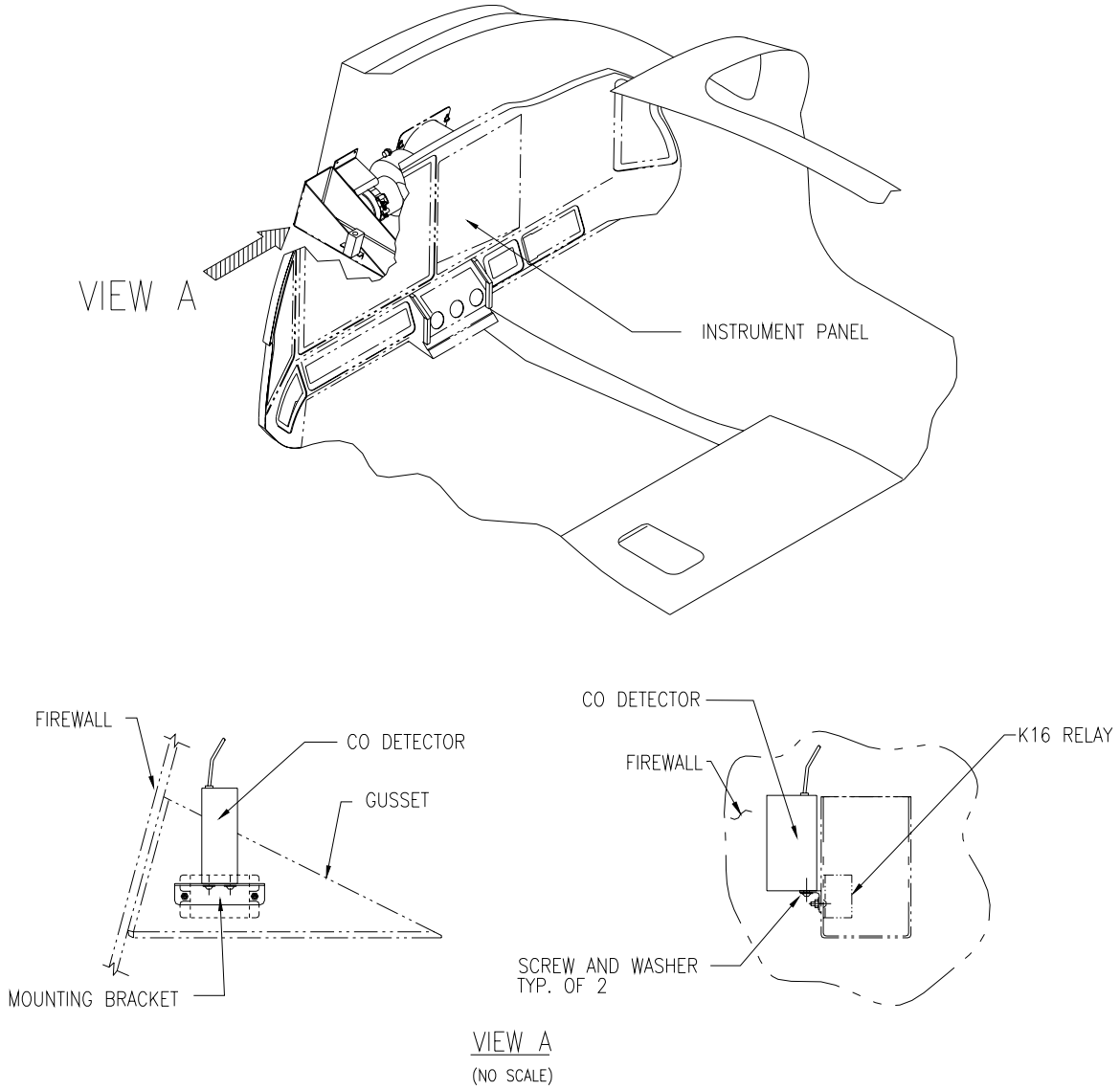

Figure 25 – 18 Carbon Monoxide Detector

25-21. TOWER

The tower is a component of the Garmin G1000 System interior and houses the Fuel Mixture Control Panel, Alternate Static control, and the ECS or ACCS Control Head.

a. **Removal and Installation**

1. Remove the eight 8-32 screws attaching the tower to the instrument panel and the cabin floor. See Figure 25 – 19.
2. Tilt and slide the tower out from underneath the instrument panel.
3. Disconnect the ECS or the ACCS Control Head, as applicable, per Chapter 21.
4. Disconnect any hoses.
5. Remove the tower.
6. Installation is the reverse of removal.

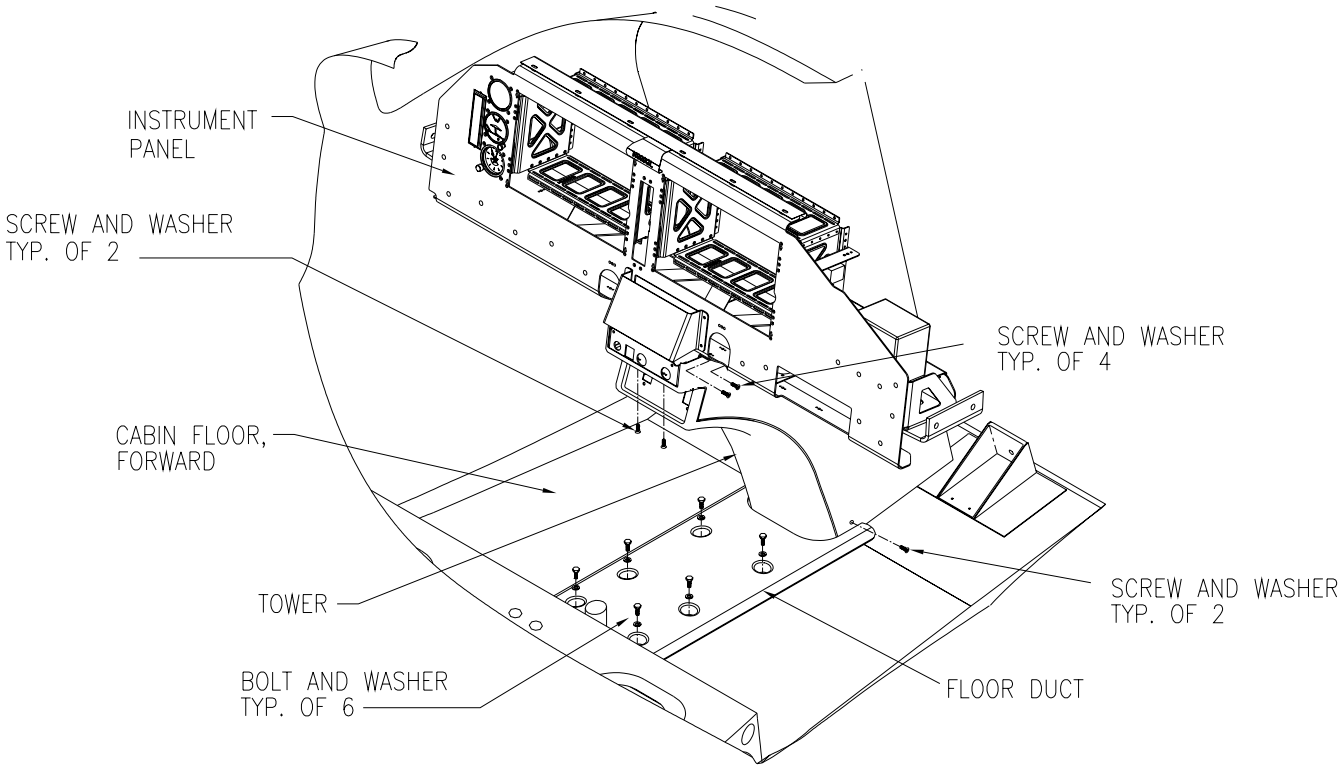

Figure 25 – 19 Tower and Floor Duct

25-22. FLOOR DUCT

The floor duct is a component of the Garmin G1000 System. It serves as both hot air duct and electrical wiring race.

a. **Removal and Installation**

1. Remove the center console per paragraph 25-23.
2. Remove the tower per paragraph 25-21.
3. Remove the six bolts and washers attaching the duct to the cabin floor See Figure 25 – 19.
4. Remove the floor duct.
5. Installation is the reverse of removal.

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25-23. CENTER CONSOLE

a. **Basic or Avidyne Option**

1. **Removal and Installation** – The center console does not normally have to be removed. An access panel on the right side below the fuel selector is provided for routine inspection and maintenance. In addition, the left and right panels in the footwells may be removed independently of the rest of the console. See Figure 25 – 20.

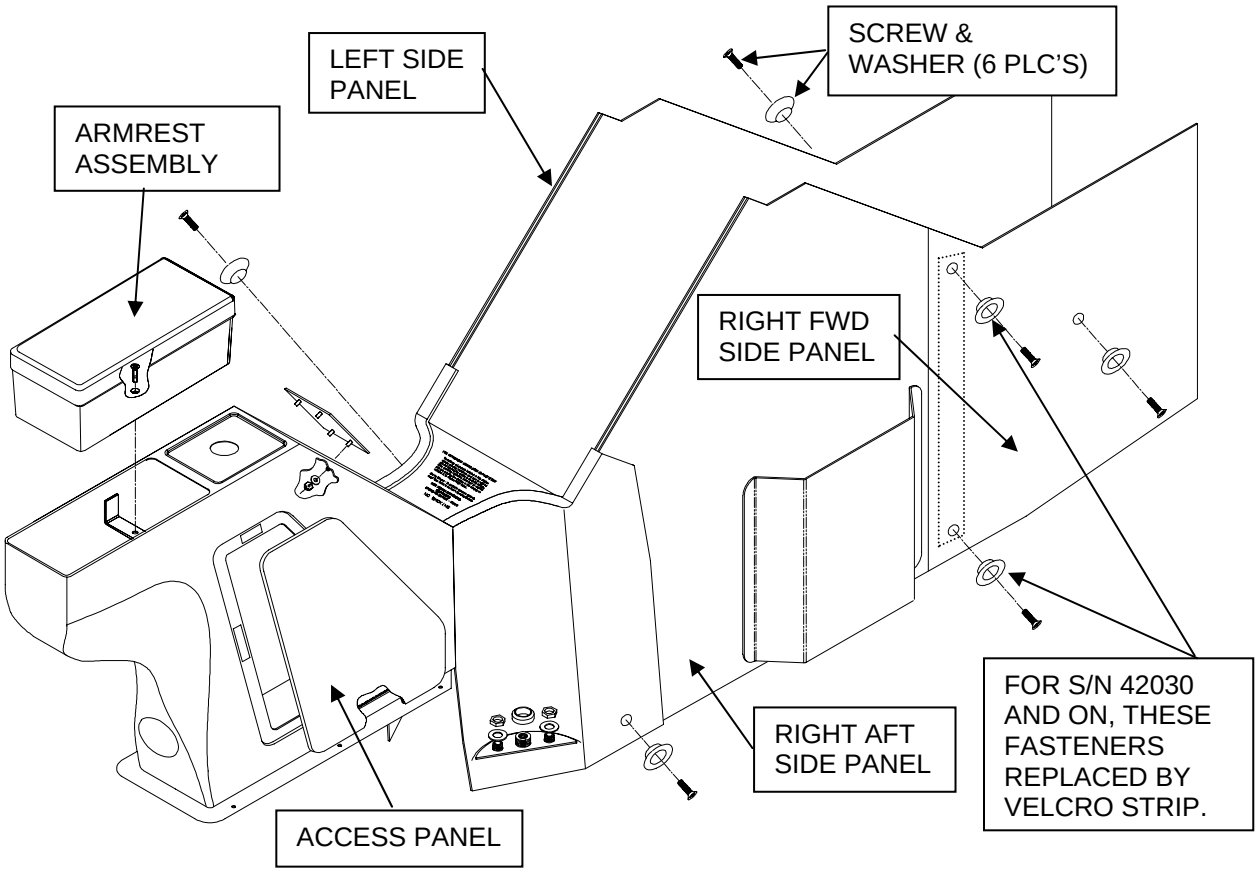


Figure 25 – 20 Center Console

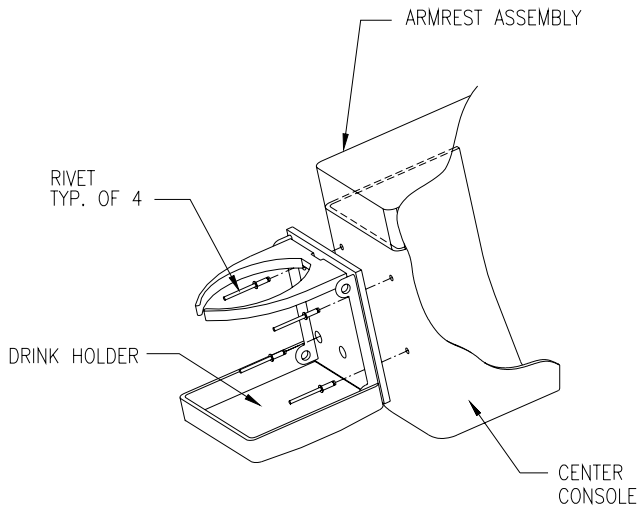


Figure 25 – 21 Drink Holder

a) **Armrest assembly** – The armrest assembly, including pencil box, is attached to the center console with one flat head screw. This screw is visible after the cushion in the bottom of the pencil box is removed. An optional stereo audio input jack may be installed within the pencil box and must be disconnected before removal of the armrest or the console. The armrest assembly does not have to be removed to remove the center console.

b) **Drink holder** – For S/N 42004 through 42006, 42008, and 42010 and on, a drink holder is installed in the aircraft. The drink holder is located on the aft end of the center console, flush with the top of the console and below the armrest. It is attached with four AAP43 rivets. The rivets may be accessed once the armrest assembly has been removed. The drink holder does not have to be removed to remove the center console. Should the drink holder need to be replaced, carefully drill out the four attachment rivets to remove the drink holder. Obtain a new drink holder, part number 588220-1, from the manufacturer and reinstall with four new rivets of the same type and size. Take care when drilling out rivets so as not to enlarge holes. See Figure 25 – 21

c) **Left side panel** – The left side panel is attached to the center console by two flat-head screws and washer. The panel is attached to the floor with a flat head screw and washer through a bracket and j-nut near the firewall. Removing the screws allows the panel to be removed. Removal of the left knee bolster may be necessary for added clearance to remove the left side panel on the center console.

d) **Right forward and aft side panel** – The right forward and aft side panels are attached to the center console by four flat-head screws and washers. The panels are attached to the floor with a flat-head screw and washer through a bracket and j-nut near the firewall. Removing the screws allows the panels to be removed. Removal of the right knee bolster may be necessary for added clearance to remove the right side panel on the center console.

e) **Access panel** – The access panel is secured with Velcro and can be removed by lifting at one corner and continuing to lift around the panel. Removal of the right seat cushion (see paragraph 25-2.a) allows better visibility of the access panel and the inside of the center console.

f) **Heater ventilation tubing** – The heater ventilation can be removed by clipping the tie wraps that hold the tubing to the eyeball vent and sliding the tubing off the eyeball vent. Since the tie wrap is cut (destroyed) during removal, a new tie wrap must be used when reassembling the heater ventilation tubing.

g) **Fuel tank selector** –See Chapter 28.
h) **Fuel selector knob** – See Chapter 28.
i) **Console** – After completing steps I through (h) above, use the following steps to remove the console (see Figure 25 – 20):
   1) Remove the three flat head 10-32 screws located at the aft of the center console under the carpet on each side.
   2) Disconnect the headphone/intercom jacks and microphone plugs and optional stereo audio input jack (if present). Refer to Chapter 23.
   3) Take precaution to ensure that the splined rod connecting the fuel tank selector to the fuel selector knob is protected from damage. This is especially important if the center console is to be removed without removing the fuel tank selector.

2. **Assembly and installation** – To assemble and install the center console, reverse the above procedure.

b. **Garmin G1000 Option**

1. **Removal and Installation** – The center console does not normally have to be removed. An access panel on the right side below the fuel selector is provided for routine inspection and maintenance. See Figure 25 – 22.

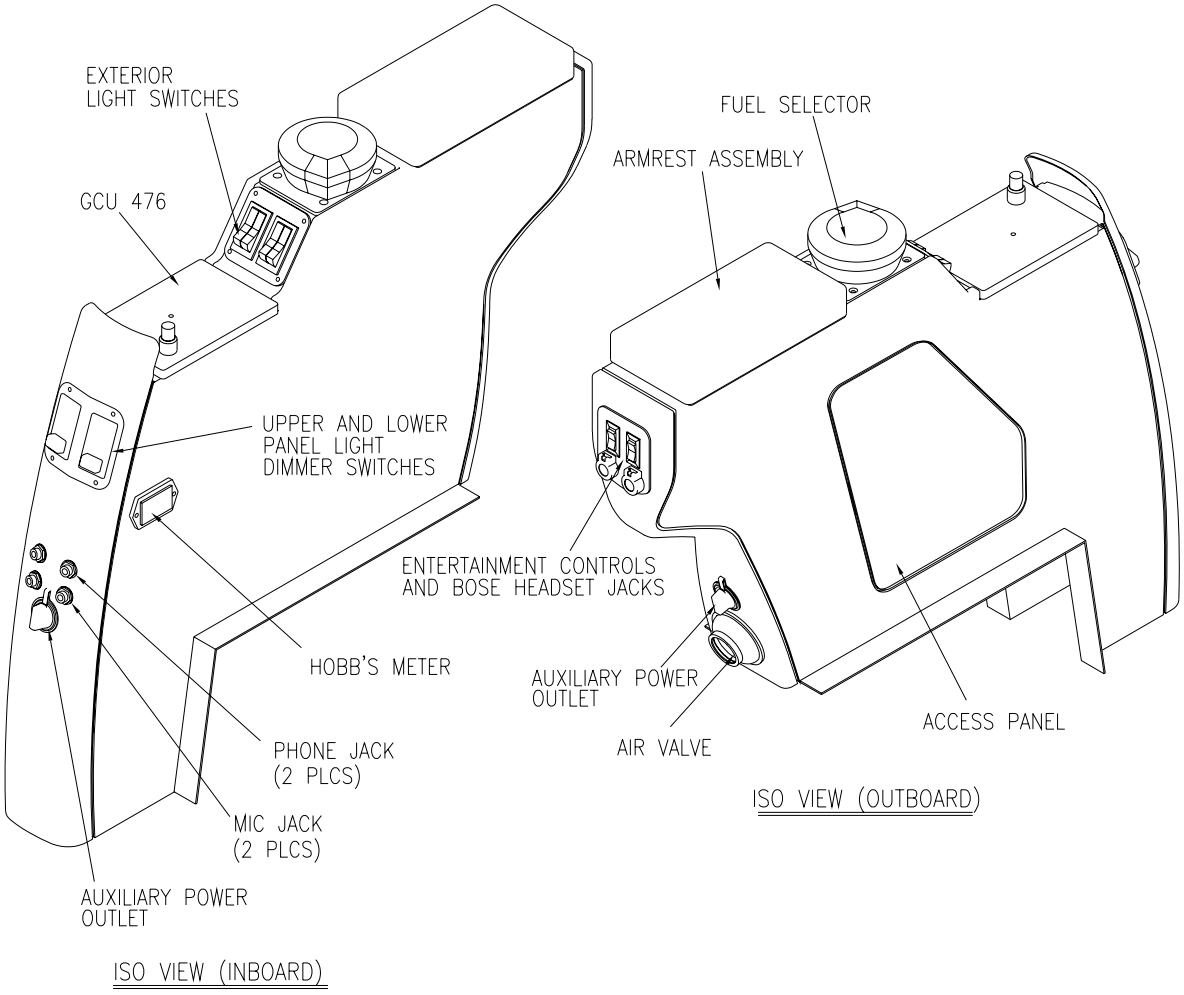


Figure 25 – 22 Center Console (Garmin G1000 Option)

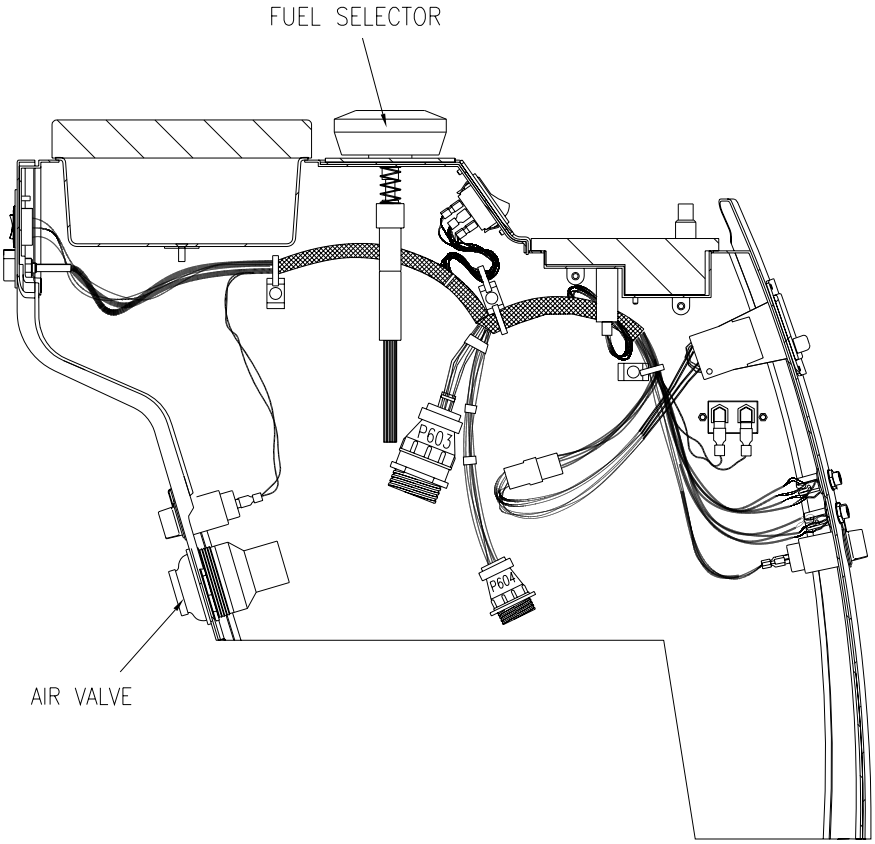


Figure 25 – 23 Center Console Wiring (Garmin G1000 Option)

a) **Armrest assembly** – The armrest assembly, including pencil box, is attached to the center console with one screw. The armrest assembly does not have to be removed to remove the center console.
b) **Access panel** – The access panel is secured with Velcro and can be removed by lifting at one corner and continuing to lift around the panel. Removal of the right seat cushion (see paragraph 25-2.a) allows better visibility of the access panel and the inside of the center console.
c) **Heater ventilation tubing** – The heater ventilation can be removed by clipping the tie wraps that hold the tubing to the eyeball vent and sliding the tubing off the eyeball vent. Since the tie wrap is cut (destroyed) during removal, a new tie wrap must be used when reassembling the heater ventilation tubing.
d) **Fuel tank selector** –See Chapter 28.
e) **Fuel selector knob** – See Chapter 28.
f) **Console Wiring** – Disconnect the wiring harness connectors P603 and P604. See Figure 25 – 23.
g) **Console** – After completing steps (b) through (f) above, use the following steps to remove the console (see Figure 25 – 22):
    1) Remove the three flat head 10-32 screws and countersunk finishing washers located at the aft of the center console under the carpet on each side.

2) Take precaution to ensure that the splined rod connecting the fuel tank selector to the fuel selector knob is protected from damage. This is especially important if the center console is to be removed without removing the fuel tank selector.

2. **Assembly and installation** – To assemble and install the center console, reverse the above procedure. Torque the six console to floor installation screws to 31 in-lbs.

25-24. SEAT BELTS AND RESTRAINTS

CAUTION

The seat belts are not interchangeable front to rear or left to right. Properly identify seat belts and shoulder harnesses when removing for correct installation later.

CAUTION

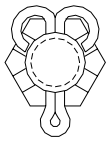
Do not remove the bolts that hold the seat belt bracket (located at the front side panel) to the floor, as these bolts are also used for wing attachment.

NOTE

Once a cotter pin is removed, it must be discarded. Do not reuse a cotter pin on the seat belt retaining bolts or on any other part in the airplane.

a. **Removal of Front Seat Belt and Shoulder Harness**
   1. Detach the lower seat belt mounts, buckle, and lap belt that are attached with $^{5}/_{16}$-24 bolts, castellated nuts, cotter pins, and cap plug (on the right side of the center bracket). Removal of the front seat bottom cushions will make accessing these mounts easier (see paragraph 25-3.a.1).
   2. Remove the front shoulder belt guide by unsnapping the guide cover, and remove the $^{1}/_{4}$-28 bolt and washer.
   3. After removal of the middle lower side panel (see paragraph 25-33), detach the shoulder belt retractors that are located behind the lower side panels just aft of the front seats. The retractors are attached with four screws, washers, and 10-32 nuts.

b. **Installation of Front Seat Belt and Shoulder Harness** – Installation is the reverse of the above with the following additions:
   1. Torque the 10-32 screws attaching the shoulder belt retractors 30 to 36 in.-lbs.
   2. Torque the $^{1}/_{4}$-28 bolts securing the shoulder belt guides to 72 to 84 in.-lbs.
   3. When securing the lower seat belt mount located at aircraft centerline, assemble the $^{5}/_{16}$-24 bolts and castellated nuts hand tight until all side play is eliminated. If the notch in the castellated nut does not align with the hole in the $^{5}/_{16}$-24 bolt, preventing the cotter pin from being installed, tighten the castellated nut until the next available notch aligns with the hole in the $^{5}/_{16}$-24 bolt, then install a new cotter pin. Fill a new MS-3 cap plug half full with Wacker SWS-951 sealant, and install over the nut with the cotter pin.
   4. When securing the lower seat belt mount located at the front side panel, assemble the $^{5}/_{16}$-24 bolts and castellated nuts hand tight until all side play is eliminated. If the notch in the castellated nut does not align with the hole in the $^{5}/_{16}$-24 bolt, preventing the cotter pin from being installed, back off the castellated nut until the next available notch aligns with the hole in the $^{5}/_{16}$-24 bolt. Install a new cotter pin as shown in the following illustration.



c. **Removal of Rear Seat Belt and Shoulder Harness**

1. Detach the lower seat belt mounts, buckle, and lap belt that are attached with $^{5}/_{16}$-24 bolts, castellated nuts, and cotter pins. Removal of the rear seat bottom cushions will make accessing these mounts easier (see paragraph 25-4.a).
2. The rear seat shoulder belt retractor is mounted on a stud protruding from the fuselage and held in place with a $^{3}/_{8}$-24 self-locking nut and two special washers. See Figure 25 – 24. The stud has a recess for a $^{5}/_{32}$ Allen wrench so that the stud can be prevented from rotating when removing or installing the self-locking nut.

CAUTION

The stud must be prevented from rotating. Failure to prevent rotation may result in cracked or chipped paint on the exterior of the fuselage at the rear shoulder belt retractor location.

d. **Installation of Rear Seat Belt and Shoulder Harness** – Installation is the reverse of the above with the following additions:

1. Install the special washers as shown in Figure 25 – 24.
2. Torque the self-locking nut 140 to 180 in.-lbs. **See Caution statement above.**
3. When securing the lower seat belt mounts, assemble the $^{5}/_{16}$-24 bolts and castellated nuts hand tight until all side play is eliminated. If the notch in the castellated nut does not align with the hole in the $^{5}/_{16}$-24 bolt, preventing the cotter pin from being installed, tighten the castellated nut until the next available notch aligns with the hole in the $^{5}/_{16}$-24 bolt, then install a new cotter pin.

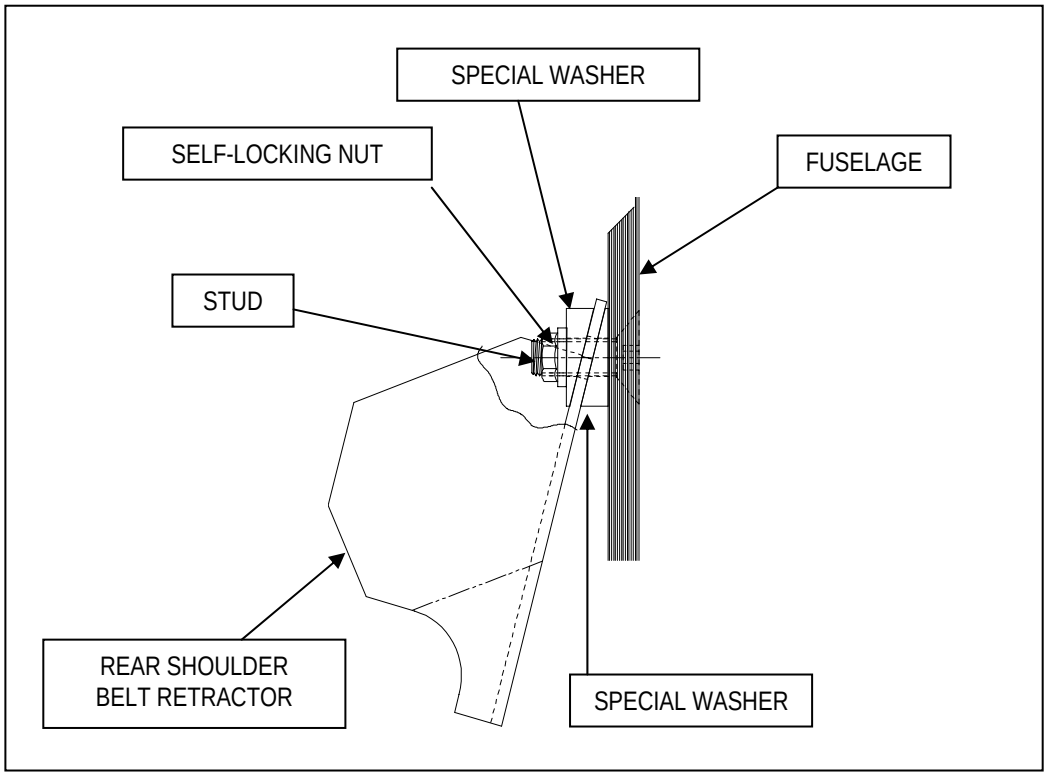


Figure 25 – 24 Rear Seat Belt Attachment

25-25. DOORS AND SIDE PANELS

a. The Cabin Doors and Baggage Door removal and installation are covered in Chapter 52.

b. **Cabin Door Armrest** – The door armrest is attached to the door with three 8-32 screws and washers accessible through the bottom of the door. Torque these nuts 20 to 25 in.-lbs. when reinstalling.

c. **Baggage Door Surround** – Removal of the baggage door surround panel is covered in Chapter 52.

d. **Baggage Door Interior Panel** – Removal and installation of the baggage door interior panel is covered in Chapter 52.

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25-26. FORWARD CENTER OVERHEAD LIGHT PANEL (FCOLP)

If built-in oxygen system is installed in the aircraft the cabin distribution manifold and the low-pressure oxygen manual shutoff valve are attached to the FCOLP.

a. **Removal of FCOLP**

1. Remove the four (three for S/N 42001 to 42003) overhead swivel lights and courtesy light per instructions in Chapter 33.
2. While supporting the forward center overhead light panel, remove the flat head 6-32 screw behind each overhead swivel light and the two flat head screws behind the courtesy light. See Figure 25 – 25, Figure 25 – 26, Figure 25 – 27, Figure 25 – 28, or Figure 25 – 29.
3. Carefully lower the forward center overhead light panel, and disconnect the wiring and the flexible oxygen hose (if built-in oxygen system is present). Immediately plug or cap both ends of the flexible oxygen hose quick disconnect fitting. Do not kink, stretch or damage the flexible oxygen hose. See Chapter 35 for specific procedures.

CAUTION

Do not disconnect the flexible oxygen hose connecting the cabin distribution manifold and the low-pressure oxygen manual shutoff valve to the quick disconnect fitting without first consulting the procedures and requirements in Chapter 35.

b. **Installation of FCOLP** – Installation is the reverse of the above. Torque the screws 30 to 36 in.-lbs.

25-27. AFT CENTER OVERHEAD LIGHT PANEL (ACOLP)

a. **Removal of ACOLP**

1. Remove the forward center overhead light panel. See paragraph 25-26 above.
2. Remove the hat rack cushion.
3. Remove the courtesy light per instructions in Chapter 33.
4. While supporting the aft center overhead light panel, remove the two pan head 10-32 screws at the forward bracket and the pan head 10-32 screw behind the courtesy light.
5. Slide the panel forward off the clip located on the rear bulkhead behind the hat rack cushion. See Figure 25 – 25, Figure 25 – 26, Figure 25 – 27, Figure 25 – 28, or Figure 25 – 29.

b. **Installation of ACOLP** – Installation is the reverse of the above. Torque the screws 30 to 36 in.-lbs.

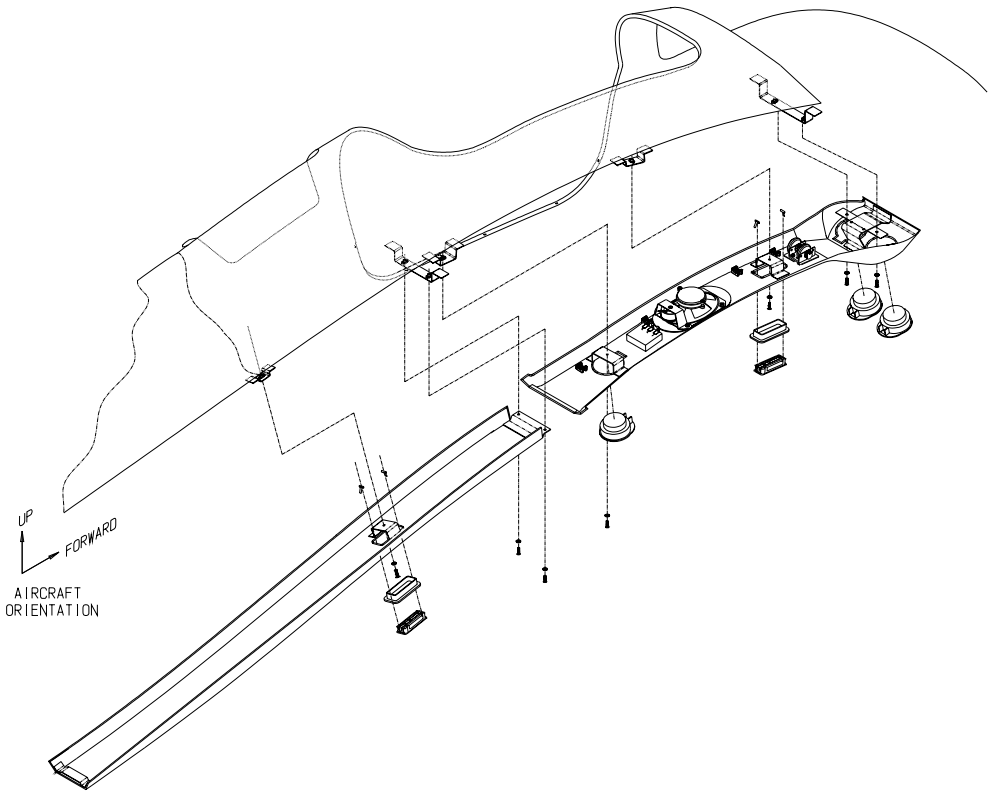

Figure 25 – 25 FCOLP & ACOLP (S/N 42003)

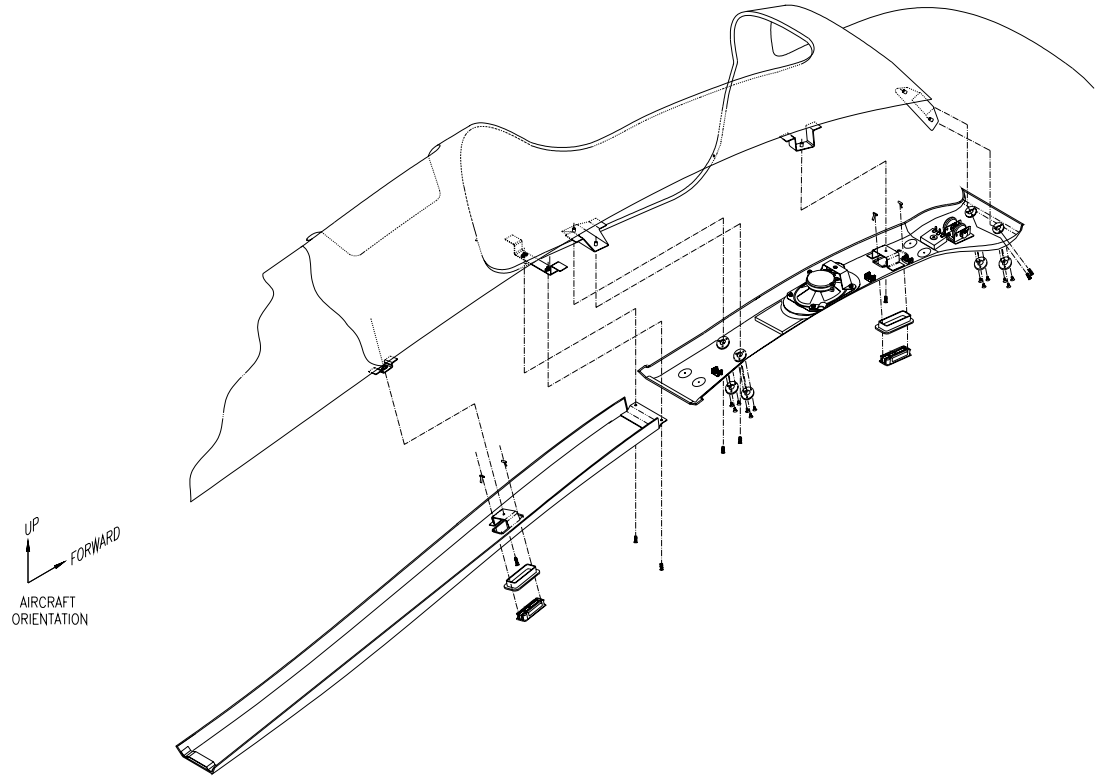

Figure 25 – 26 FCOLP & ACOLP (S/N 42004 to 42044)

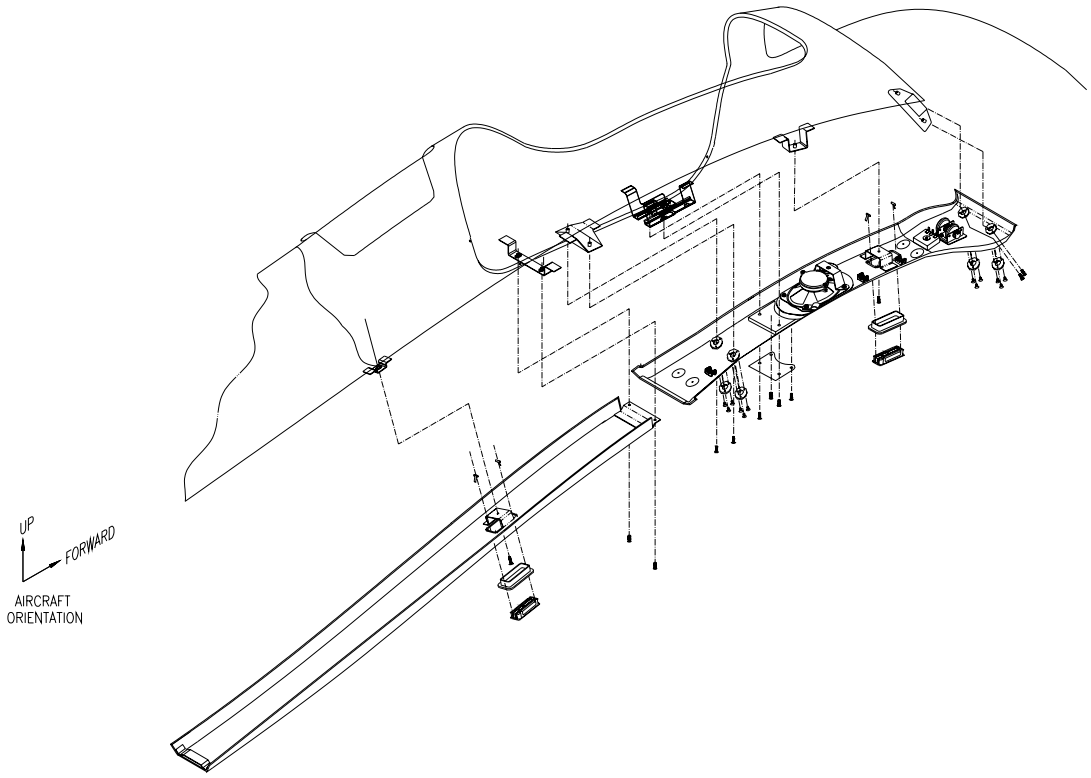

Figure 25 – 27 FCOLP & ACOLP (S/N 42045 to 42062)

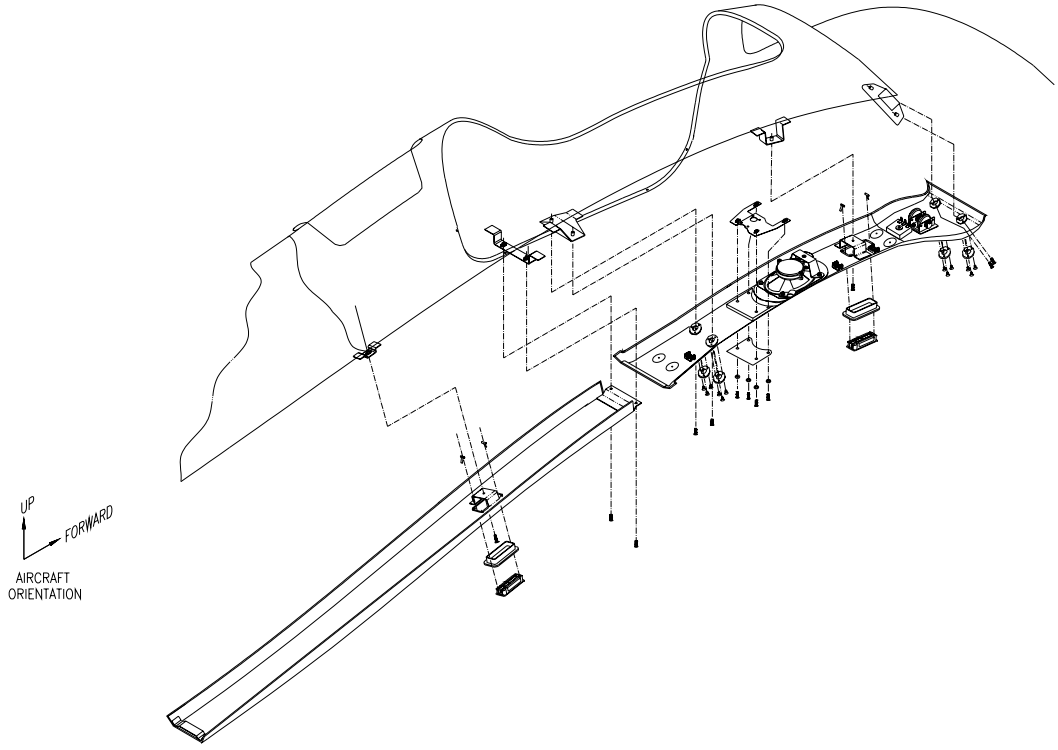

Figure 25 – 28 FCOLP & ACOLP (S/N 42063 and on)

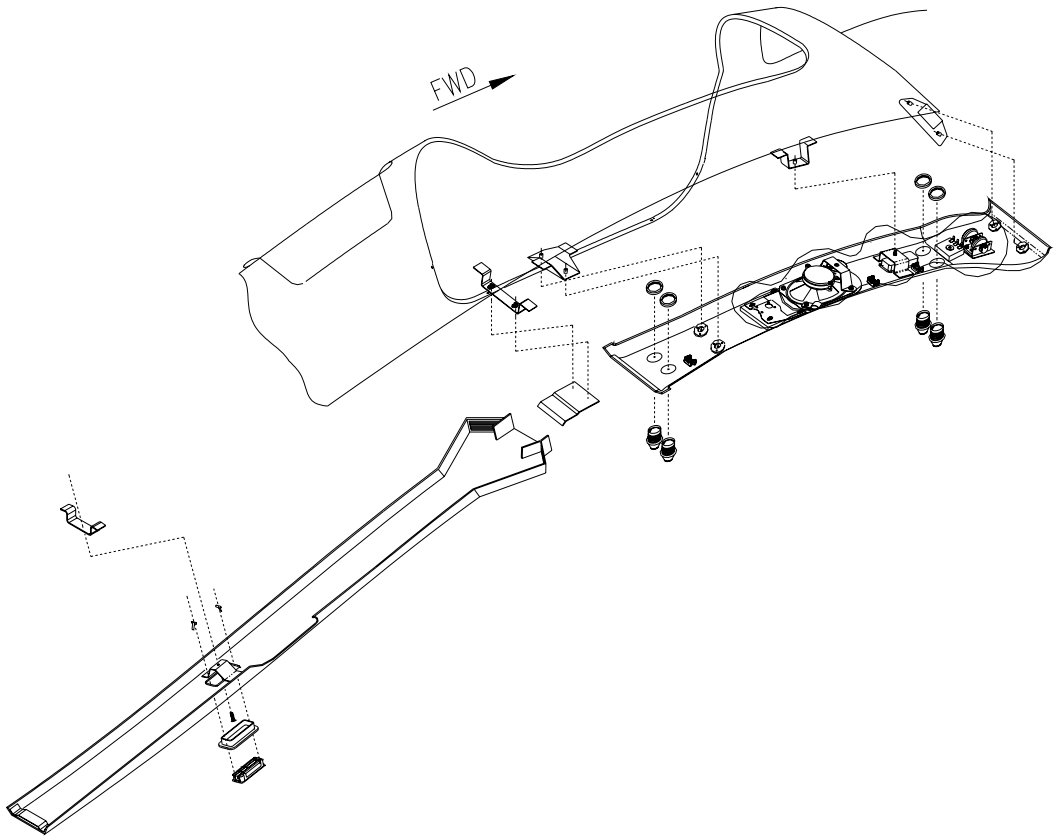

Figure 25 – 29 FCOLP & ACOLP With Air Conditioning

25-28. LEFT INSTRUMENT PANEL (Garmin G1000 Option)

a. **Removal** (See Figure 25 – 30)
   1. Remove the glare shield per paragraph 25-6.
   2. Remove the lower instrument panel per paragraph 25-30
   3. Remove the three 8-32 pan head screws, two 8-32 flat head screw and washer, and remove the left instrument panel.

b. **Installation** – Installation is the reverse of removal.

25-29. RIGHT INSTRUMENT PANEL (Garmin G1000 Option)

a. **Removal** (See Figure 25 – 30)
   1. Remove the glare shield per paragraph 25-6.
   2. Remove the lower instrument panel per paragraph 25-30
   3. Remove the three 8-32 pan head screws, two 8-32 flat head screw and washer, and remove the right instrument panel.

b. **Installation** – Installation is the reverse of removal.

25-30. LOWER INSTRUMENT PANEL (Garmin G1000 Option)

a. **Removal** (See Figure 25 – 30)
   1. Remove the two air valves (eyeball vents) per Chapter 21.
   2. Slide out and remove the lower instrument panel.

b. **Installation** – Installation is the reverse of removal.

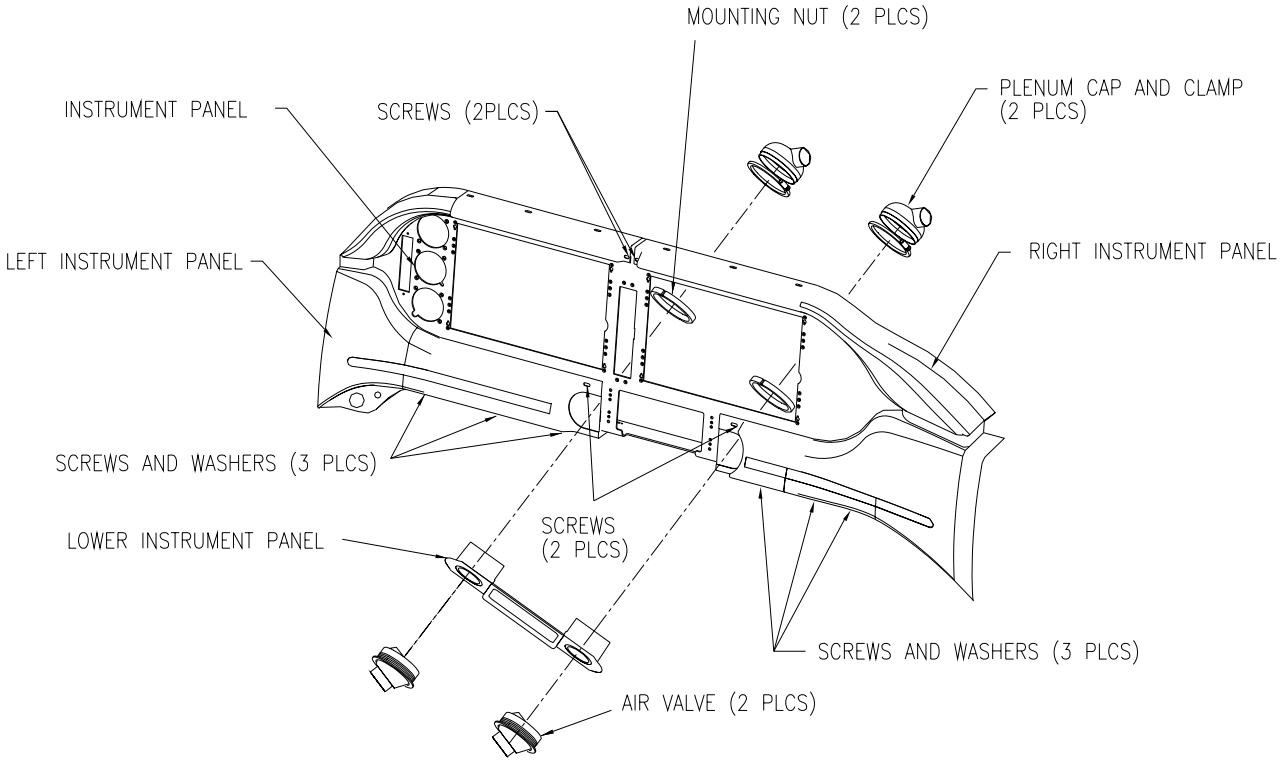

Figure 25 – 30 Upholstered Instrument Panels (Garmin G1000 Option)

25-31. DOOR TREAD

a. **Removal of Door Tread**
   1. Remove the five pan head 6-32 screws located on the exterior side of the door joggle. See Figure 25 – 32.
   2. Lift off the door tread that rests on the MLSPL and FLSPL. See Figure 25 – 31.

b. **Installation of Door Tread** – Installation is the reverse of above
   1. Torque fasteners snug.

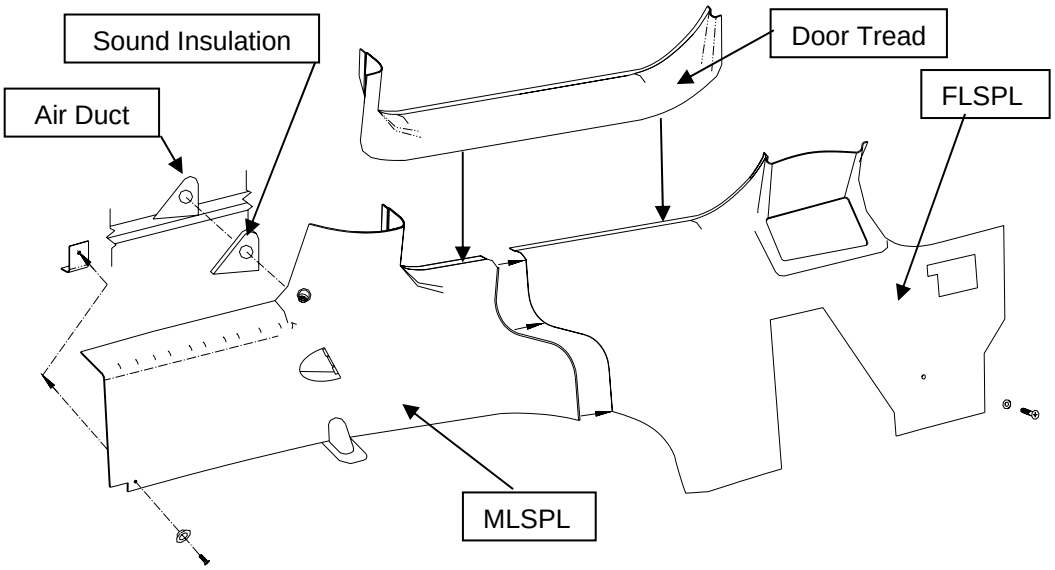

Figure 25 – 31 Door Tread, FLSPL, and MLSPL

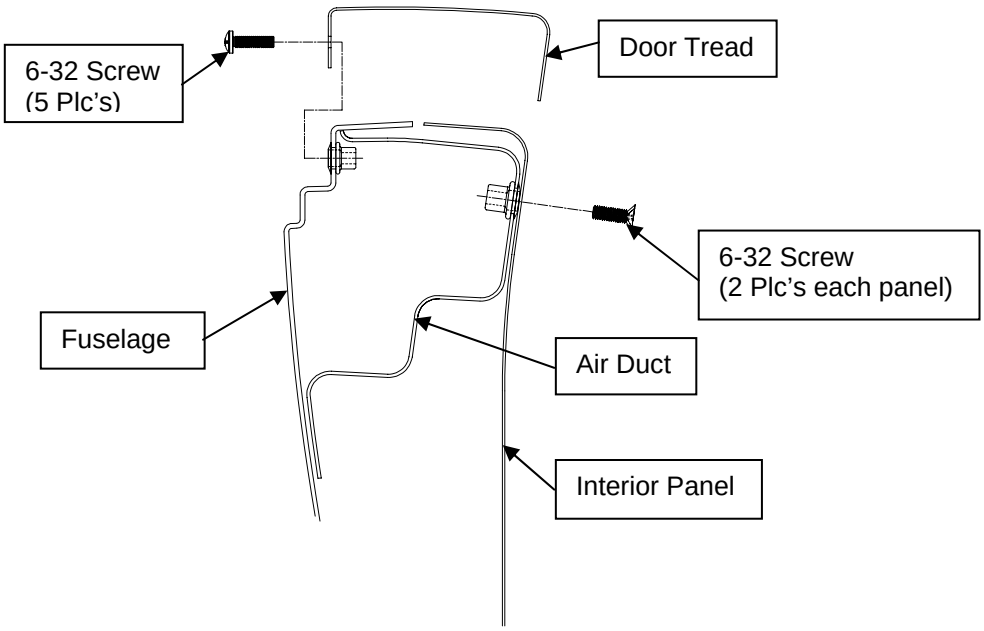

Figure 25 – 32 Side Detail of Door Tread Installation

25-32. FORWARD LOWER SIDE PANELS, LEFT (FLSPL) AND RIGHT (FLSPR)

a. **Removal of FLSPL and FLSPR**
   1. Remove the door tread per instructions in 25-23.a.
   2. Remove the two flat head 6-32 screws and washers at the top of the panel that were previously covered by the door tread. See Figure 25 – 31.
   3. Remove the flat head 10-32 screw and finishing washer located at the lower aft end of the panel.
   4. Detach the control stick boot from the FLSP. The boot is attached to the panel with Velcro.

b. **Installation of FLSPL and FLSPR** – Installation is basically the reverse of the above with the following exception to attach the control stick boot to the panel:
   1. Slide the FLSP part way down over the control stick.
   2. With the top of the panel as far inboard as possible against the control stick, reach behind the panel, from the top, and attach the outboard portion of the control stick boot to the panel.
   3. Tilt the lower edge of the panel as far inboard as possible, and reach up under the panel to finish attaching the control stick boot (forward, inboard, and aft) to the panel.
   4. Assure that the panel is not preloaded (the panel should not move out after the attachment screws are removed) so if the screw were to loosen, there would be no interference with the rudder controls.
   5. Torque fasteners snug.

25-33. MIDDLE LOWER SIDE PANELS, LEFT (MLSPL) AND RIGHT (MLSPR)

a. **Removal of MLSPL and MLSPR**
   1. Remove the door tread per instructions in paragraph 25-23.a.
   2. Remove the two flat head 6-32 screws and washers at the top of the panel that were previously covered by the door tread. See Figure 25 – 31.
   3. Remove the flat head 10-32 screw and finishing washer located at the lower forward end of the panel.
   4. Disconnect the vent by clipping the tie wrap that holds the tubing to the eyeball vent and sliding the tubing off the eyeball vent. Since the tie wrap is cut (destroyed) during removal, a new tie wrap must be used when reassembling the heater ventilation tubing.
   5. Disconnect the headphone/intercom jacks and microphone plugs per instructions in Chapter 23.

b. **Installation of MLSPL and MLSPR** – Installation is the reverse of the above. Torque fasteners snug.

25-34. FORWARD OVERHEAD SIDE PANELS, LEFT (FOSPL) AND RIGHT (FOSPR)

a. **Removal of FOSPL and FOSPR**
   1. Remove the FCOLP per instructions in paragraph 25-26.
   2. Remove the door tread per instructions in paragraph 25-23.a.
   3. Remove the six flat head 6-32 screws attaching the FOSP to the door joggle. See Figure 25 – 33.

b. **Installation of FOSPL and FOSPR** – Installation is the reverse of the above. Torque screws snug.

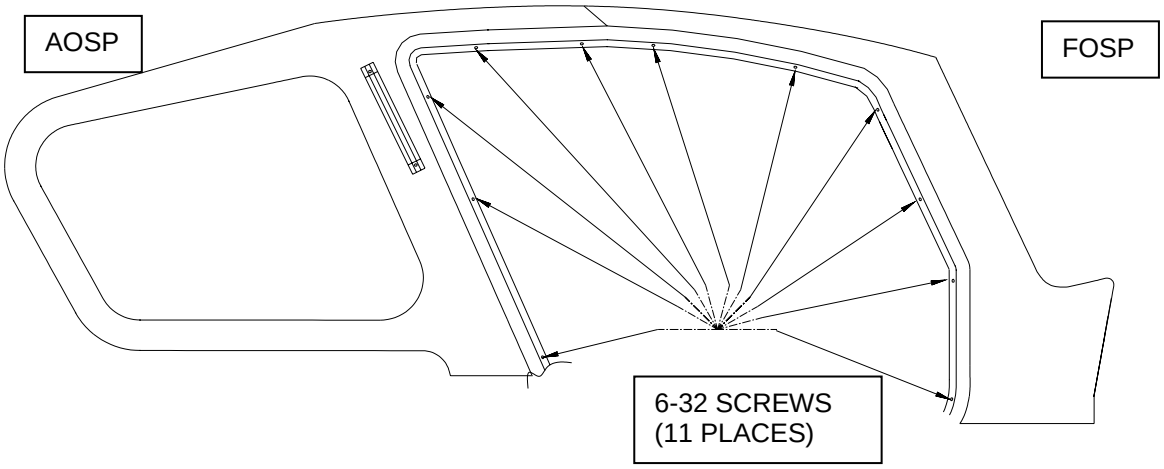

Figure 25 – 33 FOSPL/FOSPR and AOSPL/AOSPR Installation

25-35. AFT OVERHEAD SIDE PANELS, LEFT (AOSPL) AND RIGHT (AOSPR)

a. **Removal of AOSPL and AOSPR**
   1. Remove the FCOLP per instructions in paragraph 25-26.
   2. Remove the front shoulder belt guide per instructions in paragraph 25-21.a.2.
   3. Remove the rear seat shoulder belt retractor per instructions in 25-21.c.2.
   4. Remove the door tread per instructions in paragraph 25-23.a.
   5. Remove the five flat head 6-32 screws attaching the AOSP to the door joggle.

b. **Installation of AOSPL and AOSPR** – Installation is the reverse of above. Torque the flat head screws snug.

25-36. BAGGAGE LOWER SIDE PANEL, LEFT (BLSPL) AND RIGHT (BLSPR)

a. **Removal of BLSPL and BLSPR**
   1. Remove the two flat head 10-32 screws located on the topside of the baggage lower side panel at the forward and aft end.
   2. Lift the baggage lower side panel straight up off the four locating brackets bonded to the floor.

b. **Installation of BLSPL and BLSPR** – Installation is the reverse of the above. Torque the flat head screws snug.

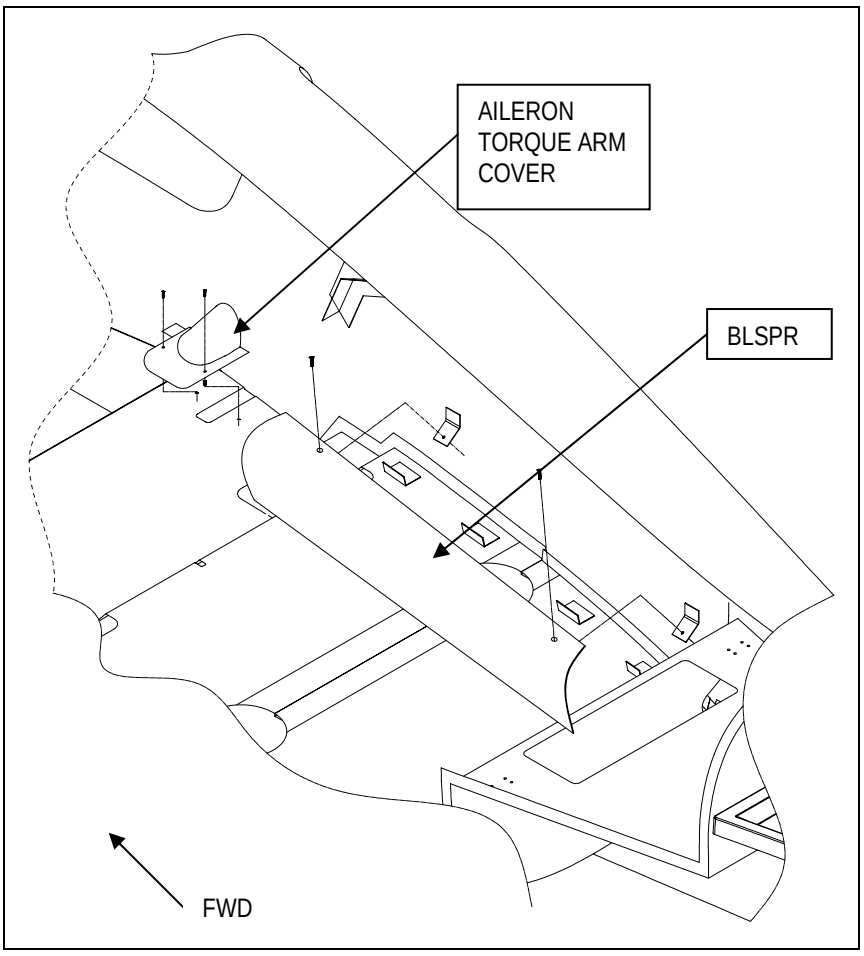

Figure 25 – 34 BLSPR and Aileron Torque Arm Cover

25-37. AILERON TORQUE ARM COVER

a. **Removal of Aileron Torque Arm Cover**
   1. Remove the left or right rear seat bottom cushion as required per instructions in 25-4.a.1
   2. Remove the two flat head 6-32 screws.

b. **Installation of Aileron Torque Arm Cover** – Installation is the reverse of the above. Torque the flat head screws snug.

25-38. HAT RACK CUSHION

a. The hat rack cushion is a friction fit against the rear bulkhead. Slip fingers between the cushion and the carpet and pull on the cushion to remove. Push into position to replace. The ELT label must face forward.

25-39. HAT RACK CARPET

a. The hat rack carpet is secured to the hat rack floor by the hat rack baggage net. Remove the hat rack baggage net (see paragraph 25-23.b) to lift out the hat rack carpet.

25-40. BAGGAGE FLOOR CARPET

On some aircraft, the baggage floor carpet contains a flap allowing removal of the access panel beneath without having to remove the carpet. Use the following procedures if baggage floor carpet removal is require.

a. **Removal of Baggage Floor Carpet**
   1. Remove the two baggage net cinch strap loops per instructions in paragraph 25-23.c.
   2. Remove the four baggage net anchor plates per instructions in paragraph 25-23.e.
   3. Remove the left and right baggage lower side panels per instructions in paragraph 25-36.
   4. Remove the hat rack carpet per instructions in paragraph 25-39.
   5. Remove the rear seat backs per instructions in paragraph 25-4.
   6. Pull up on the carpet to remove.

b. **Installation of Baggage Floor Carpet** – Installation is the reverse of the above.

25-41. CARGO COMPARTMENT

a. **Upper Hat Rack Baggage Net Tie Down Bracket** – The upper hat rack baggage net tie down bracket is attached to the fuselage with a hex head $^{1}/_{4}$-28 bolt and washer. Apply Permatex Form-A-Gasket #2 to the first two threads on the bolt prior to installation. Torque the hex head bolt 72 to 84 in.-lbs. Locate the bracket with the bent ears forward.

b. **Hat Rack Baggage Net** – The hat rack baggage net is attached to the hat rack shelf with two hex head $^{1}/_{4}$-28 bolts and washers. The bolt goes through a spacer to prevent crushing of the carpet foam. Torque the hex head bolt 72 to 84 in.-lbs. Locate the permanently attached baggage net brackets with the bent ears forward.

c. **Baggage Net Cinch Strap Loop** – Each baggage net cinch strap loop is attached to the baggage floor with two pan head 10-32 screws and washers. Torque the pan head screws 30 to 36 in.-lbs. when installing.

d. **Baggage Net** – The baggage net is attached to the floor at each corner with quick release fittings. Lift the locking block, and slide the post and locking block inward to detach.

e. **Baggage Net Anchor Plate** – The four baggage net anchor plates, located in the corners of the cargo compartment, are attached to the floor with two each flat head 10-32 screws. Torque the flat head screws 30 to 36 in.-lbs.

25-42. CONTROL STICKS

a. **Removal of Control Stick**
   1. Lift the control stick handle up far enough to gain access to the metal collar inside. If necessary, cut the tie-wraps at the base of the stick to release the attaching wires. If difficulty is still encountered, apply a light to moderate amount of heat (such as from a heat gun) to help in the handle removal.
   2. Clean the adhesive from the mating surfaces.
   3. Surface prep the mating surfaces with alcohol, acetone, or equivalent.
   4. Apply a new adhesive (Hysol 9321 or equivalent) to the top and sides of the shaft collar and to the inside wall of the grip, taking care not to allow adhesive to overlap threaded hole.
   5. The set screw for the control stick grip should face generally forward to hide the screw hole from view, but minor adjustments can be made to suit individual pilots.

6. After functionally testing the grip switches, install a new tie wrap at the base of the stick for continued security of wires.

b. **Control Stick Clearance Inspections –** The following inspections should be done when any maintenance is done to the control stick or if interior panels around the control stick are removed and replaced.

1. **Inspection No. 1 –** While sitting in the left seat, grasp the control stick **below** the grip, and move it throughout its entire range of travel, i.e., control stop to control stop. Ensure that neither the left of right stick grips make **any** contact with either of the side panels. If interference is noted, contact the factory immediately.

2. **Inspection No. 2 –** While sitting in the left seat, grasp the left control stick grip normally, and move it to the full left aileron position. Next, while maintaining the full left aileron position, move the stick to the full aft elevator position. Ensure that the left hand does not interfere with the full aft travel of the stick. The left hand may brush the side panel; however, if the left aileron does not move, the clearance is adequate. If interference is noted, contact the factory immediately.

3. **Inspection No. 3 –** While sitting in the right seat, grasp the right control stick grip normally, and move it to the full right aileron position. Next, while maintaining the full right aileron position, move the stick to the full aft elevator position. Ensure that the right hand does not interfere with the full aft travel. The right hand may brush the side panel; however, if the right aileron does not move, the clearance is adequate. If interference is noted, contact the factory immediately.

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25-43. FORWARD WING SADDLE ACCESS PANEL

a. **Removal of Wing Saddle Access Panel**
   1. Remove the right front seat pan as required per instructions in paragraph 25-2.a.
   2. Remove the four truss head 8-32 screws securing the panel to the floor. See Figure 25 – 35.

b. **Installation of Wing Saddle Access Panel**– Installation is the reverse of above. Torque the truss head screws 30 to 36 in.-lbs.

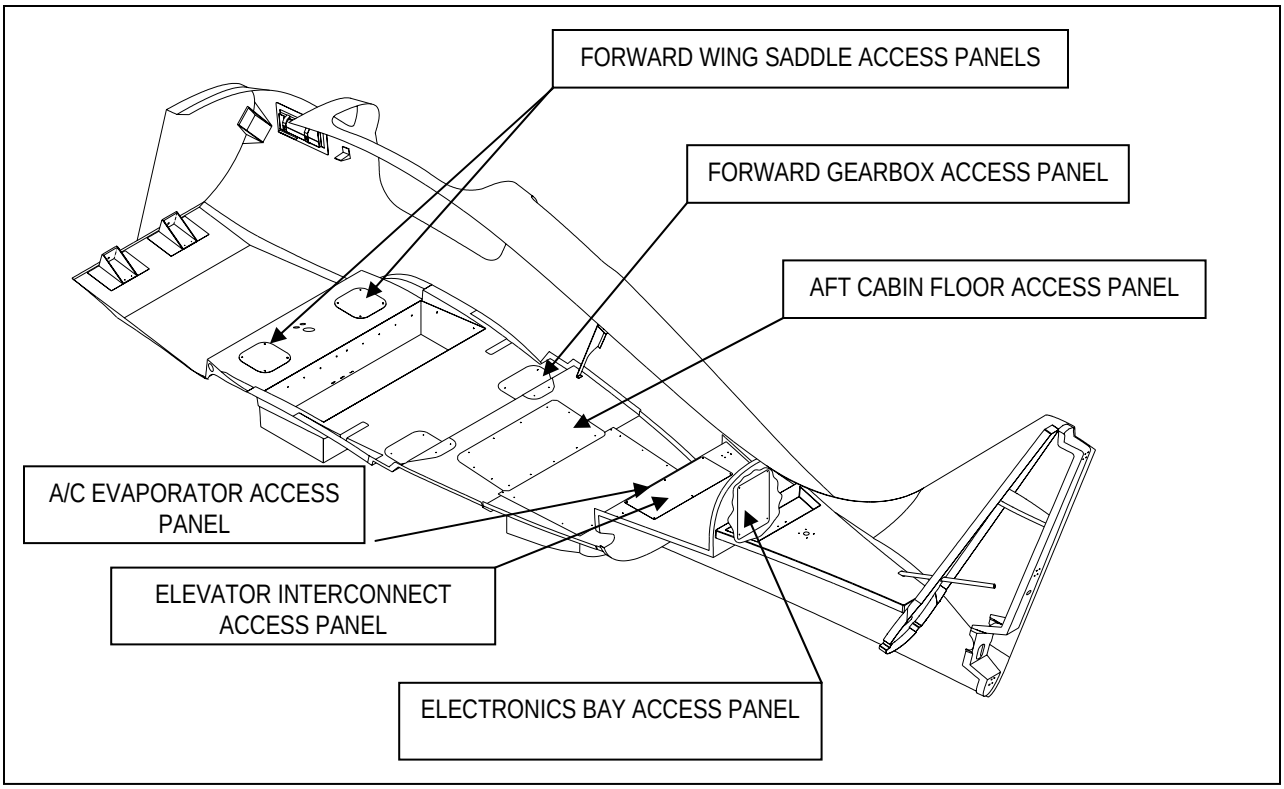

Figure 25 – 35 Access Panel Locations

25-44. FORWARD GEARBOX ACCESS PANEL, LEFT AND RIGHT

a. **Removal of Forward Gearbox Access Panel**
   1. Remove the rear seats per instructions in paragraph 25-4.
   2. Remove the baggage floor carpet per instructions in paragraph 25-40.
   3. Remove the four truss head 8-32 screws securing the panel to the floor. See Figure 25 – 35.

b. **Installation of Forward Gearbox Access Panel** – Installation is the reverse of above. Torque the truss head screws 30 to 36 in.-lbs.

25-45. AFT CABIN FLOOR ACCESS PANEL

a. **Removal of Aft Cabin Floor Access Panel**
   1. Remove the baggage floor carpet per instructions in paragraph 25-40.
   2. Remove the 13 truss head 8-32 screws securing the panel to the floor. See Figure 25 – 35.

b. **Installation of Aft Cabin Floor Access Panel** – Installation is the reverse of above. Torque the truss head screws 30 to 36 in.-lbs.

25-46. ELEVATOR INTERCONNECT ACCESS PANEL

a. **Removal of Elevator Interconnect Access Panel**
   1. Remove the hat rack carpet. See paragraph 25-39.
   2. Remove the 6 truss head 8-32 screws securing the panel to the floor. See Figure 25 – 35.

b. **Installation of Elevator Interconnect Access Panel** – Installation is the reverse of above. Torque the truss head screws 30 to 36 in.-lbs.

25-47. ELECTRONICS BAY ACCESS PANEL

a. **Removal of Electronics Bay Access Panel**
   1. Remove the hat rack cushion. See paragraph 25-38.
   2. Remove the hat rack carpet. See paragraph 25-39.
   3. Remove the four truss head 8-32 screws securing the electronics bay access panel to the aft bulkhead.

b. **Installation of Electronics Bay Access Panel** – Installation is the reverse of above. Torque the truss head screws 30 to 36 in.-lbs.

25-48. A/C EVAPORATOR ACCESS PANEL

a. **Removal and installation –** See Chapter 53 for removal and installation instructions.

25-49. EMERGENCY LOCATOR TRANSMITTER (ELT)

a. **Removal of ELT**
   1. Remove the electronics bay access panel per instructions in paragraph 25-47.
   2. For the Artex 200 move the ELT switch from the ARM position to the OFF position. The Artex ME406 ELT switch does not have an OFF position.
   3. Disconnect the ELT antenna cable from the ELT.
   4. Disconnect the ELT remote switch cable from the ELT.
   5. Release the ELT retaining buckle, or velcro strap as applicable, to free the transmitter from the mounting bracket.

b. **Installation of ELT** – Installation is the reverse of above with the following exceptions:
   1. After following step 5 through 2 (the switch is in the ARM position), make sure the ELT is not transmitting. If the ELT is transmitting:

      Reset the Artex 200 by turning the unit OFF, then back to ARM. If the ELT does not reset, turn the unit OFF, remove, and service.

      Reset the Artex ME406 by turning the unit ON, then back to ARM.

c. For additional information concerning the emergency locator transmitter, consult the ELT manufacturer's manual that came with the aircraft, or call the factory.

25-50. EMERGENCY LOCATOR TRANSMITTER ANTENNA

a. **Removal of ELT Antenna**
   1. Remove the electronics bay access panel per instructions in paragraph 25-47.
   2. For the Artex 200 move the ELT switch from the ARM position to the OFF position. The Artex ME406 ELT switch does not have an OFF position.
   3. Disconnect the antenna cable from the ELT antenna.
   4. Unscrew the antenna mounting nut and washer from the antenna mounting stud (lower part of antenna).

b. **Installation of ELT Antenna** – Installation is the reverse of above with the following exceptions:
   1. After following step 4 through 2 (the switch is in the ARM position), make sure the ELT is not transmitting. If the ELT is transmitting:

      Reset the Artex 200 by turning the unit OFF, then back to ARM. If the ELT does not reset, turn the unit OFF, remove, and service.

      Reset the Artex ME406 by turning the unit ON, then back to ARM.

c. For additional information concerning the ELT antenna, consult the ELT manufacturer's manual that came with the aircraft or call the factory.

25-51. EMERGENCY LOCATOR TRANSMITTER REMOTE SWITCH – If basic instruments only or Avidyne avionics are installed, the ELT remote switch is located in the right knee bolster. If the Garmin G1000 system is installed, the ELT remote switch is located to the right of the MFD.

a. **Removal of ELT Remote Switch**
   1. Remove the right knee bolster per instructions in paragraph 25-10 or reach under the instrument panel, as applicable.
   2. Disconnect the wiring from the remote switch.
   3. Remove the remote switch by removing the four pan head 4-40 screws that attach through the switch faceplate. The screws are secured with nuts and washers.

b. **Installation of ELT Remote Switch** – Installation is the reverse of the above with the following exceptions:
   1. After switch installation, make sure the ELT is not transmitting. If the ELT is transmitting:

      Reset the Artex 200 by turning the unit OFF, then back to ARM. If the ELT does not reset, turn the unit OFF, remove, and service.

      Reset the Artex ME406 by turning the unit ON, then back to ARM.

c. For additional information concerning the ELT remote switch, consult the ELT manufacturer's manual that came with the aircraft, or call the factory.

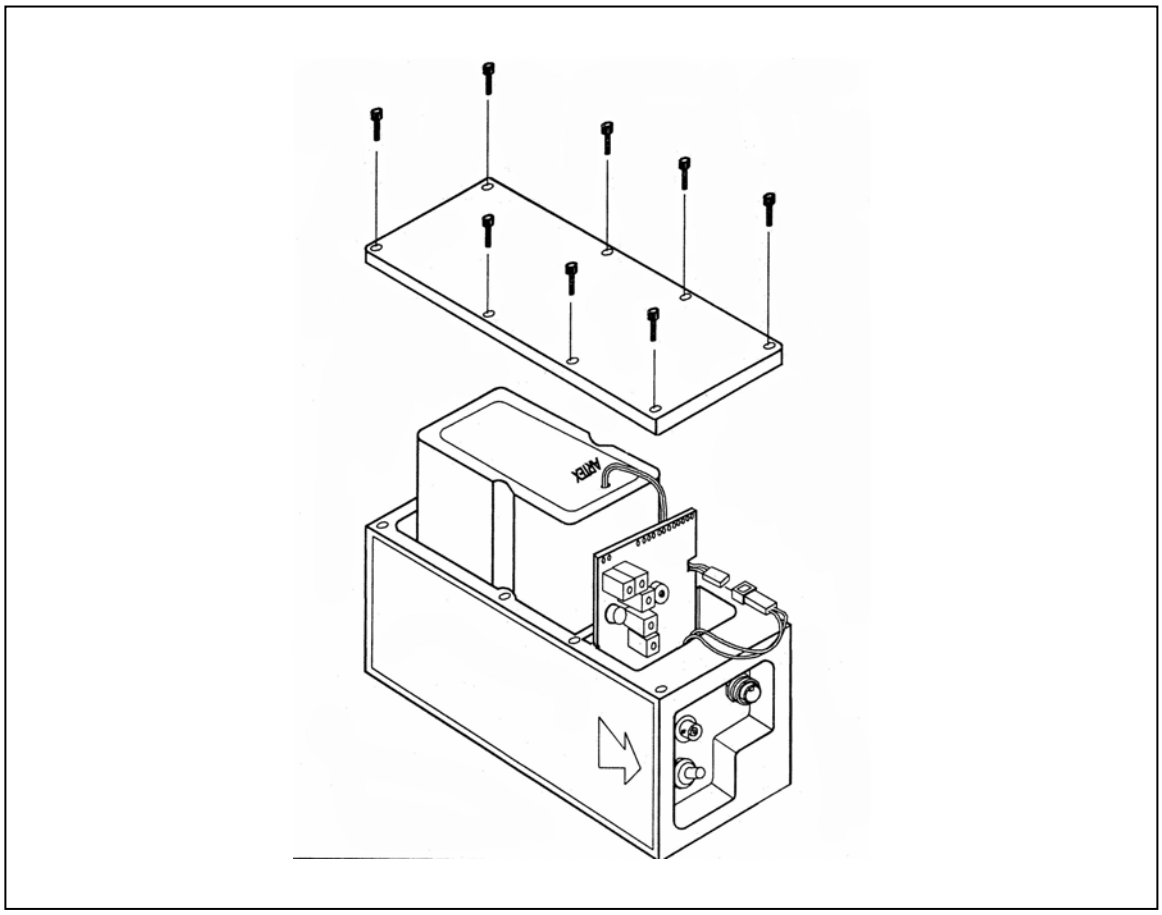

Figure 25 – 36 Artex 200 ELT Battery

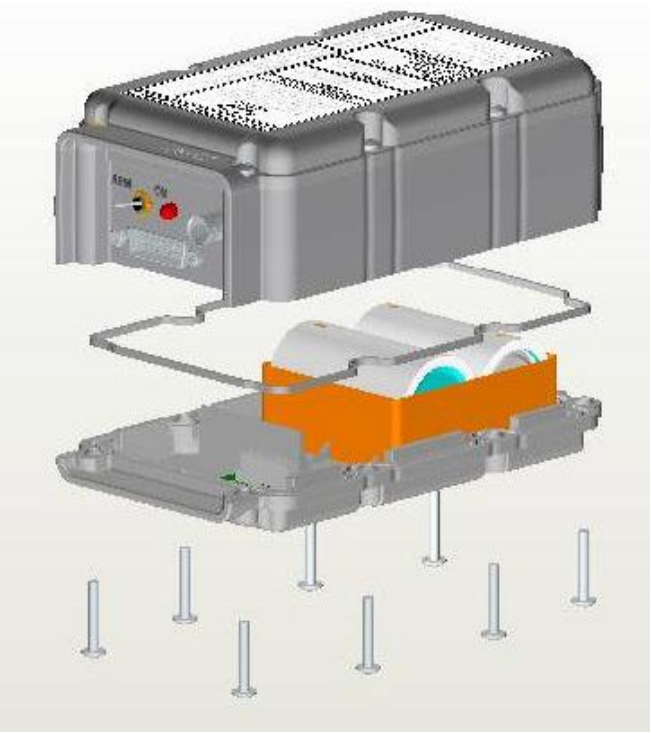

Figure 25 – 37 Artex ME406 ELT Battery

25-52. EMERGENCY LOCATOR TRANSMITTER BATTERY

a. **Artex 200 ELT**

1. **Removal of ELT Battery**
   (a) Remove the ELT per instructions in paragraph 25-49.
   (b) Remove the eight screws from the top ELT cover using a 3/32 Allen wrench.
   (c) Gently pull the circuit board up until the slot with the battery pack wires is above the top of the ELT case.
   (d) Unplug the battery from the circuit board.
   (e) Lift the battery pack (the large rectangular box with the wires protruding from it as shown in Figure 25 – 36) out of the ELT case.
2. **Inspection and Testing of ELT Battery**
   (a) Inspect the battery pack for leakage or corrosion. Replace with a new battery pack if any is found.
   (b) Check the battery pack voltage under load. Replace if the voltage under load is less than 9.0 vdc.
   (c) Measure the current drain. The ELT will need to be armed and not transmitting. Measured current should be less than 10 microamps.
   (d) Further inspection and testing procedures, if needed, can be found by consulting the ELT manufacturer's manual that came with the aircraft or by calling the factory.

3. **Installation of ELT Battery**
   (a) Plug the battery into the circuit board.
   (b) Route the battery pack wires through the circuit board slot, and push the circuit board back into place.
   (c) Route the battery pack wires into the notch in the center divider between the circuit board and the battery pack compartment.
   (d) When installing the top cover screws, do not exceed 5 in.-lbs. of torque.
   (e) Enter pertinent battery replacement information in the aircraft log book and fill out any other documentation required by local authority.
   (f) Reinstall the emergency locator transmitter.
   (g) Make sure the ELT is not transmitting. If the ELT is transmitting, reset by turning the switch OFF, then back to ARM. If the ELT does not reset, turn the unit OFF, remove, and service.

b. **Artex ME406 ELT**

1. **Removal of ELT Battery**

NOTE

The battery pack contains static sensitive parts, take electrostatic discharge precautions before handling.

   (a) Remove the ELT per instructions in paragraph 25-49.
   (b) Remove the eight screws from the battery-side cover, see Figure 25 – 37. Battery pack is identified by the embossed text: "BATTERY ACCESS ON THIS SIDE".
   (c) Carefully lift the battery cover (battery pack) away from the ELT and unplug the flex-cable connected to the pack. Do not pull on the flexible portion of the cable – use the rigid section of the flex circuit at the connector as a handle.
   (d) Inspect the battery pack and ELT chassis. The battery cells, components and connectors should be free of corrosion. Inspect flex-circuit for broken connections or damage. Ensure the battery housing is free of cracks or other visible damage.

2. **Installation of ELT Battery**.
   (a) Lay the battery pack on the work surface with the batteries facing up.
   (b) Install a replacement seal in the slot along the perimeter of the housing.
   (c) Leaving the battery as it is, position the ELT over the battery pack with one hand and plug the flex-cable connector into the battery assembly using the other. The cable should not be twisted and the connector should 'click' into place. The battery connector is keyed to prevent incorrect installation.
   (d) Mate the ELT to the battery, making sure that the seal is positioned correctly during the process.
   (e) Replace the 8 securing screws and torque to 10 – 12 inch-lbs.
   (f) Enter pertinent battery replacement information in the aircraft log book and fill out any other documentation required by local authority.
   (g) Reinstall the emergency locator transmitter.
   (h) Make sure the ELT is not transmitting. If the ELT is transmitting, reset by turning the switch ON, then back to ARM.

25-53. SOUNDPROOFING AND INSULATION

a. **Firewall and Footwell**

1. The soundproofing material on the firewall and in the footwells is constructed of fire resistant foam with a permanent adhesive backing. The adhesive backing results in destruction of the material if removed; therefore, the material cannot be repaired, but must be replaced if damaged. However, the entire panel does not have to be replaced. Only the damaged section has to be replaced, provided it can be cleanly removed without damaging the surrounding area. The adhering surfaces must be cleaned with an approved cleaner and method for composite materials before applying a new part. Large areas to be covered may be cut into smaller pieces for easier installation. Each panel must be trimmed to fit prior to installation. The precision of installation is critical, as the adhesive is so tenacious it is not possible to reposition afterwards. Avoid adhesive contacting skin. Avoid covering the mountings of maintenance items. Consult the factory if needed.

b. **Aft of Doors**

1. The sound proofing material aft of the doors and in the cargo area (headliner and sides) is a one piece, cloth covered, fire resistant foam with a permanent adhesive backing. The adhesive backing results in destruction of the material if removed; therefore, the material cannot be repaired, but must be replaced if damaged. The one-piece design necessitates the removal or partial removal of all of the following interior components prior to stripping off of the old material:

   a) Rear seats. See paragraph 25-4.
   b) Front seat belts. See paragraph 25-21.a.
   c) Rear seat belts. See paragraph 25-21.c.
   d) Baggage door surround. See Chapter 52.
   e) Forward center overhead light panel. See paragraph 25-26.
   f) Aft center overhead light panel. See paragraph 25-27.
   g) Middle lower side panels, left and right. See paragraph 25-33.
   h) Aft overhead side panels, left and right. See paragraph 25-35.
   i) Baggage lower side panel, left and right. See paragraph 25-36.
   j) Hat rack cushion. See paragraph 25-38.
   k) Hat rack carpet. See paragraph 25-39.
   l) Upper hat rack baggage net tie down bracket. See paragraph 25-23.a.

NOTE

After stripping the old material off, the adhering surfaces must be cleaned with an approved cleaner and method for composite materials before applying the replacement part. The precision of installation is critical, as the adhesive is so tenacious it is not possible to reposition afterwards. Avoid adhesive contacting skin. Avoid covering the mountings of maintenance items. Consult the factory if needed.

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25-54. Emergency Exit Hatchet

a. The emergency exit hatchet is located on the vertical face of the floor under the pilot's seat. The blade of the hatchet points down and is inserted in an aluminum sheath, and the unit is secured with Velcro strips.

b. **Removal of Hatchet and Brackets**
   1. Remove the Velcro strips, and remove the hatchet.
   2. If removal of the brackets is required, remove the two 8-32 flat head screws, washers, and self-locking nuts on the handle bracket.
   3. Remove the three 8-32 flat head screws, washers, and self-locking nuts on the blade bracket.

c. **Installation of Hatchet and Bracket –** Installation is the direct reversal of removal. Torque all screws 20 to 25 in.-lbs. Route the handle Velcro strap behind the bracket and around the handle as shown in Figure 25 – 38. Route the blade Velcro strap through the slot and under the blade as shown in Figure 25 – 38.

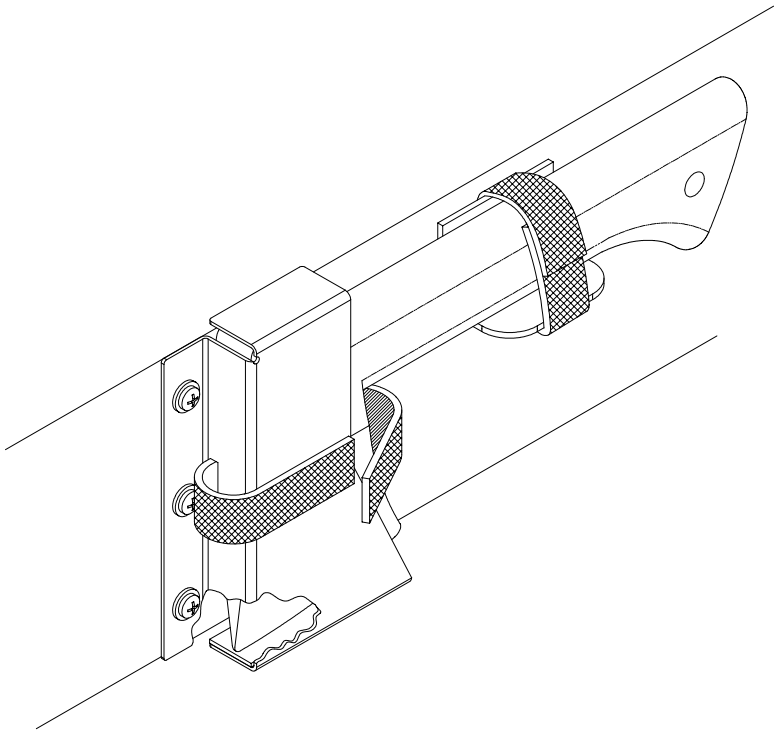

Figure 25 – 38 Emergency Exit Hatchet

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25-55. Cupholder (Garmin G1000 Option)

a. On aircraft equipped with the Garmin G1000, two retractable cupholders may be installed underneath the instrument panel. One left of the tower and one right of the tower. For the cupholder installed for Basic or Avidyne aircraft see 25-23.a.1.b).

b. **Removal and Installation of Cupholder**

1. Remove the two 8-32 x 5/16" MS51957-42 pan head screws and NAS1149FN832P washers affixing the cupholder to the offset spacers attached to the bottom flange of the instrument panel. See Figure 25 – 39.
2. Remove the cupholder.
3. Note the position and orientation of the offset spacers.
4. Installation is the reverse of removal, except as follows: Rotate the offset spacers as required to create a minimum of 1/4 in. clearance between the control stick and the cupholder pull when the cupholder is in the extended position. Verify that the inboard slide of the cupholder is located a minimum of 3.5 inches away from the outside of the center console. Up to two additional washers may be added to each 5/16" screw, as necessary, to obtain horizontal clearance between the bolster and the top edge of the cupholder pull. Dry torque the screws 17 to 21 in.-lbs.

c. **Service and Maintenance –** The cupholders do not require any service or maintenance.

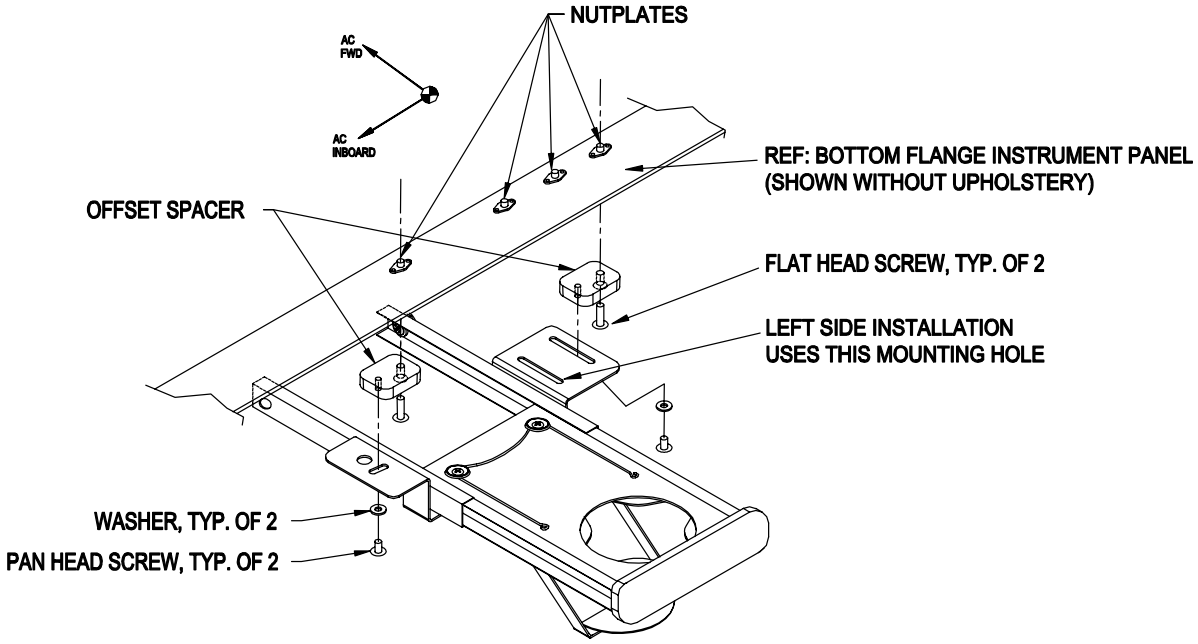

Figure 25 – 39 Cupholder

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25-56. Footwell – The footwell is a structural member of the fuselage and contains the front seat belt attachment bracket. It is installed with AN526C1032R10 screws and NAS1149F0363P washers (AN960-10). The length of these screws should be appropriate for the attachment.

a. Removal
   1. Remove all four seats.
   2. Lift the carpet in the footwell, and remove the screws and washers connecting the footwell to the flanges.
   3. Remove the footwell from the plane.

b. Installation
   1. Place the footwell within the flanges and install the screws and washers. Torque the screws 30 to 36 in.-lbs.
   2. Install the carpet.
   3. Install the four seats.

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CHAPTER 26

FIRE PROTECTION

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Chapter 26

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26-5	Removal and Installation of the Fire Extinguisher Bracket	26-20-00 / Page 2

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26-1. GENERAL – FIRE EXTINGUISHER DESCRIPTION AND LIMITATIONS

a. The airplane fire extinguisher is located below the copilot's seat in a metal bracket and is mounted horizontally. The extinguisher is stored with the top of the unit near the middle of the airplane so that it is accessible from the pilot's seat.

b. The extinguisher is filled with a 1211/1301 Halon mixture (commonly called Halonaire) that chemically interrupts the combustion chain reaction rather than physically smothering the fire. The hand extinguisher is intended for use on Class B (flammable liquids, oil, grease, etc.) and Class C (energized electrical equipment) type fires.

c. The installed fire extinguisher is an H_3R, Inc. Halon blend model 1200. The unit has a 5BC U/L rating and contains 2.5 lbs. of recycled Halon with a 14-16 second discharge time. It has been demonstrated that 2.5 lbs. of extinguishing agent, when totally discharged at one time within the cabin of the airplane during flight, can be safely ventilated from the cabin within an FAA prescribed time period. The ventilation time period is important since Halon has some levels of toxicity. For this reason, using other types of fire extinguishers, with lower U/L ratings, different extinguishing agents, or greater weights of Halon is not approved.

d. **Temperature Limitations** – The fire extinguisher has temperature storage limitations that may need to be considered depending on the operating environment of the airplane. If it is anticipated that the cabin temperature will exceed the extremes shown in the table below, Figure 26 - 1, the extinguisher must be removed and stored in a more temperate location.

Temperature Extremes	Maximum/Minimum Temperatures
Lowest Cabin Temperature	-40ºF (-40ºC)
Highest Cabin Temperature	120º F (49ºC)

Figure 26 - 1 Temperature Limits for Fire Extinguisher Storage

26-2. INSPECTING THE CHARGE OF THE FIRE EXTINGUISHER

a. Since the fire extinguisher does not have a pressure gauge to indicate its charge, it must be weighed at the annual or 100 hour inspection intervals to ensure there is an adequate amount of extinguishing agent. The extinguisher unit has a gross weight of 3.33 lbs. (± 1 ounce) when it is installed in the airplane and must be replaced if the gross weight is less than 3.1 lbs. Use the following steps to determine if the fire extinguisher is properly charged.

1. Remove the extinguisher from the airplane.
2. Place the unit on a sensitive scale that measures weight within about half an ounce (14 grams).
3. Verify that the gross weight of the unit is 3.1 lbs. (1420 grams) or more.

b. The variation between the weight of 3.33 pounds and the minimum allowable weight of 3.1 lbs. equates to a difference of 0.23 lb. or almost 105 grams. The fire extinguisher has a calibrated loss and can lose up to 4 grams of agent per year, or 24± grams during the

extinguisher's useful six year life. Clearly, a loss of over 105 grams is an indication of a serious problem, and the extinguisher must be replaced.

c. In any event, the extinguisher must be replaced after six years in service. It is not cost effective to reclaim the extinguisher's Halon, disassemble and inspect the unit, replace worn or defective parts, and then recharge it.

26-3. INSPECTING THE CONDITION OF THE FIRE EXTINGUISHER

a. Check the following items on the fire extinguisher and then reinstall in the airplane.
   1. Check condition of tamper seal for security.
   2. Check for damage to the discharge lever.
   3. Check condition of the valve nozzle.
   4. Check legibility and security of instructions decal.
   5. Ensure there are no dents or deformation to the cylinder.
   6. Note the extinguisher's gross weight on the 100 hour or annual inspection form.

26-4. INSPECTION OF THE FIRE EXTINGUISHER BRACKET

a. Inspect the bracket mounting screw holes for evidence of fatigue and pull-through.

b. Inspect the clamp mechanism for evidence of fatigue, and look for bends or crimps in the circular band. Install the extinguisher.

c. Ensure the latching mechanism functions properly. Tug on the extinguisher a few times. The extinguisher should remain in place even when a moderate force is applied. If the extinguisher has a tendency to break out of the bracket when a moderate force is applied to the unit, replacement of the bracket is required.

26-5. REMOVAL AND INSTALLATION OF THE FIRE EXTINGUISHER BRACKET

a. The fire extinguisher bracket is mounted to the aft side of the footwell on the copilot's side of the airplane. It is attached with two ¼ – 28 screws, two flat head washers, and two ¼ – 28 self-locking nuts.
   1. The bracket is mounted by inserting the screws into the countersunk portion of the bracket and then inserting it through the pre-drilled holes in the panel of the footwell. The washer and self-locking nut are applied to the back side of the panel and torqued to 72 in.-lbs.
   2. The bracket is uninstalled by removing the two ¼ – 28 self-locking nuts and flat head washers on the backside of the mounting panel.

b. Access to the back side of the mounting panel is available from two locations. There is an access panel under the copilot's seat, which requires removal of the seat (See Chapter 25) and an access panel on the underside of the airplane below the copilot's seat. While access is more convenient using the panel under the airplane, the removal and installation of the bracket requires two people.

CHAPTER 27

FLIGHT CONTROLS

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Chapter 27

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27-1. GENERAL – SCOPE AND DEFINITION

a. **General Description of System** – The Cessna 350 (LC42-550FG) has dual flight controls and may be flown from either the pilot or copilot's seat. Dual pairs of foot pedals actuate the rudder and control the brakes, which are also used for steering (differential braking). Push-pull tubes actuate the ailerons and elevator. The rudder is actuated via foot pedals and control cables. The wing flaps are electrically actuated via a drive motor and push-pull tubes. Rod end bearings are used throughout the control system. These bearings do not require any maintenance. Longitudinal pitch trim is achieved via an electrically actuated trim tab located in the elevator. Lateral trim is achieved via an electrically actuated trim tab on the right aileron. Directional trim is achieved via a fixed trim tab on the rudder.

b. **Tolerances and Play** – Figure 27 - 1 lists the allowable control and trim tab deflections, free play limits, and tolerances.

Control Surface Deflections, Tolerances, and Free Play Limitations

		Deflection	Tolerance	Maximum Free Play Limits*
Wing Flaps	Cruise Takeoff Landing	0° 12° 40°	±1°	---
Ailerons	Up Down	21.6° 17.7°	±1.0°	---
Aileron Trim Tab	Up Down	22.4° 19.6°	±1.0°	0.063 in. (2.5%c)
Aileron Servo Tab	Up Down	13° @ 17.7° ail. deflection 19° @ 21.6° ail. deflection	±2°	0.063 in. (2.5%c)
Elevator	Up Down	13.0° 12.0°	+0°/-0.5° ±1.0°	---
Elevator Trim Tab	Up Down	21° 30°	±1°	0.080 in. (2.5%c)
Rudder	Left Right	17° 17°	±1°	---
Rudder Limiter	Left	11.5°	±0.5°	---
Rudder Trim Tab	---	±30°	N/A	---

* The tab free play at the tab trailing edge should be less than 2.5% of the tab chord aft of the hinge line, measured at the station where the free play is measured.

Figure 27 - 1 Control Surface Requirements

c. **Rigging and Flight Controls** – Rigging and adjustment of the control system and surfaces is covered in their respective sections.

27-2. TRIM SYSTEM – GENERAL

a. The Cessna 350 (LC42-550FG) has a two axis electric trimming system. The elevator trim tab is located on the right side of the elevator, and the aileron trim tab is on the right aileron. Both tabs are electrically controlled by a hat switch on each control stick, and the trim position is annunciated on the trim panel located to the right of the rocker switch panel for the Basic or Avidyne option or the Engine Indication System (EIS) page of the MFD for the Garmin G1000 system. The trim servos are protected by one-amp circuit breakers. See Figure 27 - 2 or Figure 27 - 3 for an illustration of the system.

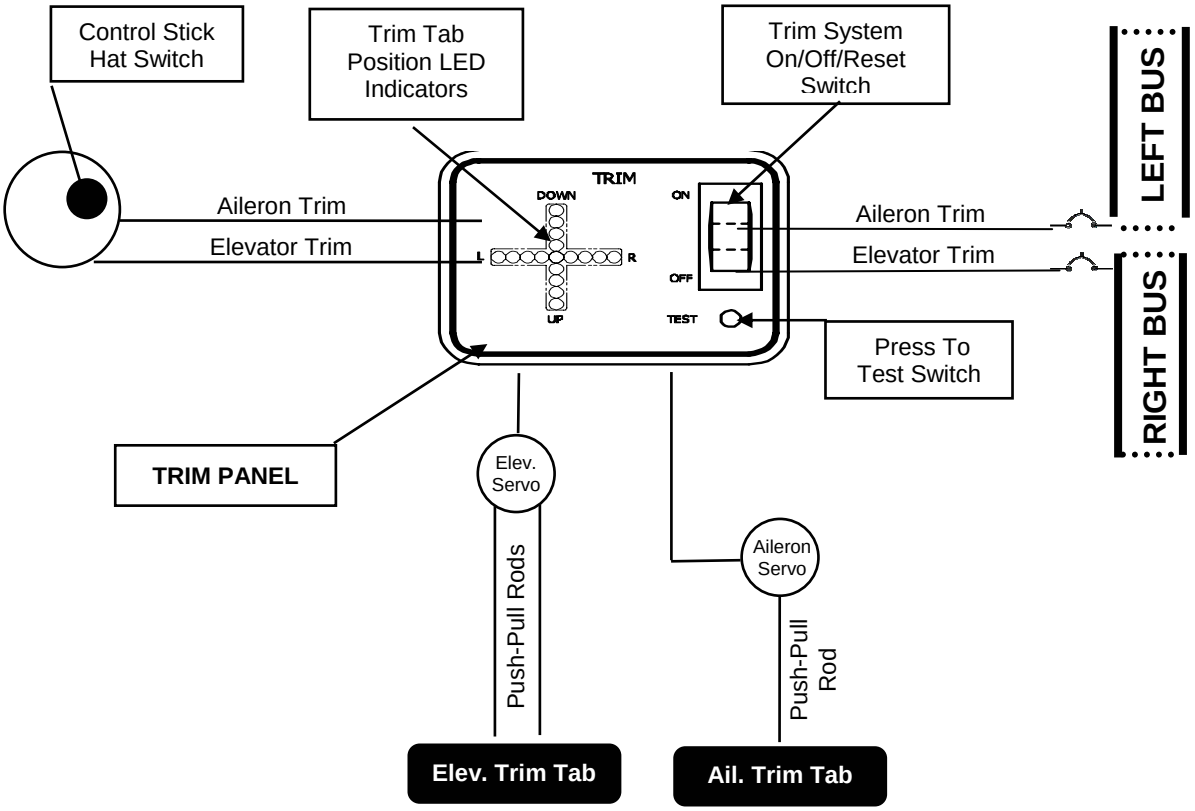


Figure 27 - 2 Trim System Layout (Basic or Avidyne Option)

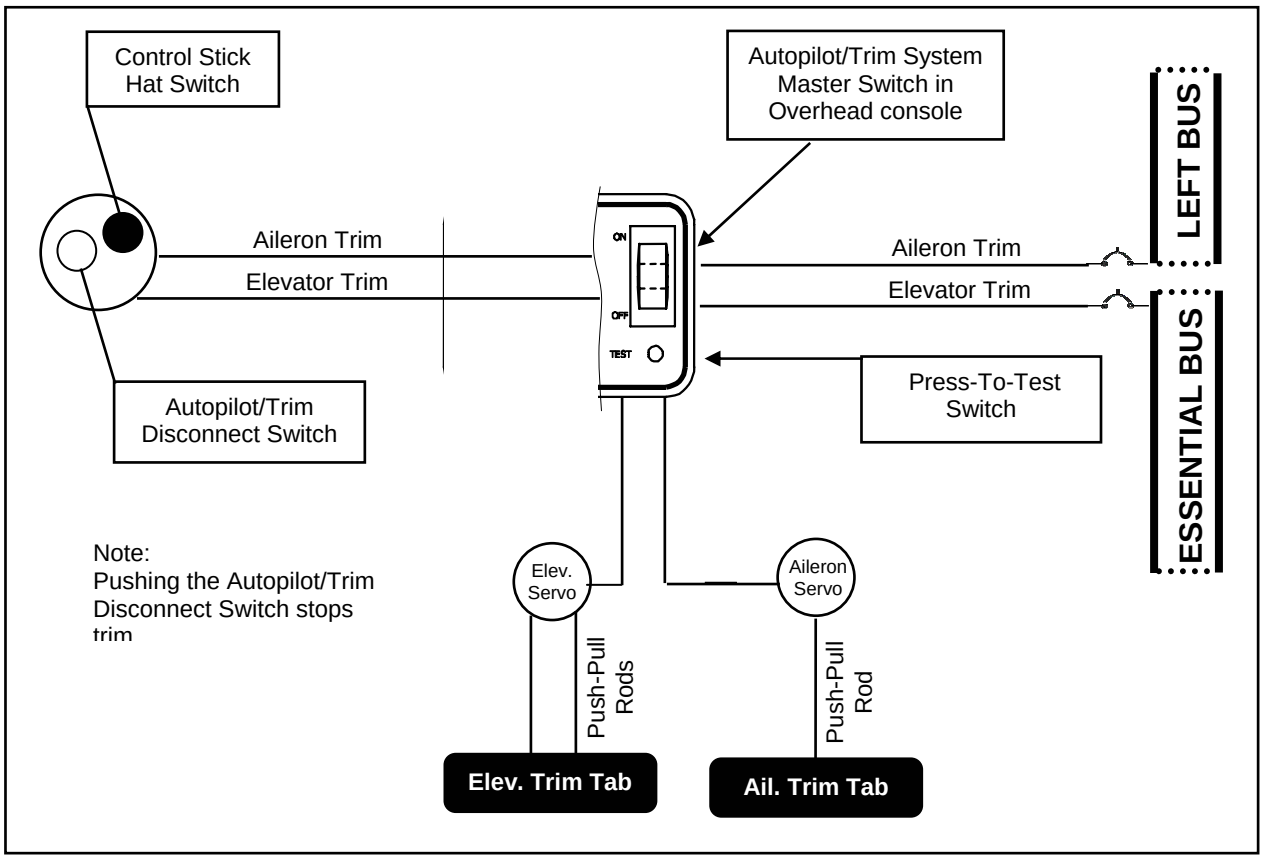


Figure 27 - 3 Trim System Layout (Garmin G1000 Option)

The trim surfaces are moved by push rods connected between each tab and a servomotor. The aileron tab has one actuating rod and the elevator tab has two. The second actuating rod on the elevator is a redundant system and is provided for the more critical tab in the system. The frictional device installed on the aileron tab should never be lubricated.

b. **Hat Switches** – The trim tabs are controlled through the use of a hat switch on the top portion of the pilot's and copilot's control stick at the three and nine o'clock positions, respectively. Moving the switch forward will correct a tail heavy condition, and moving it back will correct a nose heavy condition. Moving the hat switch left or right will correct right wing heavy and left wing heavy conditions, respectively.

c. **Simultaneous Trim Application** – If both switches, pilot's and copilot's, are moved in the same direction at the same time, the trim will operate in the direction selected. If the switches are simultaneously moved in opposite directions, e.g., pilot's is nose down and copilot's is nose up, the pilot's selection overrides the copilot's, and the trim will move to a nose down position (trim will not move with the Garmin G1000 option in this case). Finally, if the switches are simultaneously selected in different directions, e.g., elevator trim is input by one pilot and aileron trim is input by the other, each trim tab will move in the direction selected.

d. **Trim Position Indicator**

1. **Basic or Avidyne Option –** The trim position is displayed on two light bars using a series of blue LED's and two green LED's that are arranged on the trim panel in the shape of a plus sign. The vertical lights indicate the position of the elevator trim, and the horizontal lights show the position of the aileron trim. The middle green lights in each bar indicate the approximate neutral position.

   The blue lights are sequentially lit and extinguished as the trim tab moves through its range of travel. If the two green LED's in the middle of the "+" are lit and no blue lights are illuminated, both tabs are in the approximate neutral position. The LED's level of brightness is controlled by the position lights switch. When the position lights are on, the trim lights are in the dim mode, and when the position lights are off, the trim lights are in the bright mode.

   Should one or all of the trim indicator LED's stop functioning, remove the trim panel, and have it serviced by a qualified avionics repair technician.

2. **Garmin G1000 Option –** The trim position is displayed in the Trim Group on the Engine Indication System (EIS) page of the MFD. The vertical mark indicates the position of the elevator trim and the horizontal mark shows the position of the aileron trim. The blue band for each axis indicates the approved takeoff ranges.

e. **Trim On/Off/Reset Switch**

1. **Trim On/Off/Reset Switch (Basic or Avidyne Option)** – The trim system on/off/reset switch on the right side of the panel turns off power on all the trim tabs. This switch is used if a runaway trim condition is encountered. Refer to the POH for an expanded discussion of this issue. The press-to-test switch verifies the operation of all the LED's associated with the trim, flaps, fuel tank position, and annunciator panel. When the test position is selected, all related LED's illuminate in the bright mode. A light that fails to illuminate should be replaced.

2. **Autopilot/Trim Master Switch –** The autopilot/trim master switch, to the right of the avionics master switch in the overhead console, turns off power on all the trim tabs. This switch is used if a runaway trim condition is encountered. The switch can be cycled to reset or restore normal trim operations. Refer to the POH for an expanded discussion of this issue. The press-to-test switch is discussed in chapter 7 of the POH under the heading *Flaps, and 5 Pack Switch Annunciation (Press- to-Test)*.

27-3. CONTROL SURFACE STATIC BALANCING – GENERAL

Control surface balance must be checked when control surfaces are repaired, altered, or repainted. All control surfaces must be repainted in accordance with Chapter 51. The control surface balance limits shown in Figure 27 - 4 apply to a complete painted control surface only. Complete control surfaces include (as applicable) balance weight, all attachment hinges and hardware, trim tabs, grounding straps, and static wicks.

		Maximum Total Weight	TE Hinge Moment*
Aileron	Left	7.40 lbs.	+2.25 to +3.25 in.–lbs.
	Right	7.80 lbs.	
Elevator	---	32.30 lbs.	+37.00 to +39.00 in.–lbs.
Rudder	---	14.35 lbs.	+6.20 to +9.20 in.-lbs.
* All hinge moments are trailing edge heavy.			

Figure 27 - 4 Control Surface Static Balance Limits

NOTE

If specified moments cannot be met, heavier balance weights may be obtained from the manufacturer. This approach should be considered before reworking any repair or restripping and repainting the affected control surface. A lighter weight can be produced by shaving existing balance weight per the instructions outlined in the sections below.

27-4. REQUIRED EQUIPMENT FOR BALANCING

a. Obtain or otherwise fabricate two knife-edges (shown in Figure 27 - 5 Typical Knife-Edge) and supports approximately 6 - 12 in. in height such that they may be placed on a table and stabilized to prevent tipping. The knife-edge hinge supports must be level and perpendicular to the control surface hinge axis. The knife-edges should have attached a 1.5 to 2.0 in. section of MS20001P5 piano hinge to match the elevator attachment hinge. The knife-edges should also be small enough in height to fit within the rudder at the hinge pin locations.

b. A calibrated scale accurate to ± 0.01 lbs.

c. Calculator.

d. Calibrated leveling device (digital inclinometer, bubble level, etc) accurate to ± 0.2°.

27-5. BALANCING PROCEDURES – AILERON

a. **Aileron static balancing** – Use the following procedure for static balancing the ailerons.

1. Mount the two knife-edges horizontally such that a line drawn through the hinge line support points is level and perpendicular to the supporting knife-edges.
2. Set the aileron trim tab to the neutral position.
3. Remove the ailerons per the instructions in section 27-10.
4. On the left aileron, tape the servo tab in the neutral position with a small piece of masking tape. Tape the two drive rods to the servo tab hinge. Include all attachment hardware for the control rods.

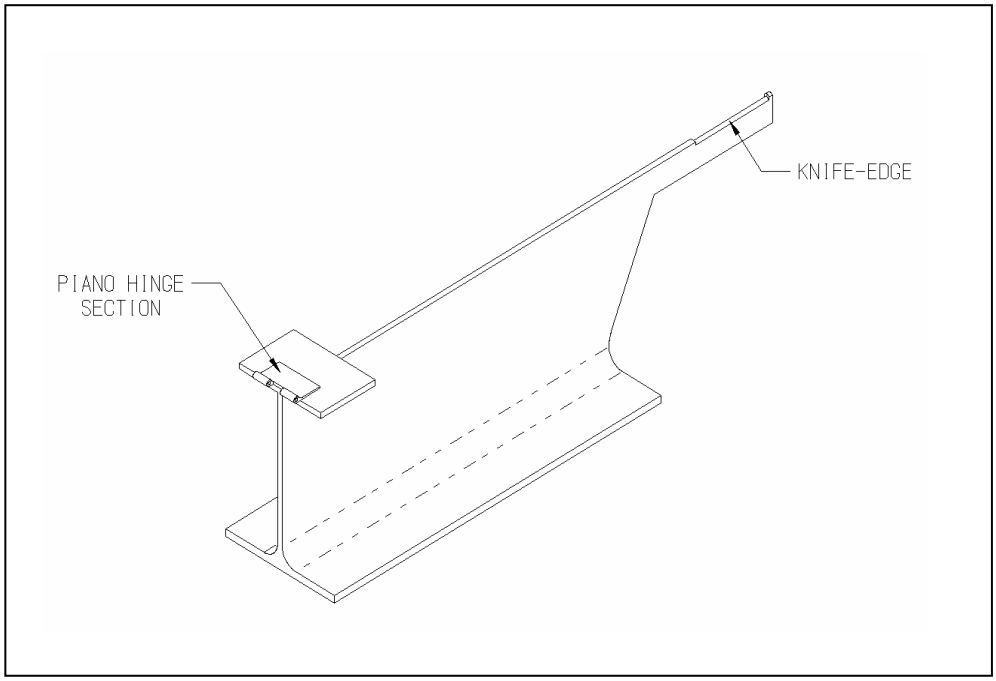


Figure 27 - 5 Typical Knife-Edge

5. Install the attachment hardware into each of the three hinges.
6. Place either aileron on the knife-edges such that the inboard and outboard attachment bolts are resting on the knife-edges. Swing the aileron up and down to ensure that the knife-edges do not interfere with the aileron and hinges.
7. Position the aileron such that the chord line is level, and measure the trailing edge weight at the inboard end. To check that the chord line is level, measure the angle of the top surface of the aileron at the inboard end, and confirm that it is 10.9° ± 0.2° from horizontal. See Figure 27 - 6.

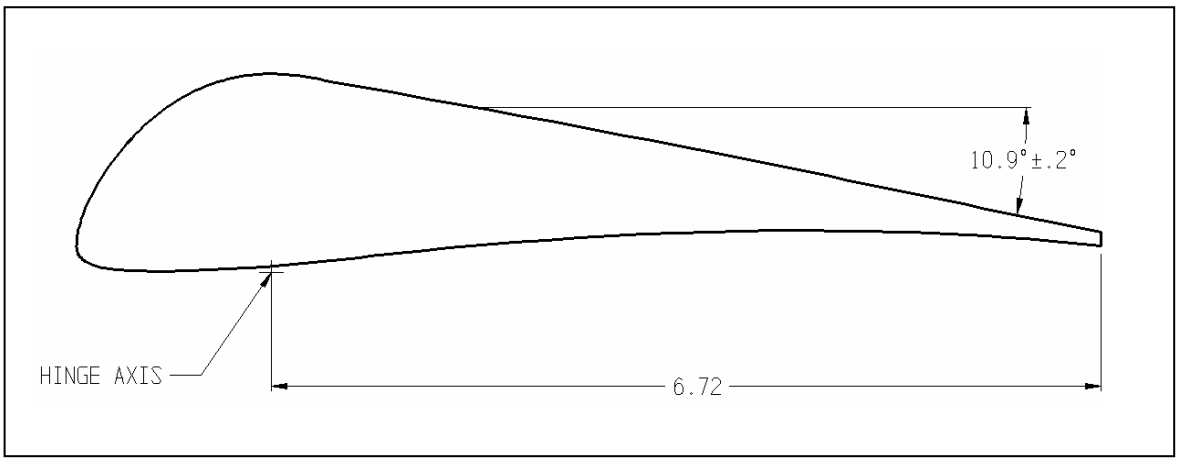


Figure 27 - 6 Level Aileron Chord Line and Moment Arm

8. Calculate the moment by multiplying the control surface load by 6.72 in. (moment arm). Compare this value with that in Figure 27 - 4.

9. If the moment is too light, material may be removed from the balancing weight to achieve the correct moment. Remove material from the aft face of the balancing weight first. If further weight reduction is needed, remove material from between the two fastener holes. The cross-hatched areas in Figure 27 - 7 indicate the areas where material may be removed from the aileron balancing weights. The width may be adjusted as long as a minimum of 0.125 in. gap is maintained between the counterweight and the aileron cove in the wing.
10. If the moment is too heavy, a heavier balancing weight may be obtained from the manufacturer.

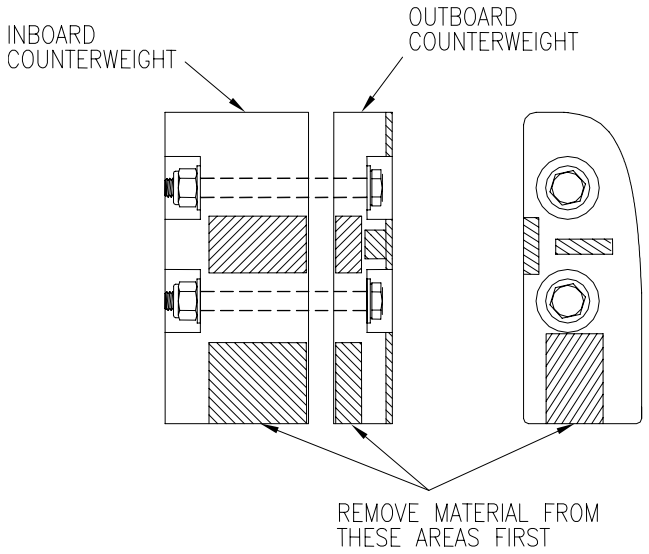


Figure 27 - 7 Allowable Areas of Material Removal, Aileron Balancing Weight

WARNING

Do not remove material from the forward or lower edge of the balancing weights. It is important that the contour of the forward and lower edge and the width of the balancing weights be maintained.

27-6. BALANCING PROCEDURES – ELEVATOR

a. **Elevator static balancing** – Use the following procedure for static balancing the elevator.

1. Mount the two knife-edges horizontally such that a line drawn through the hinge line support points is level and perpendicular to the supporting knife-edges.
2. Set the elevator trim tab to the neutral position.
3. Remove the rudder per the instructions in section 27-15.
4. Remove the elevator per the instructions in section 27-18.
5. Remove the forward half of the elevator attachment hinge by cutting the safety wire and removing the hinge wire.
6. Turn the elevator upside down, and align the hinge with the hinge sections located on the knife-edges. Reinsert the hinge wire into the piano hinge on the elevator.

NOTE

Position the knife-edges such that they are equal distant from each end of the elevator.

7. With the elevator chord line level, accurately measure the control surface load at the trailing edge a distance of 20.00 inches ± 0.125 inches from either end. The elevator chord line is level when the angle of the upper or lower surface is 6.0° ± 0.2°, measured perpendicular to the hinge line. See Figure 27 - 8.

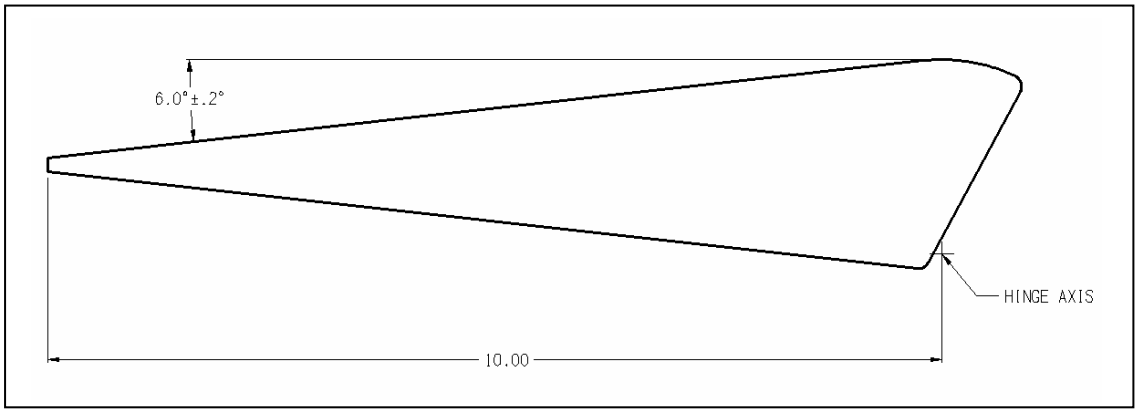


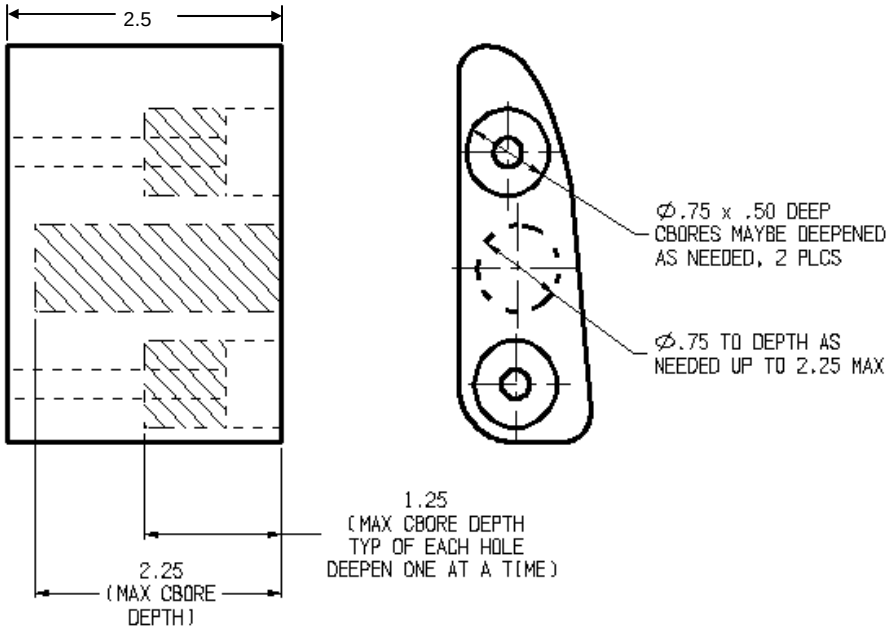
Figure 27 - 8 Level Elevator Chord Line and Moment Arm

8. Calculate the moment by multiplying the control surface load by 10.00 inches (moment arm). Compare this value with that in Figure 27 - 4.
9. If the moment is too light, material may be removed from the balancing weight to achieve the correct moment. Determine which weight is installed from Figure 27 - 9; cross-hatched areas indicate where material can be removed. Remove material from the aft mounting hole first, followed by the center hole, and the forward mounting hole last. When removing material from the balancing weights, remove equal amounts of material from the balancing weight on each end of the elevator.

WARNING

Do not remove material from the forward or lower edge of the balancing weights and do not reduce the overall width of the balancing weights. It is important that the contour of the forward and lower edge and the width of the balancing weights be maintained.

10. Use new MS20365-428A self locking nuts at final installation of the elevator balancing weights to the elevator. Torque the nuts 45 to 50 in.-lbs.



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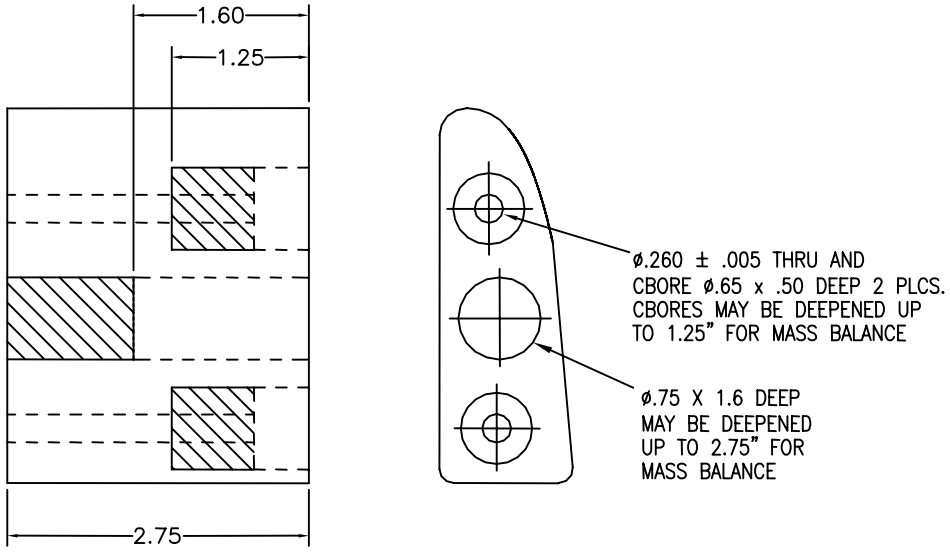

P/N LA55274107I and later

Figure 27 - 9 Allowable Areas of Material Removal, Elevator Balancing Weight

27-7. BALANCING PROCEDURES – RUDDER

a. **Rudder Static Balancing** – Use the following procedure for static balancing the rudder.
   1. Remove the rudder per the instructions in section 27-15.
   2. Mount both knife-edges horizontally and parallel to each other.
   3. Place the rudder on the knife-edges such that there is one knife-edge on the top and lower hinge pins. A line drawn through the hinge line support points must be level and perpendicular to the supporting knife-edges. Move the rudder up and down to

ensure that the knife-edges do not interfere with the rudder and cause friction that could affect the surface load.

4. With the rudder chord line level, accurately measure the control surface load at the trailing edge at the lower edge of the trim tab, see Figure 27 - 10. The rudder has a symmetrical cross section and thus the chord line is level when the center of the leading edge is at the same height as the center of the trailing edge.
5. Calculate the moment by multiplying the control surface load by 10.00 in. (moment arm). Compare this value with that in Figure 27 - 4.
6. If the moment is too light, the balancing weight may be shaved down to achieve the correct moment. When shaving the balancing weight, maintain a minimum thickness of 0.375 in. from the edge of the bolt hole to the top edge of the part, see Figure 27 - 11 for details.

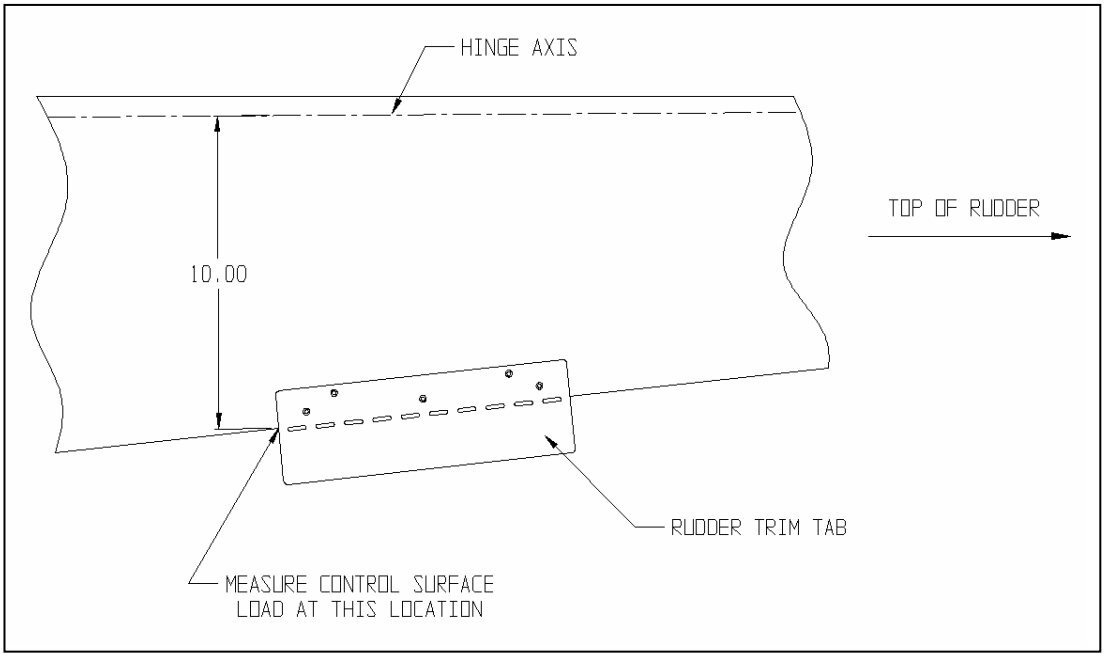

Figure 27 - 10 Location for Measuring the Rudder Surface Load

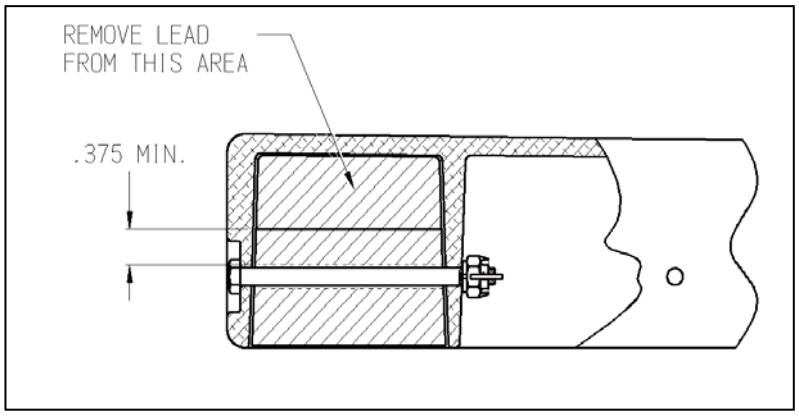

Figure 27 - 11 Allowable Area of Material Removal, Rudder Balancing Weight

27-8. RUDDER LIMITER SYSTEM

a. **Description of Rudder Limiter System**

1. **Operation –** The rudder limiter is an integral part of the stall system on the Cessna 350 (LC42-550FG). The system is a part of the safety-of-flight design of the Cessna 350 and is an important factor in spin resistance. It is designed to limit full left rudder deflection from a normal (approximately) 17.0° ± 1.0° to approximately 11° ± 0.5° during certain aircraft angle of attack and engine conditions. The system is activated when two conditions simultaneously exist: (1) the airplane's stall warning is active and (2) the engine manifold pressure is more than 12 in. of Hg. When the system is activated, a solenoid near the left rudder pedal moves a cam that physically limits the travel of the left rudder pedal. There is a time delay of approximately one second before the system is activated. This delay feature prevents inadvertent activation of the rudder limiter in turbulent air.
2. **Fail-Safe Feature –** The rudder limiter comes with a fail-safe feature. The system is armed when the airplane's electrical power is turned on; however, all electronic and electrical switching is in the relaxed position. When the stall warning is active and manifold pressure is more than 12 in. of Hg, the system activates from this "relaxed armed" position. If either of the two inputs to the system should fail, the rudder limiter will still operate. For example, if the manifold pressure transducer becomes inoperative, the rudder limiter will be activated by the sole input from the stall warning. Conversely, if the stall warning fails, the rudder limiter will be functional, provided the stall warning detector is operative, i.e., freely moves up and down. The operating condition of this fail-safe system can be verified from time to time through the use of a special ground testing procedure.

   The rudder limiter operates only when the stall warning is active and the throttle is advanced to a setting that produces more than 12 in. of Hg manifold pressure. Deflection of the right rudder pedal is unaffected by the rudder limiter system.

b. **Rudder Limiter Assembly Removal**

1. The rudder limiter assembly is located on the pilot's side of the cabin, on the left side of the rudder crossover tube behind the rudder pedals. To easily access the rudder limiter assembly, you must first remove the pedals, the knee bolster, and the pilot's seat. See Chapter 25 and 27 for details on removal of these items.
2. Unfasten the rudder limiter assembly by unscrewing the cap screws as shown in Figure 27 - 12.

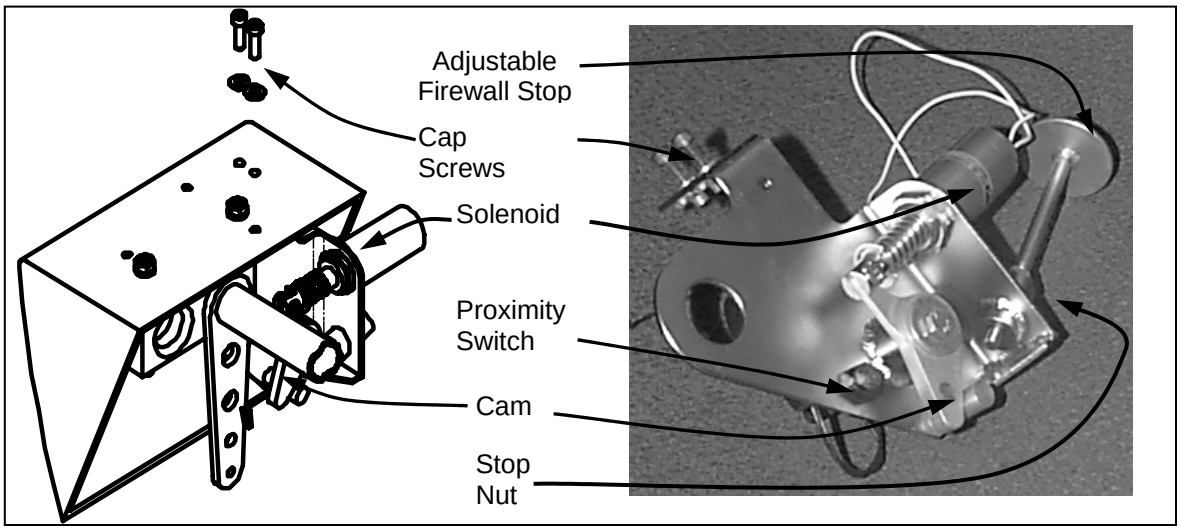

Figure 27 - 12 Rudder Limiter Assembly

3. Disconnect the wires on the solenoid and proximity switch.
4. Remove the rudder limiter assembly.

c. **Rudder Limiter Assembly – Installation**

1. Perform installation in the reverse sequence of the removal procedure. See Figure 27 - 12.
2. Adjust the firewall stop by loosening the stop nut and screwing out the stop until it firmly touches the firewall. Re-tighten the stop nut.
3. Re-connect the wires to the proximity switch and solenoid.
4. Test function of the rudder limiter assembly by following the steps listed in paragraph f.

d. **Rudder Limiter Solenoid Replacement –** It is not necessary to remove the entire rudder limiter assembly when replacing the solenoid, shown in Figure 27 - 13.

1. Disconnect the solenoid.
2. Screw the nut off the solenoid, and slide the solenoid off the bracket.

NOTE

Care must be taken to prevent the spring that is located on the other side of the nut holding the solenoid from coming off.

3. Installation is the opposite of removal. Torque the nut 100 to 110 in.-lbs.
4. Test function of the rudder limiter assembly by following the steps listed in paragraph f.

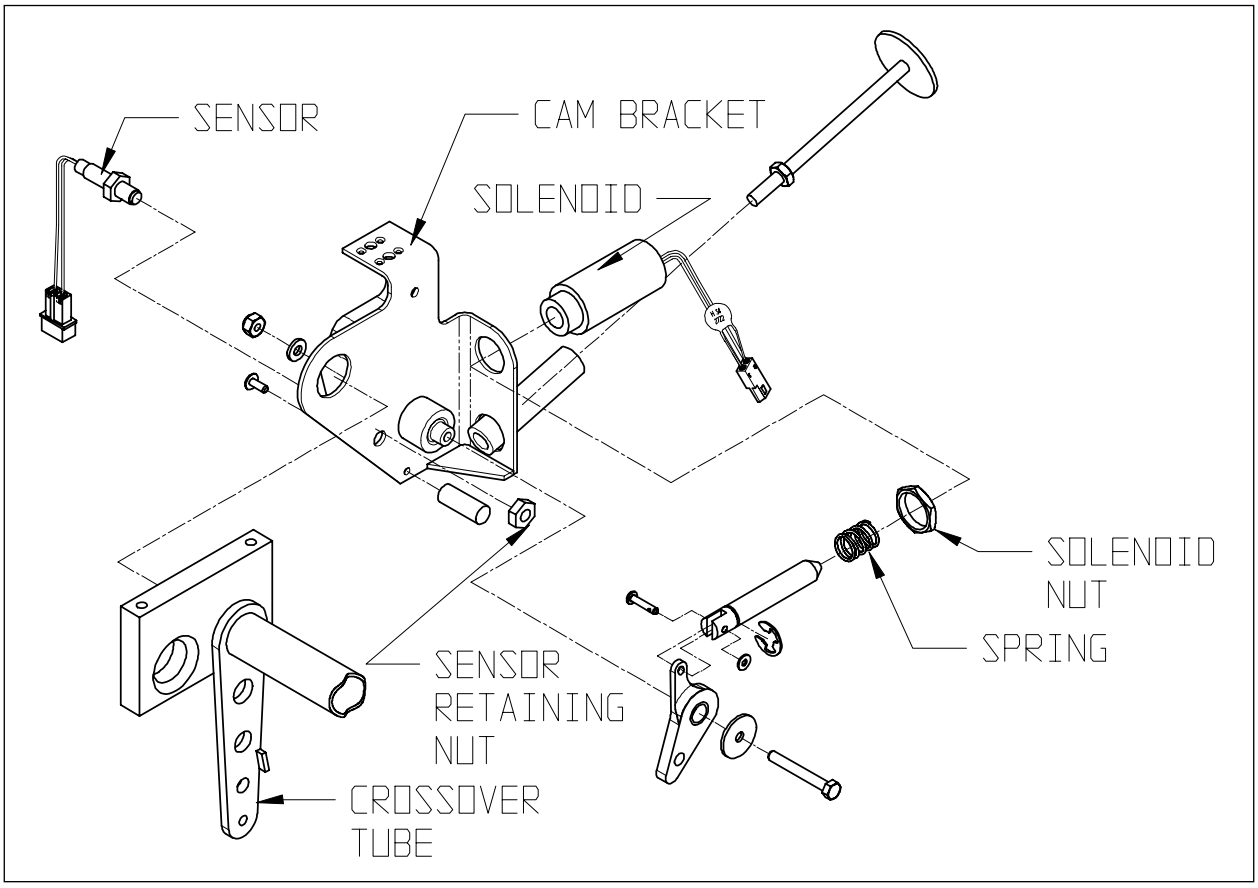

Figure 27 - 13 Rudder Limiter Assembly

e. **Rudder Limiter Sensor Replacement –** It is necessary to remove the entire rudder limiter assembly when replacing the sensor, shown in Figure 27 - 13.
   1. Remove the rudder limiter assembly.
   2. Remove the retaining nut from the sensor, and slide the sensor off of the rudder limiter bracket.
   3. Installation is the opposite of removal.
   4. Ensure the gap between the end of the sensor and limiter cam is 0.15 ± 0.07 in.
   5. Ensure proper on and off operation by checking continuity. With the cam nearest the sensor, the circuit should close. With the cam against the stud the circuit should be open.

f. **Rudder Limiter Test –** The purpose of this test is to verify operation of the manifold pressure transducer, the solenoid and rudder pedal cam, and the general integrity of the rudder limiter system.
   1. While sitting in the pilot's seat with the master switch on and the engine off, depress the test button on the trim panel. See Figure 27 - 14 for Basic or Avidyne option only. With the Garmin G1000 option the rudder limiter test button is in the overhead console beneath the A/P trim switch.
   2. When the test button is depressed, the pilot will hear (after a one-second delay) and feel the solenoid near the left rudder pedal engage, at which time the RUDR LMTR annunciator will illuminate for the Basic or Avidyne option or display in the

annunciation window of the PFD for the Garmin G1000 option, and the left rudder travel will be restricted.

3. When the operation is verified, release the test switch. The rudder limiter test switch is also used to test the integrity of the trim, flap, fuel tank LED's (Basic or Avidyne option), and 5 pack switches (Garmin G1000 option). The pilot should remember that anytime these lights are tested, the rudder limiter will engage.
4. With the master switch on and the engine off, pull the stall warning circuit breaker and have someone move the stall vane to the up position.
5. The rudder limiter should engage even though there is no aural stall warning.
6. Repeat the procedure with the engine instrument's circuit breaker pulled and the stall warning breaker reset.
7. This time the rudder limiter will engage with an aural stall warning, even though there is no manifold pressure indication.

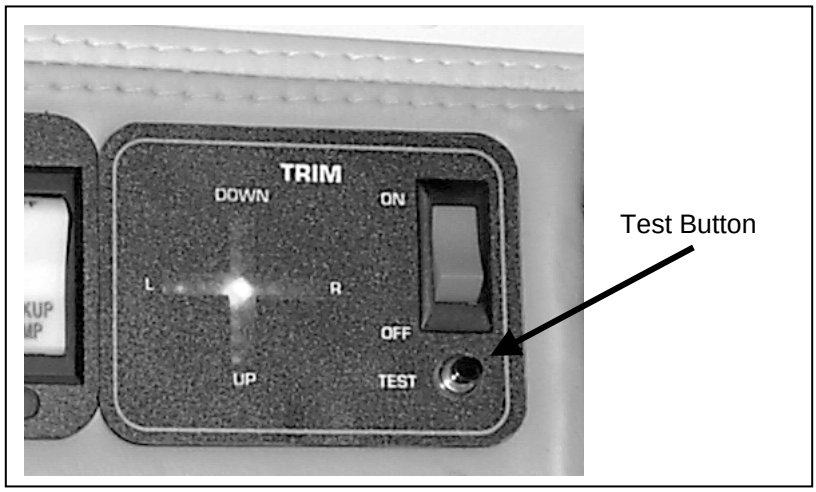

Figure 27 - 14 Rudder Limiter Test Button (Basic or Avidyne Option Only)

WARNING

The rudder limiter is an integral part of the Cessna 350 aircraft. It is mandatory that the rudder limiter be in full working order for any flight. Should problems exist that are not described in this manual, contact the factory for assistance.

NOTE

The rudder limiter assembly is factory-set and should be replaced as a unit instead of field modification of components.

g. **Troubleshooting the Rudder Limiter Cam System**

SYMPTOM	PROBABLE CAUSE	ACTION
Rudder limiter fails to activate	a) Solenoid does not activate	1. Check the rudder limiter cam system circuit breaker. 2. Press the test button, and listen for the solenoid to activate. 3. Push the stall vane up. After about a second, a "click" near the pedals should be heard as the rudder limiter engages.
	b) Bad wiring or connectors (refer to wiring manual for further details)	1. Check for continuity and current at the solenoid. Replace bad connectors or wiring. 2. Check the circuit breaker, and replace breaker if needed. 3. If wiring is acceptable, then replace the rudder limiter assembly.
	c) Mechanical blockage	1. Check that any foreign material such as carpet is not blocking the cam. 2. Check that solenoid shaft moves freely when activated. If it does not move freely, replace the rudder limiter assembly.
Rudder limiter annunciator fails to light with rudder limiter engaged	a) Light bulb burned out	1. Test the annunciator light using the annunciator test switch. If light does not operate, replace LED. (See Chapter 33)
	b) Proximity switch failure	1. With electrical power on, place magnetic screwdriver between proximity switch and rudder limiter cam. 2. If the annunciator fails to light, check the wiring and connections. 3. If they are correct, replace the rudder limiter assembly.

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27-9. AILERON SYSTEM – GENERAL

The ailerons are of a carbon fiber composite construction. The left aileron has a servo tab for reducing pilot forces during flight. The right aileron has an electrically actuated trim tab that is used for lateral trim during flight. Lead counterweights static balance the ailerons.

27-10. AILERONS

a. **Removal and Installation**

1. Disconnect the aileron final drive from the aileron drive horn by removing the cotter pin, castellated nut, and bolt.
2. Disconnect the grounding strap at the middle aileron hinge from the aileron by removing the AN526-832R8 screw.
3. On the right wing, disconnect the aileron trim servo wire, accessed through the outer wing-bay access panel.
4. On the left aileron, disconnect the servo tab drive rods from the wing aileron hinge by removing the cotter pins and clevis pins.
5. Remove the aileron from the wing aileron hinges by removing the cotter pin, castellated nut, and bolt at each of the three hinge locations.
6. Slide the aileron straight aft. Take care not to move the right aileron too far aft and break the wiring connector for the trim system.
7. Gently pull the wiring harness for the aileron servo from out of the wing, and disconnect the wiring connector.
8. Reinstall the ailerons in the reverse order.
9. After installing the ailerons, check that the deflections per Figure 27 - 1 are correct.

b. **Rigging and Adjustment**

The aileron control system is rigged at the factory and should not need any further adjustments. For new ailerons, or ailerons requiring significant readjustment in order to achieve the correct deflection limits, use the following procedure. For ailerons requiring only minor adjustment, use steps 1, 2, 3, 4, 9, 10, 11, 12, 13, and 14 of the following procedure.

NOTE

Keep the flaps in the fully-retracted position (0º) during the rigging process.

1. With the aileron final drives disconnected, verify smooth operation throughout the full range of travel for each aileron.
2. Place the flap/aileron deflection template, part number TA57270000, on the upper surface of the wing between the outboard end of the flaps and the inboard end of the aileron.
3. The downward deflection of an aileron should be approximately 17.7º. If the deflection of an aileron is not enough, carefully sand material from its down-stop to achieve this deflection. If the deflection is too great, contact the manufacturer for new aileron down-stops.
4. Level the aircraft about the roll axis before proceeding.
5. Disconnect the aileron crossover control rod in the center section of the wing from the aileron torque tubes. Set both control sticks in the cockpit to a position of 88º ± 2º as shown in Figure 27 - 15. Adjust the rod ends on the crossover tube so the assembly

may be installed between the torque tubes and tighten the jam nuts. Reinstall the torque tube.

6. With the aileron control rods disconnected, lock both the left-hand and right-hand aileron bellcranks in the neutral position within the bellcrank brackets by placing a 1/4 in. diameter pin through the rigging hole. An AN4 bolt works well for this.
7. With the control sticks still at 88º ± 2º, adjust the rod ends on the aileron control rods as required to allow them to fit between the aileron torque tubes and the bellcranks. Finger-tighten the control rod connections.
8. Clamp the ailerons in the neutral position (0º deflection on the rigging template). Adjust the length of the final drive to fit between the bellcrank and the aileron hinge and finger-tighten the connections.
9. Remove the pins from the bellcranks and release the control sticks. Unclamp the ailerons and rotate them until they reach the up-stop (which is controlled by the down-stop on the opposite aileron). The up-stop on one aileron and the down-stop on the opposite aileron should make contact at the same time while ensuring the trailing edge is inside the template up-travel window. Both right-hand and left-hand sides are set in this manner. If more travel is needed, a combination of up-stop adjustment and/or careful sanding of material from the down-stop on the opposite aileron may be required. Continue this procedure until both ailerons meet all of the deflection requirements of Figure 27 - 1.
10. Ensure that the gap at the outboard edge of the aileron between the wing and the aileron is 0.123 to 0.309 in.
11. Ensure that the gap at the inboard edge of the aileron between the flap and the aileron (when the flaps are retracted) is 0.125 to 0.375 in. Ensure minimum 0.125 in. clearance when the flaps are extended.
12. While the aileron is in neutral position ensure that the gap between the wing trailing edge and the aileron is 0.150 to 0.250 in. along the entire span of the aileron.
13. Verify there is sufficient thread engagement on all rod ends by checking that the threads are visible within the witness hole on each.
14. Once the correct aileron deflections and proper gaps are achieved, retighten all jam nuts and torque 12 to 15 ft.-lbs.

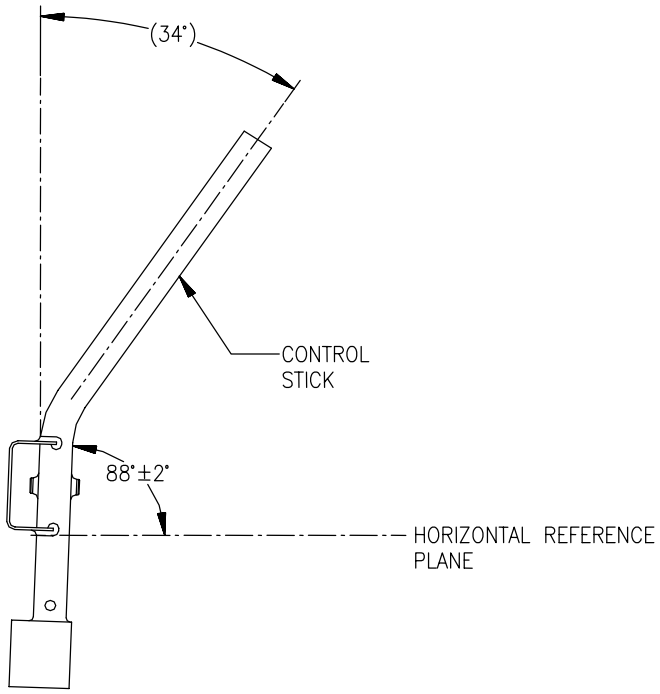

Figure 27 - 15 Control Stick Neutral Position for Aileron Rigging

c. **Service and Maintenance**

The ailerons do not require any service or maintenance. The ailerons should be visually inspected annually for cracking at all hinge attachment locations. A tap test should be conducted on the aileron at the outboard end cap to aid in detection of any latent damage, delaminations, or disbonds. The rod end bearings used throughout the aileron control system do not require maintenance other than cleaning during routine inspections to remove any accumulation of dust or grime. Liquid lubricants should not be used on the rod end bearings as this may cause the Teflon liner to swell, creating excess friction in the bearing. Should the rod end need to be replaced, please contact the manufacturer.

The aileron linear bearing should be inspected every 100 hours or annually. Inspect each side of the bearing for foreign object debris. Ensure there is no debris or adhesive attached to the control rod, bearing housing, bearing race, or ball bearings. Move the aileron controls through their full range of motion and ensure there is no scarring or damage to the control rod, especially in the vicinity of the linear bearing. Normal wear at the linear bearing is indicated by a slightly darker color on the control rod. If damage is noted contact the manufacturer.

The aileron control rods should be visually inspected every 500 hours or biennially. To remove the push-pull tubes for inspection:

1. Remove the aileron bellcrank access panel located on the lower surface of the wing directly in front of the inboard aileron hinge.
2. Disconnect the aileron push-pull tube from the aileron bellcrank.
3. Remove the aft wing belly access panel to gain access to the aileron crossover tube.
4. Disconnect the aileron push-pull tube from the aileron torque tube bellcrank.

5. Slide the push-pull tube out and down to clear the wing belly.
6. Reinstall the push-pull tubes in the reverse order.
7. Reinstall all access panels and check the aileron controls for binding and friction.

CAUTION

Never lubricate the bearings that support the aileron torque tube.

27-11. AILERON TRIM TAB

a. **Removal and Installation**

1. Remove the four MS24693-C28 countersunk screws from the aileron trim servo access panel located on the lower surface of the right aileron.
2. Disconnect the aileron trim tab drive rod and friction device clevis from the trim tab drive horn by removing the cotter pins and the clevis pins.
3. Remove the MS24665-151 cotter pins on either end of the piano hinge, and remove the hinge wire. Once the hinge wire is removed, the aileron trim tab may be removed.
4. Remove the four MS24693-C28 countersunk screws from the top surface of the right aileron.
5. Carefully remove the trim tab servo from within the aileron.
6. Once the trim motor is clear of the aileron, disconnect the wiring harness.
7. Reinstall the trim motor and trim tab in the reverse order. Torque all screws 10 to 12 in.-lbs.

b. **Rigging and Adjustment**

The aileron trim tab and servo are preset at the factory, and further adjustment should not be necessary. Should the trim tab servo motor need to be replaced, contact the manufacturer for a replacement part and for detailed rigging instructions. The trim tab deflections should be checked periodically to ensure that the trim system is functioning properly. Use the following procedure for checking the trim tab deflections.

1. Place the aileron/flap deflection template, part number TA57270000, on the upper wing surface between the outboard end of the flap and the inboard end of the aileron.
2. Set the aileron to the neutral, 0°, position.
3. Using a digital inclinometer or other calibrated angle-measuring device, check the deflections shown in Figure 27 - 16.

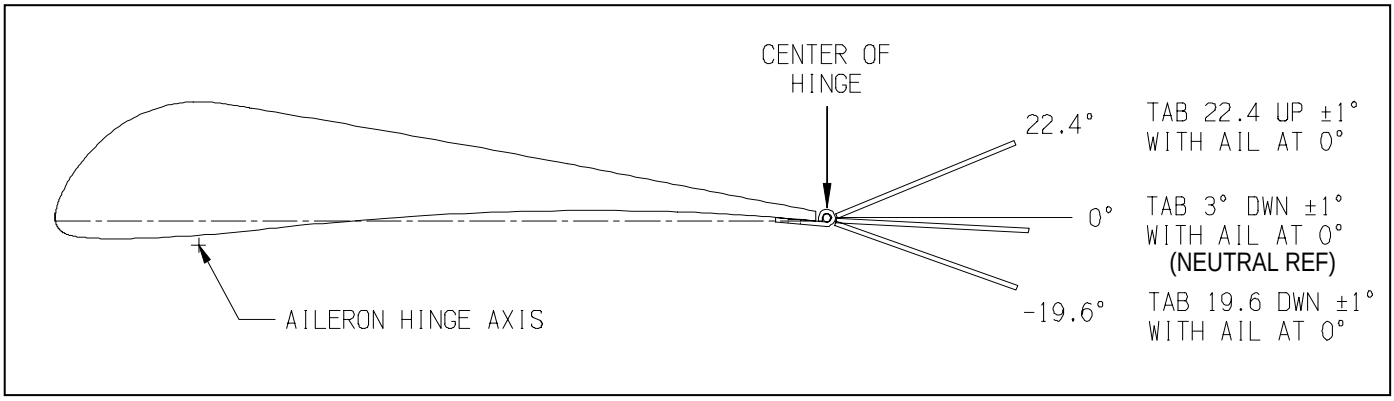

Figure 27 - 16 Aileron Trim Tab Deflection Angles

c. **Service and Maintenance**

The aileron trim tab is a flat aluminum plate. The trim tab does not require any maintenance. The trim tab hinge should be inspected annually for wear and corrosion. The hinge may be lubricated with MIL-L-7870 general purpose lubricating oil to help reduce wear and friction. The aileron trim tab friction device should be inspected every 500 hours, or biennially, and adjusted as necessary to 13.4 to 19.0 lbs. sliding friction. The friction device should be free from oils, grease, dirt, etc. If excessive wear is noted or smooth operation is not possible, the friction device should be replaced. A replacement, part number LA57273240, may be obtained from the manufacturer. The attachment bracket should be inspected every 500 hours or biennially for wear and replaced as necessary. The trim tab motor is a sealed unit and does not require service or maintenance.

27-12. AILERON TRIM TAB SERVO

a. **Removal and Installation**

1. Remove the four MS24693-C28 countersunk screws from the aileron trim servo access panel.
2. Disconnect the aileron trim tab drive rod and friction device clevis from the trim tab drive horn by removing the cotter pins and clevis pins.
3. Remove the four MS24693-C28 countersunk screws from the upper surface of the right aileron.
4. Carefully remove the trim tab motor from within the aileron.
5. Once the trim motor is clear of the aileron, disconnect the wiring harness.
6. Reinstall the trim tab motor in the reverse order. Torque all screws 10 to 12 in.-lbs.

b. **Rigging and Adjustment**

The trim tab motor is preset at the factory and should not need further adjustment. Should adjustment be necessary, use the following procedure.

1. Disconnect the friction device drive rod from the trim tab drive horn.
2. Extend the trim motor drive arm all the way out.
3. Set the trim tab at the up deflection limit, and adjust the drive rod as necessary to line up with the trim tab drive horn.
4. Retract the trim motor drive arm, and check that the trim tab deflection angle is correct. Adjust the clevis on the drive rod as necessary to achieve the correct deflection. Maintain proper thread engagement (threads must be visible through the inspection hole located in the clevis).
5. With the trim tab connected to the motor, retract the tab to the neutral position.
6. Adjust the friction device drive rod, and connect it to the trim tab drive horn.
7. Reinstall all clevis pins and cotter pins.
8. Recheck deflection angles, and repeat above procedure if necessary.

c. **Service and Maintenance**

The aileron trim tab servo is a sealed unit and does not require service or maintenance. Should the motor begin to slip or stop functioning properly, contact the manufacturer for a replacement.

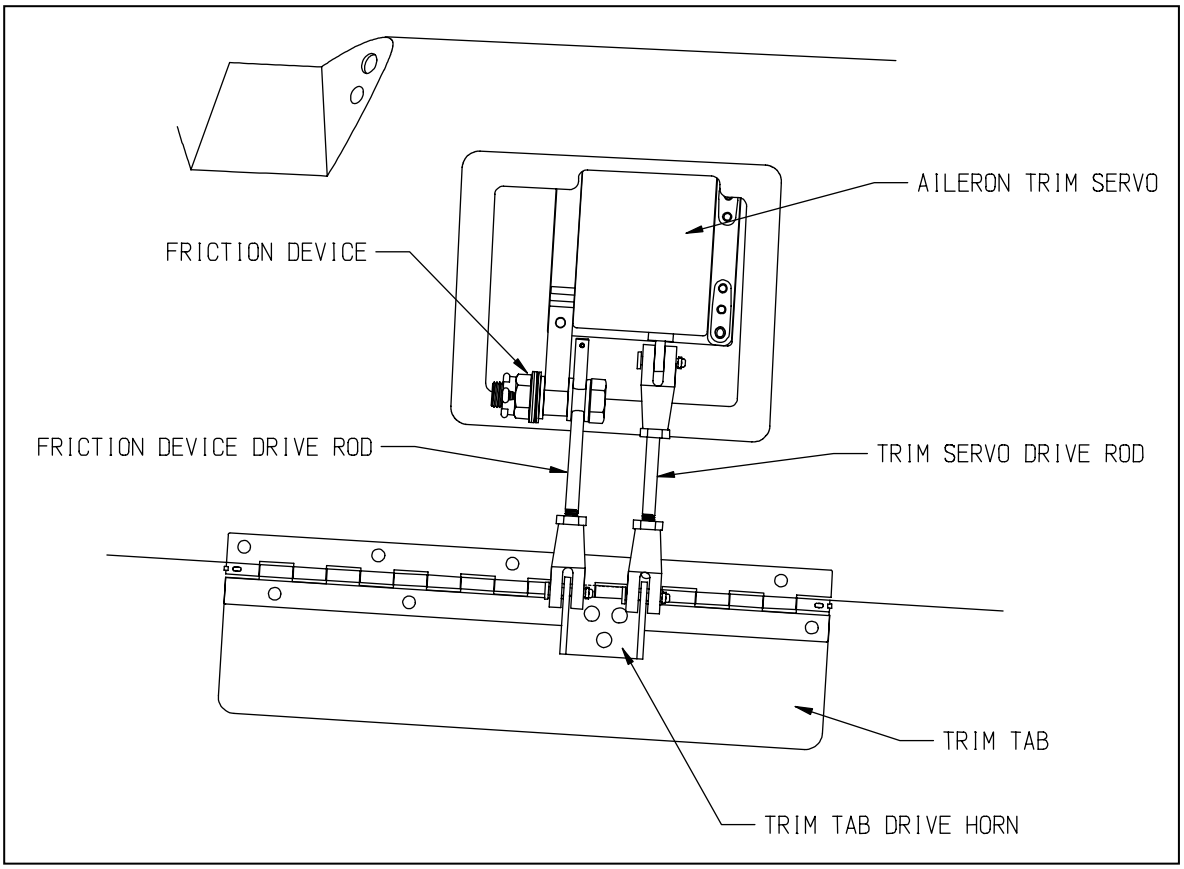

Figure 27 - 17 Aileron Trim System

27-13. AILERON SERVO TAB

a. **Removal and Installation**

1. Remove the cotter pins and clevis pins that attach the threaded drive rods to the servo tab bracket.
2. Remove the MS24665-151 cotter pins from both ends of the piano hinge.
3. Remove the hinge wire from the hinge, and remove the servo tab from the aileron.
4. Reinstall the servo tab in the reverse order. Apply zinc oxide to the threaded push rod where the clevis attaches. Recheck deflections.

b. **Rigging and Adjustment**

The aileron servo tab is preset at the factory and further adjustment should not be necessary. The servo tab deflections should be checked periodically to ensure that the system is functioning properly. Use the following procedure for checking the servo tab deflections.

1. Place the aileron/flap deflection template, part number TA57270000, on the upper wing surface between the outboard end of the flap and the inboard end of the aileron.
2. Set the aileron to the neutral, 0°, position.
3. Set the servo tab to the neutral, 2° trailing edge down, position (see Figure 27 - 18) by adjusting the drive rod clevises as necessary. Maintain proper thread engagement (threads must be visible through the inspection hole located in the clevis).

4. Cycle the aileron to the up and down deflection limits, and check the servo tab deflection angles as shown in Figure 27 - 18.
5. Adjust the drive rods as necessary to achieve the correct deflections. Maintain proper thread engagement (threads must be visible through the inspection hole located in the clevis). Torque all jam nuts 30 to 36 in.-lbs.

NOTE

Aileron servo tab deflection angles are measured with respect to a nominal horizontal reference plane.

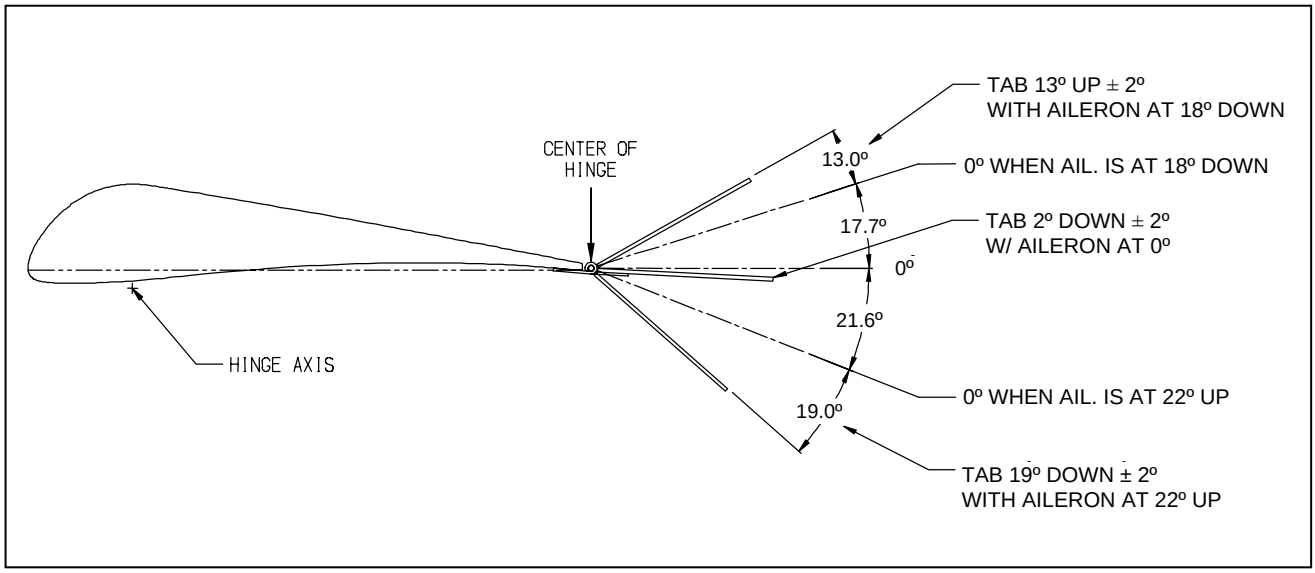

Figure 27 - 18 Aileron Servo Tab Deflection Angles

c. **Servicing and Maintenance**

The aileron servo tab does not require any service or maintenance. The attachment hinge should be inspected every 500 hours or biennially for wear and replaced if necessary. The hinge may be lubricated with MIL-L-7870 general purpose lubricating oil to help reduce friction and wear.

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27-14. RUDDER SYSTEM – GENERAL

The rudder attaches to the aft vertical closeout rib at three hinge points. Rudder cables attached to the rudder drive horn link the rudder to the rudder pedals. A fixed rudder trim tab is attached to the rudder at approximately mid span. The trim tab is ground adjustable for pilot comfort during cruise or takeoff power settings. A lead counterweight is used to static balance the rudder.

27-15. RUDDER

a. **Removal and Installation**

1. Detach the two rudder cables from the drive horn by removing the cotter pin from each nut, and then removing the bolt.
2. Remove the cotter pin from the lower hinge, and remove the castellated nut.
3. Remove the rudder by lifting it up and aft. Take care not to pull too far aft and break the metal braid that is still attached to the rudder leading edge.
4. Once the rudder is removed, detach the metal braid by removing the attachment screw.
5. Install the rudder in the reverse order. A maximum of one NAS1149F0432P washer may be used to ensure there is a minimum clearance of 0.062 to 0.125 inches between the rudder actuator and the rudder attachment assembly (see Figure 27 - 19).
6. Recheck all attachments for security and safety.

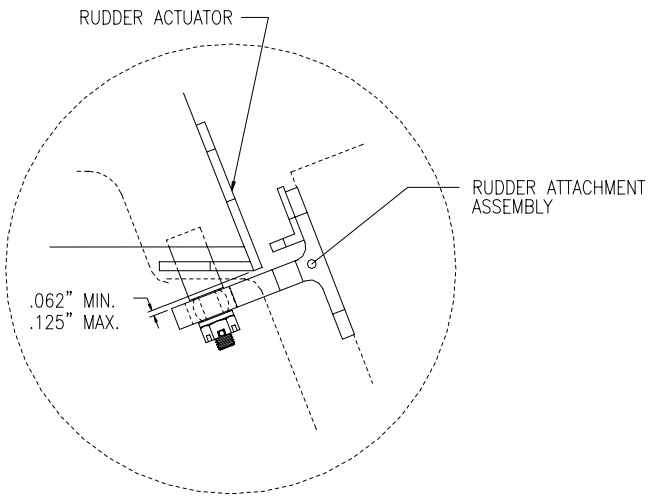

Figure 27 - 19 Rudder Actuator Clearance

b. **Rigging and Adjustment**

The rudder deflections are preset at the factory and should not need any further adjustment. If the rudder cable is replaced, the deflections will need to be rechecked and adjusted as necessary. Should this need to be done, use the rigging procedure as outlined below.

NOTE

Rigging of the rudder will be easier with at least two individuals: one to hold the template in the proper position and one to make the required adjustments.

1. Remove the locking clips from the left and right turnbuckles at the forward end of the rudder cable near the rudder pedals. Remove the locking clips from the rudder pedal interconnect turnbuckle (located behind the instrument panel near the firewall), and loosen the interconnect cable.
2. Ensure that the rudder cables are not attached to the rudder pedal crossover arms.
3. Thread the left rudder cable, right rudder cable and rudder pedal interconnect turnbuckles so that the clevis threads are just visible. Apply three turns to each clevis to put the threads inside the turnbuckle.
4. Clamp the elevator in the neutral position such that the trailing edge is in alignment with the trailing edge of the horizontal stabilizer.
5. Adjust the rudder to the neutral position by moving it left and right until it is centered about the vertical tail.
6. Position the rudder deflection template, part number TA53550000, on the rudder such that the radius portion is located at the bottom edge of the rudder trim tab and the middle portion is aligned with the leading edge of the rudder. This action will position the template perpendicular to the hinge axis of the rudder.
7. Move the rudder to the left and right until it contacts the rudder stop.
8. Confirm that the deflection limits are 17° ± 1°. If the deflection limits are greater than the allowed tolerance, shim the rudder stop with thin washers as necessary to achieve the correct deflection. If the limits are less than the allowed tolerance, grind on the angled face of the stop until the desired angle is achieved.

NOTE

If grinding on the rudder stop is required to achieve the required rudder deflection, apply zinc oxide primer or a similar corrosion protecting treatment to the exposed metal.

9. Attach the rudder cables to the rudder pedal crossover rods.
10. Align the rudder to the 17° ± 1° position on the right and push the pilot's side right rudder pedal to its full forward position.
11. Two methods, as follows, may be used for attaching the rudder cables to the rudder drive horn. The first method employs a cable thimble, compression sleeve, and shackle. The second method utilizes a fork end cable terminal. Whichever method is used, contact Cessna for parts and instructions.
    a) Cable Thimble, Compression Sleeve and Shackle (See Figure 27 - 20)
        1) At the aft of the plane pull the left side rudder cable taut and mark the cable, with an indelible marker, where it intersects with the forward edge of the connection hole in the rudder drive horn.
        2) Using a cable thimble, clevis, and temporary clamp form a loop at the end of the rudder cable with the indelible mark at the apex of the loop. DO NOT remove excess cable.
        3) Align the rudder to the 17° ± 1° position on the left and push the pilot's side left rudder pedal to its full forward position.
        4) At the aft of the plane pull the right side rudder cable taut and mark the cable, with an indelible marker, where it intersects with the forward edge of the connection hole in the rudder drive horn.

5) Using a cable thimble, clevis, and temporary clamp form a loop at the end of the rudder cable with the indelible mark at the apex of the loop. DO NOT remove excess cable.
6) Temporarily connect the rudder drive cables to the rudder drive horn. Insert excess cable into the fuselage. Ensure that the temporary clamp does not catch on the fuselage as the rudder deflections are adjusted.
7) Align the rudder to the 11.5° ± 0.5° position on the template.
8) On the pilot's side, pull the left rudder pedal to its full aft position.
9) Engage the rudder limiter, and depress the left rudder pedal until the connecting arm contacts the rudder limiter cam.
10) Place an adjustable rod or similar device between the left rudder pedal and the vertical face of the floor directly beneath the pilot's seat. Adjust the rod so as to maintain enough pressure on the left rudder pedal to keep the rudder limiter cam in the engaged position.
11) Adjust the left rudder cable turnbuckle until the rudder cable is tight.
12) Unblock the rudder pedal and the rudder. Set the rudder to 0° deflection.
13) Adjust the right side rudder cable turnbuckle until the right rudder pedal is aligned with the left rudder pedal.
14) With the rudder at 0° deflection, adjust the rudder pedal interconnect turnbuckle until it takes a weight of 5 lbs. to deflect the cable 1.0 in. ± 0.25 in. See Chapter 22 for cable tension if autopilot is installed.
15) Recheck all deflections, and repeat the rigging procedure as necessary until the proper deflections are achieved.
16) Without disturbing the turnbuckles, disconnect the rudder cables from the rudder pedal crossover rods.
17) Disconnect the rudder cables from the rudder drive horn.
18) Using an indelible marker, make three marks on the rudder cable and cable thimble to facilitate alignment of the cable onto the thimble after the temporary clamp has been removed.
19) Remove the temporary clamp.
20) Insert the cable end into a compression fitting and form a loop.
21) Align the cable onto the thimble using the three indelible marks as a guide.
22) Snug the compression fitting up against the thimble and compress the fitting.
23) Reattach the rudder cables to the rudder horn and to the pedal crossover rods and recheck all deflections. Repeat the rigging procedure as necessary until the proper deflections are achieved.
24) Cut off excess rudder cable at the compression fitting.

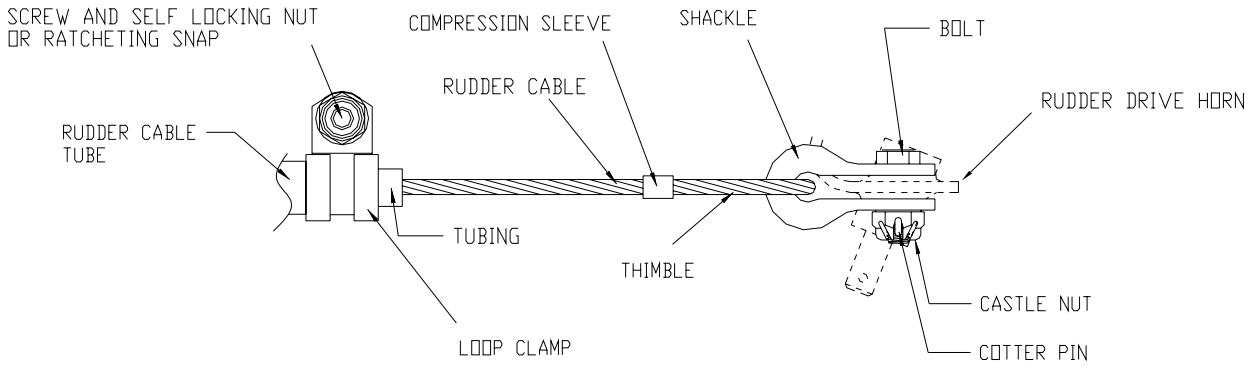

Figure 27 - 20 Cable Thimble, Compression Sleeve and Shackle

b) Fork End Cable Terminal (See Figure 27 - 21)
   1) Temporarily attach fork end cable terminals to the left and right side of the rudder drive horn.
   2) At the aft of the plane insert the rudder cable into the left fork end cable terminal and pull it taut. Mark the cable, with an indelible marker, where it intersects with the inner edge of the terminal sleeve just behind the fork of the terminal.
   3) Align the rudder to the 17° ± 1° position on the left and push the pilot's side left rudder pedal to its full forward position.
   4) At the aft of the plane insert the rudder cable into the right fork end cable terminal and pull it taut. Mark the cable, with an indelible marker, where it intersects with the inner edge of the terminal sleeve just behind the fork of the terminal.
   5) Disconnect the rudder cables from the pedal crossover rods.
   6) Remove the terminals and cut the rudder cables 1/8 inch behind the mark toward the fore of the aircraft.
   7) Insert the rudder cable into the left cable terminal so that the cable is 1/8 inch inside the terminal sleeve at the fork end.
   8) Swage the fork end cable terminal.

NOTE

The swaging tool required to swage the fork end cable terminal is a special tool not available in most repair shops. Contact Cessna for instructions.

   9) Repeat steps 7 and 8 for the right cable terminal.
   10) Reattach the rudder cables to the rudder horn and to the pedal crossover rods.
   11) Align the rudder to the 11.5° ± 0.5° position on the template.
   12) On the pilot's side, pull the left rudder pedal to its full aft position.
   13) Engage the rudder limiter, and depress the left rudder pedal until the connecting arm contacts the rudder limiter cam.
   14) Place an adjustable rod or similar device between the left rudder pedal and the vertical face of the floor directly beneath the pilot's seat. Adjust the rod

so as to maintain enough pressure on the left rudder pedal to keep the rudder limiter cam in the engaged position.

15) Adjust the left rudder cable turnbuckle until the rudder cable is tight.
16) Unblock the rudder pedal and the rudder. Set the rudder to 0° deflection.
17) Adjust the right side rudder cable turnbuckle until the right rudder pedal is aligned with the left rudder pedal.
18) With the rudder at 0° deflection, adjust the rudder pedal interconnect turnbuckle until it takes a weight of 5 lbs. to deflect the cable 1.0 in. ± 0.25 in. See Chapter 22 for cable tension if autopilot is installed.
19) Recheck all deflections, and repeat the rigging procedure as necessary until the proper deflections are achieved.

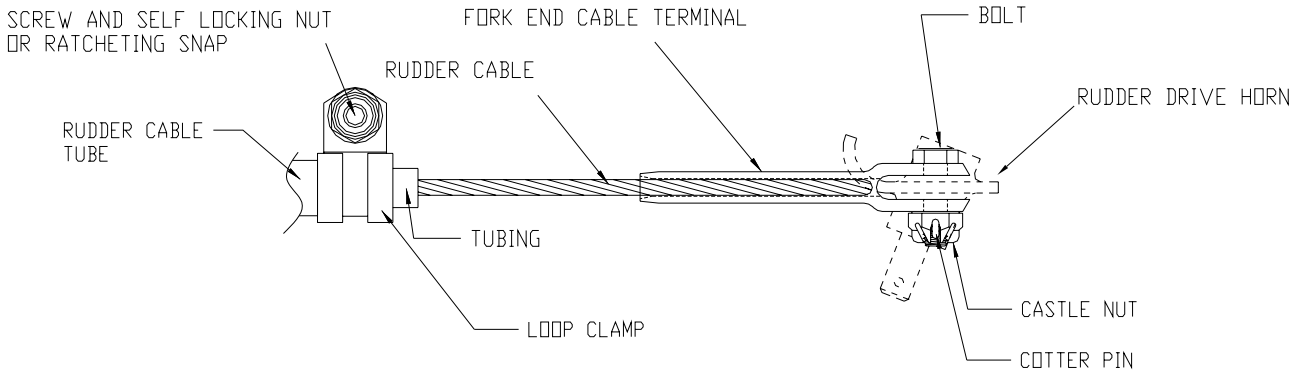

Figure 27 - 21 Fork End Cable Terminal

12. Reinstall the locking clips into all turnbuckles.
13. Cycle the rudder from left to right to ensure that there is no binding or excessive friction in the system.

c. **Service and Maintenance**

The rudder does not require any service or maintenance. However, due to its composite construction, the rudder should be visually inspected annually for cracking at all hinge attachment locations and the trim tab attachment location. A tap test should be conducted on the rudder at the drive horn location to aid in detection of any latent damage or disbonds along the lower rudder rib. The rudder does not have any access panels, therefore, the counterweight needs to be removed in order to inspect the internal structure. To remove the rudder counterweight, remove the attachment screws, and slide the counterweight straight off. The rudder may also be inspected via the holes at the middle and top rudder hinge pin locations. Common inspection tools that may be used or needed are mirrors, bore scopes, etc.

The rudder should be removed and inspected for wear of the hinge pins. Spherical bearings are used in the attachment brackets on the vertical closeout rib. These bearings do not require any maintenance. The hinge brackets should be inspected for corrosion or other damage and replaced if necessary.

d. **Rudder Cable Removal**

The rudder cable should be inspected thoroughly and replaced if any excessive wear or damage is observed. To remove the rudder cable use the following steps.

1. Contact the manufacturer for a replacement.

2. Detach the rudder cable from the rudder drive horn per the rudder installation and removal instructions in this section.
3. Cut the rudder cable to remove the turnbuckle, and slide the cable aft through the aft end of the fuselage.
4. Install the new rudder cable by inserting the end without turnbuckle into the fore end of the rudder cable tube and sliding it aft to exit at the rudder drive horn.
5. Follow the procedures in 27-16.b for connection, rigging, and adjustment. Ensure that the requirements listed in Figure 27 - 1 are met.

27-16. RUDDER TRIM TAB

a. **Removal and Installation**

Should the trim tab need to be replaced, carefully drill out the five CCR264CS-4-02 attachment rivets to remove the trim tab. Obtain a new trim tab, part number LA53550001, from the manufacturer and reinstall with five new rivets of the same type and size. Take care when drilling out rivets so as not to enlarge holes.

CAUTION

Due to the carbon fiber composite construction of the rudder, do not use aluminum rivets. The new rivets must be of a corrosion resistant material (i.e. stainless steel) in order to prevent galvanic corrosion.

b. **Adjustment**

The rudder trim tab is a flat aluminum plate that is riveted to the trailing edge of the rudder at approximately mid-span. The trim tab is attached with five CCR264CS-4-02 countersunk rivets. The trim tab may be adjusted for pilot comfort during cruise or takeoff power settings. The trim tab is not adjustable during flight and must be adjusted while on the ground. To adjust the rudder trim tab use the following instructions.

1. Clamp two pieces of wood or other rigid material to either side of the trim tab. The wood blocks should be equal in length to the trim tab.
2. Align the edges of the wood block with the slots in the trim tab.
3. Once the wood blocks are positioned correctly, simply bend the trim tab in the desired direction. Do not to exceed the deflection angles described in Figure 27 - 1.

c. **Service and Maintenance**

The rudder trim tab should be inspected for wear and cracking of the slots that run the length of the tab (bend line). If excessive wear or cracking is observed, the trim tab should be replaced.

27-17. ELEVATOR SYSTEM – GENERAL

The elevator is of a carbon fiber composite construction and is attached to the horizontal stabilizer with two piano hinges. A series of push-pull tubes connects the elevator to the left and right control sticks. Simple rod end bearings are used throughout the elevator control system. Lead counterweights balance the elevator.

CAUTION

Due to its size and weight, removal of the elevator may require more than one individual.

27-18. ELEVATOR

a. **Removal and Installation**

1. Remove the rudder per the instructions in section 27-15.
2. Remove the cotter pin and retaining hardware from the aft elevator push rod and the elevator drive horn.
3. Remove the 36 MS24694-C50 screws that attach the elevator hinge to the horizontal stabilizer.
4. Remove the elevator by sliding it straight aft. Take care not to move it too far aft and break the wiring connector.
5. Disconnect the wiring connector for the elevator trim servo and the grounding strap at the elevator horn, and remove the elevator completely from the aircraft.
6. Install the elevator in the reverse order. Recheck all attachment hardware for security and safety. Torque all attachment screws 30 to 36 in.-lbs.

b. **Rigging and Adjustment**

1. Remove the following interior panels per the instructions in Chapter 25: FLSPL, FLSPR, MLSPL, MLSPR, right and left door tread.
2. Remove the baggage bulkhead access panel per the instructions in Chapter 53.
3. With the elevator securely fastened to the horizontal stabilizer, clamp the elevator such that its trailing edge is aligned with the trailing edge of the horizontal stabilizer.
4. Set the control stick to the vertical, 0°, position as shown in Figure 27 - 22.
5. Loosen the jam nuts on the rod end bearings on all elevator push-pull tubes as shown in Figure 27 - 23.
6. Place the elevator deflection template, part number TA55270000, on the horizontal stabilizer 76.5 inches inboard from the tip of the horizontal stabilizer and perpendicular to the hinge axis. This template has three slots that define the neutral, upper, and lower elevator deflection locations. The middle slot defines the neutral position, while the upper and lower slots define the upper and lower deflections limits, respectively. See Figure 27 - 24.
7. With the elevator in the neutral position (trailing edge in alignment with the middle slot on the template) rotate the elevator upwards until it just contacts the stop. Check that the trailing edge of the elevator is between the upper and lower edges of the uppermost slot on the template.

CAUTION

Do not press too hard on the elevator against the elevator up stop, as this will cause a false indication of the elevator deflection.

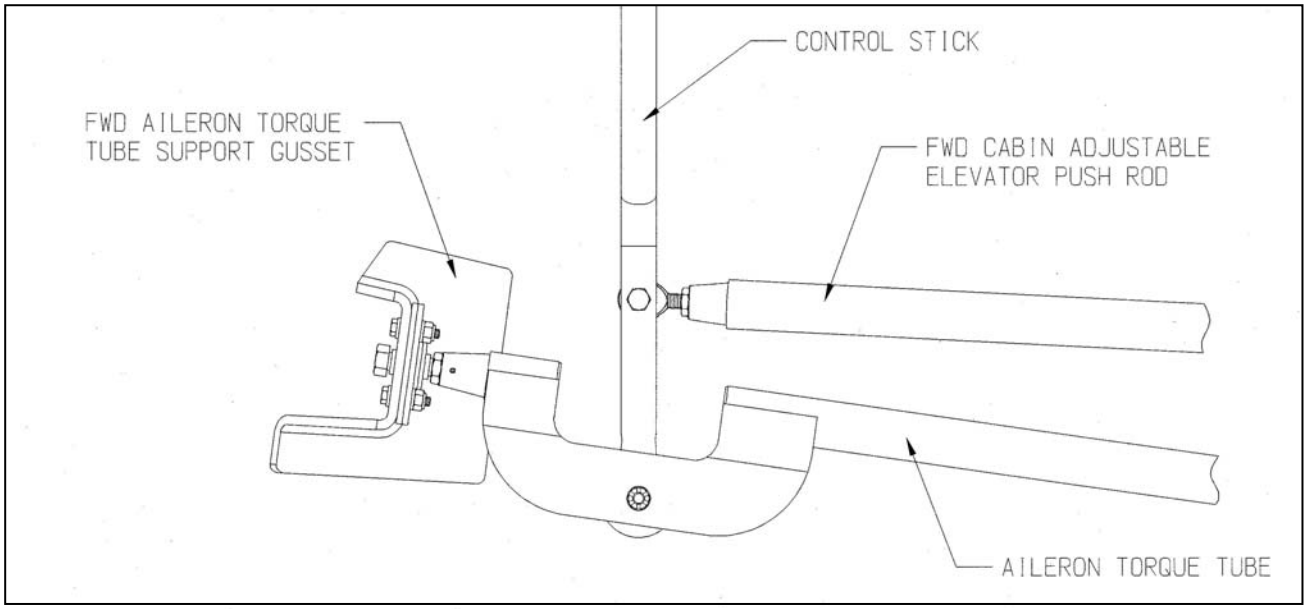

Figure 27 - 22 Control Stick Neutral Position

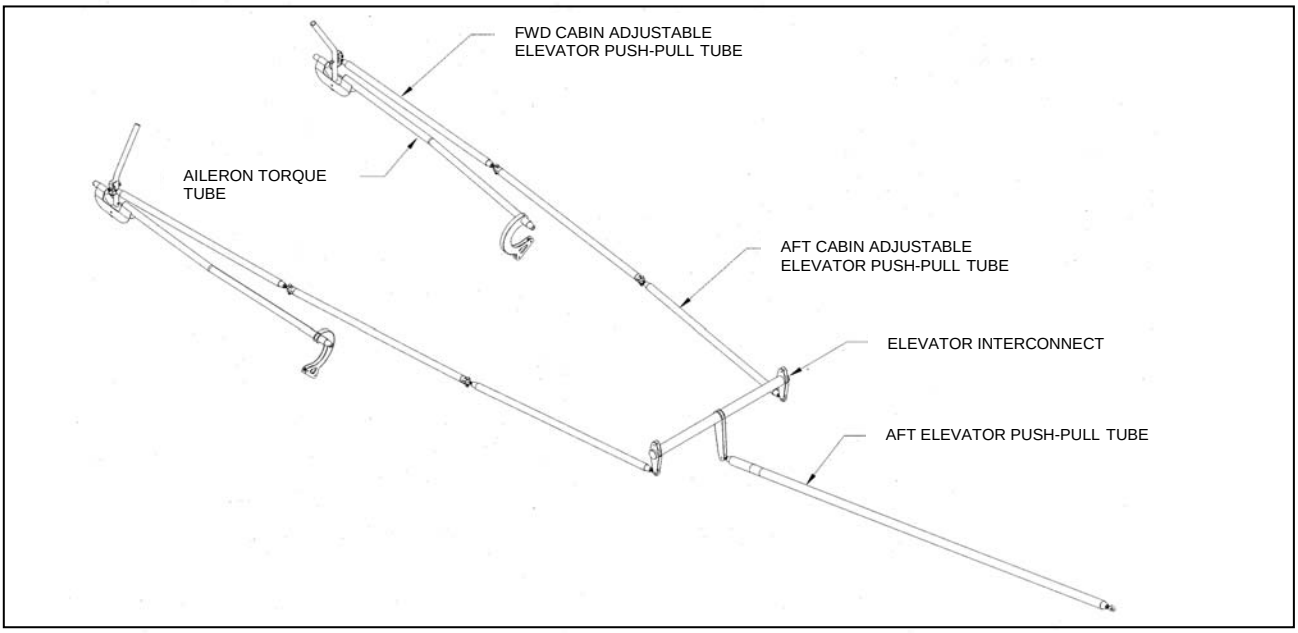

Figure 27 - 23 Fuselage Control System Layout

8. Rotate the elevator to the down position, and allow the elevator to rest freely against the down stops. Check that the trailing edge is between the upper and lower edges of the lower slot on the template.

9. Adjust the elevator up and down stops as necessary to align the trailing edge of the elevator between the upper and lower edges of the top and bottom slots in the template.
   a) To adjust the elevator up stop, loosen the jam nut, turn the socket head cap screw in the desired direction until the proper up deflection is met, then re-tighten the jam nut and torque 100 in.-lbs.
   b) To adjust the elevator down stops and increase the amount of down deflection, sand on the surface of the stop located in the counterweight pockets until the proper down deflection is achieved. If the down deflection is too much, contact the manufacturer for new elevator stops.

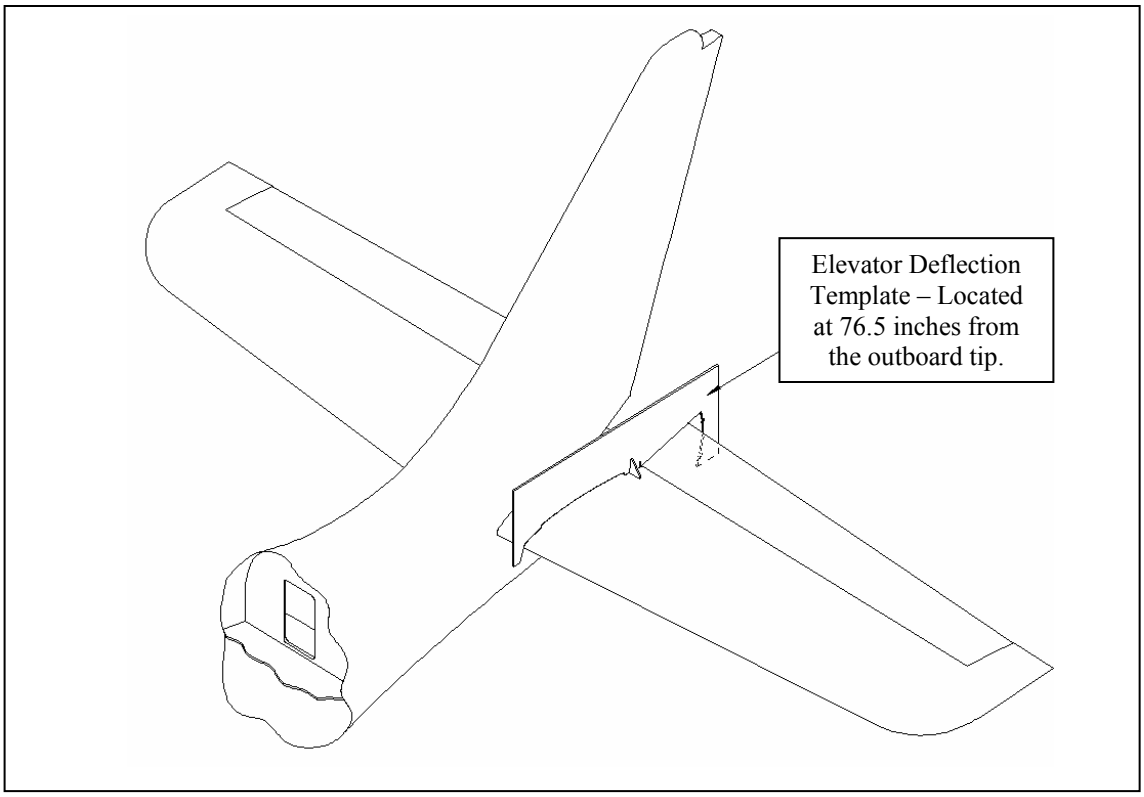

Figure 27 - 24 Elevator Deflection Template

10. Adjust the rod end bearings in the elevator push-pull tubes such that the following criteria are met:
    a) The elevator deflections per Figure 27 - 1 are achieved.
    b) The control stick is in a vertical, 0°, position when the elevator is at neutral, 0°, deflection, see Figure 27 - 22.
    c) The elevator interconnect weldment has adequate clearance (0.125 in. minimum) with the adjacent aircraft structure.
    d) Control stick is free from interference with adjacent side interior panels and the instrument panel.

e) All rod ends have a minimum 0.5 in. thread engagement (threads should be visible through inspection hole in ends of push-pull tubes).

11. Ensure that the gap between the outboard end of the elevator and the horizontal stabilizer is 0.125 to 0.250 in.
12. Torque all AN316-6 jam nuts at control rod ends to 95 to 110 in.-lbs. and all AN316-5 jam nuts to 70 to 85 in.-lbs.
13. Cycle the control stick fore and aft to ensure that there is no binding in the system.

b. **Servicing and Maintenance**

The elevator is attached to the horizontal stabilizer with two piano hinges. Each hinge should be inspected annually for excessive wear and cracks. The hinge may be lubricated with MIL-L-7870 general purpose lubricating oil to help reduce wear and friction. The rod end bearings used throughout the elevator control system do not require maintenance other than cleaning during routine inspections to remove any accumulation of dust or grime. Liquid lubricants should not be used on the rod end bearings as this may cause the Teflon liner to swell creating excess friction in the bearing. Should the rod end need to be replaced, contact the manufacturer.

27-19. ELEVATOR TRIM TAB

a. **Removal and Installation**

1. Gain access to the elevator trim motor by removing the access panel located on the lower surface of the elevator.
2. Disconnect the trim motor drive rods from the trim tab drive horn by removing the cotter pins and clevis pins.
3. Rotate the trim tab upwards to gain access to the screws attaching the trim tab hinge to the elevator.
4. Remove the 11 AN526C632R screws attaching the hinge to the elevator.
5. Reinstall the trim tab in the reverse order, and torque all screws 10 to 12 in.-lbs.
6. Run the trim tab to the up and down deflection positions to ensure that there is no binding of the system.

b. **Rigging and Adjustment**

The elevator trim tab deflections are preset at the factory and should not require any further adjustment. The trim tab deflections should be checked periodically to ensure that the system is functioning properly. If the trim tab deflections are checked and found to be outside the acceptable limits per Figure 27 - 1, contact the manufacturer for further instructions.

c. **Servicing and Maintenance**

The trim tab is attached to the elevator with one piano hinge. The hinge should be visually inspected annually for excess wear and cracking and replaced as necessary. The hinge should be replaced if the hinge becomes worn to the point that the trim tab free play limits per Figure 27 - 1 are exceeded. The hinge may be lubricated with MIL-L-7870 general purpose lubricating oil. Should the nutplates located on the nutplate assembly, part number LA55273818, inside the elevator become worn or stripped, contact the manufacturer for a replacement.

27-20. ELEVATOR TRIM TAB SERVO

a. **Removal and Installation**

1. Gain access to the elevator trim motor by removing the access panel located on the lower surface of the elevator.
2. Disconnect the trim motor drive rods from the trim tab drive horn by removing the cotter pins and clevis pins.
3. Remove the six AN526C632R10 screws that attach the trim motor hinge to the elevator hinge.
4. Carefully remove the trim motor from within the elevator.

CAUTION

Take care when removing the trim motor so as not to damage the wiring and connectors located on the outside of the trim motor housing. Should any of the wiring or connectors become damaged or break, contact the manufacturer for a replacement.

5. After removing the trim motor, disconnect the wiring harness.
6. Reinstall the trim motor in the reverse order. Torque all screws 10 to 12 in.-lbs.

b. **Rigging and Adjustment**

The elevator trim motor is an integral part of the elevator trim tab. The trim tab deflections are preset at the factory, therefore, the trim tab and motor should not require any further adjustment unless it needs to be removed and replaced. If the trim tab deflections are checked and found to be outside the acceptable limits per Figure 27 - 1, the following procedure should be used.

1. Using elevator trim template TA55273800 Rev B, measure out from the right side of the vertical 23.875 in. to BL 26.175. Carefully place the template on the elevator and over the trim tab. Have an assistant drive the elevator trim tab to the full down position (trim tab up) as indicated by the template.

NOTE

Actuation of the elevator trim system may have to be stopped before the `DOWN' microswitch is activated. If the microswitch is activated before full down travel is achieved, back-off the screw on the potentiometer drive bracket so that the full throw of the trim tab can be accomplished.

2. With the trim tab in the full down position (trim tab up), adjust the down screw on the potentiometer drive bracket (see Figure 27 - 25) so that the screw just closes the microswitch, deactivating the motor. Tighten the jam nut on the adjusting screw.
3. Using trim tab template TA55273800 Rev B as a reference, have an assistant drive the elevator trim system to the full up position (trim tab down).

NOTE

Actuation of the elevator trim system may have to be stopped before the `UP' microswitch is activated. If the microswitch is activated before full up travel is achieved, back-off the up-screw on the potentiometer drive bracket so that the trim tab can be moved to match the required template markings.

4. With the trim tab in the full up position (trim tab down) adjust the up screw on the potentiometer drive bracket so that the screw just closes the microswitch deactivating the motor. Tighten the jam nut on the adjusting screw.
5. Drive the elevator trim system to the full down position using the control stick trim button while an assistant watches the actuator and trim tab linkage for clearances and smooth operation.

CAUTION

If the microswitch is severely misadjusted, damage to the aircraft and/or components may occur while trimming to either stop. Watch carefully during this phase of operation for clearances and any indication of binding.

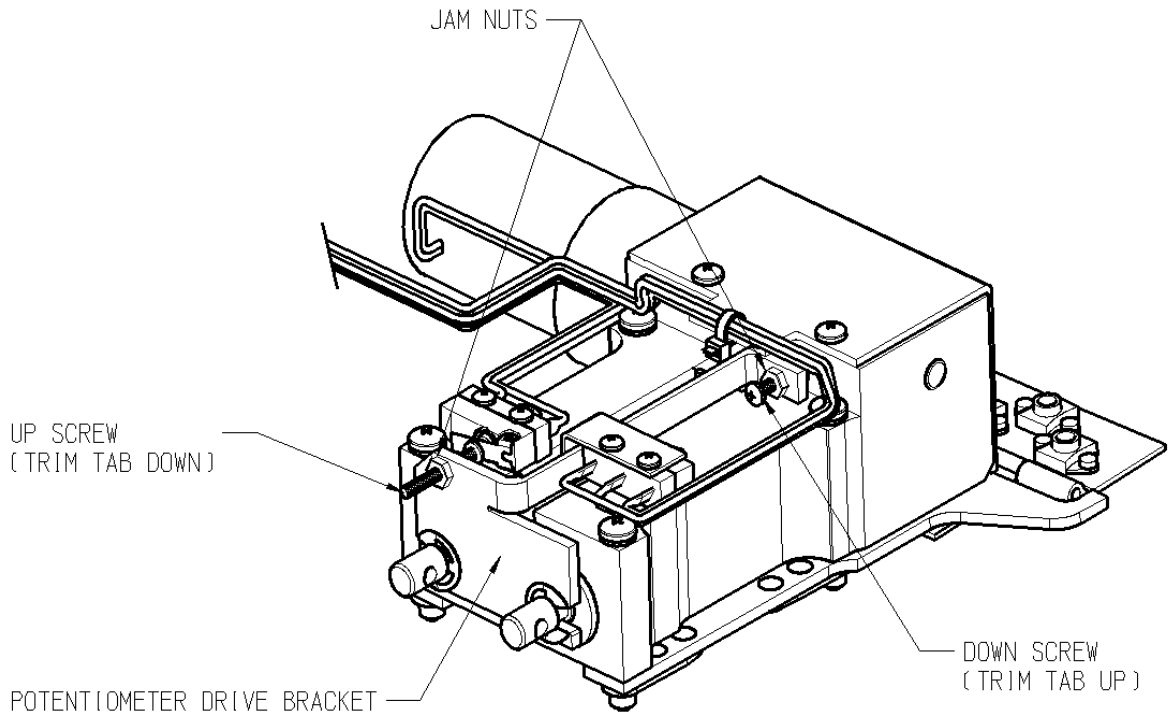

Figure 27 - 25 Elevator Trim Actuator

6. Using template TA55273800 Rev B, check position of the trim tab. Repeat Step 2 if necessary. Check trim panel position indications.
7. Drive the elevator trim system to the full up position while an assistant checks actuator and trim tab linkage for clearances and smooth operation.

CAUTION

If the microswitch is severely misadjusted, damage to the aircraft and/or components may occur while trimming to either stop. Watch carefully during this phase of operation for clearances and any indication of binding.

8. Using template TA55273800 Rev B, check position of trim tab. Repeat Step 6 if necessary. Check trim panel position indication.
9. Cycle the elevator trim system through full deflections, checking for security, proper travel, and indications.

NOTE

When the trim tab is in the neutral position (faired evenly with elevator), the trim panel should indicate one green light above center.

10. Install the elevator trim actuator cover panel.

c. **Servicing and Maintenance**

The elevator trim motor should be removed for inspection and service every 3000 hours. The trim motor drive rod gears should be inspected for wear and lubricated with heavy-duty gear grease. The worm gears attached to the drive motor shaft do not need to be lubricated but should be inspected for wear. If any of the gears are worn, they should be replaced. Replacements may be obtained from the manufacturer.

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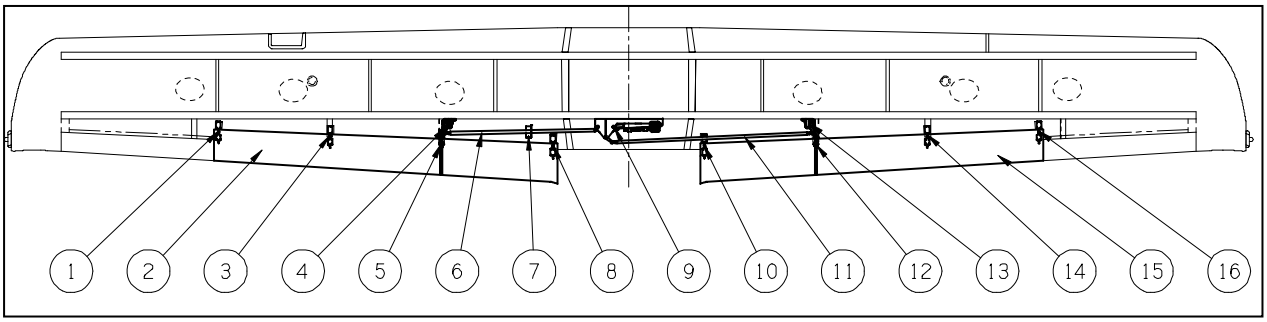
27-21. FLAP SYSTEM - GENERAL

a. The flap system is located in the wing, aft of the rear spar, with the exception of the elctrical system linking the flap switch to the electric flap actuator. The system uses four control rods with rod ends mounted on each threaded control rod. The rod ends use a teflon liner which requires no lubrication.



No.	ITEM	No.	ITEM
1	WS 143.5 flap hinge assembly	9	Flap central bellcrank and actuator assembly
2	Left flap control surface	10	WS 26.5 flap hinge assembly
3	WS 104.5 flap hinge assembly	11	Right flap control rod
4	Flap drive bellcrank and final drive – left wing	12	WS 65.5 flap hinge assembly (drive hinge)
5	WS 65.5 flap hinge assembly (drive hinge)	13	Flap drive bellcrank and final drive – right wing
6	Left flap control rod	14	WS 104.5 flap hinge assembly
7	Left flap control rod guide	15	Right flap control surface
8	WS 26.5 flap hinge assembly	16	WS 144 flap hinge assembly

Figure 27 - 26 Flap System Overview

b. Primary flap stops are electric provided by the limit switch assembly. The flap system secondary stops are mechanical with up stops provided by the flap hinges and the flap central bellcrank assembly. Down stops are provided by the flap actuator. Refer to Chapter 33 for additional information regarding the electrical connections and flap switch.

27-22. FLAPS

a. **Removal and Installation**

1. Lower flaps to the landing position.
2. Remove the cotter pin, castellated nut, bolt, and washers from the flap drive control rod to the flap. Note that small washers may be installed between the flap drive hinge and the control rod end. These washers are used to increase the misalignment capability of the rod ends, and each aircraft may have a different number of washers, or none at all.
3. Remove the cotter pins, castellated nuts, washers, and bolts from the flap hinge pivots.
4. Remove the flap by pulling the flap aft and away from the wing.

5. Reinstall in the reverse order.
6. Torque all castellated nuts 72 to 84 in.-lbs., and install cotter pins.

b. **Rigging and Adjustment**

Once the flap system is properly rigged, only minor adjustments are required unless components such as control rods, actuator, or limit switches are replaced. It is important to start the rigging with the flap actuator and the flap central bellcrank. To rig the flap system, use the following steps.

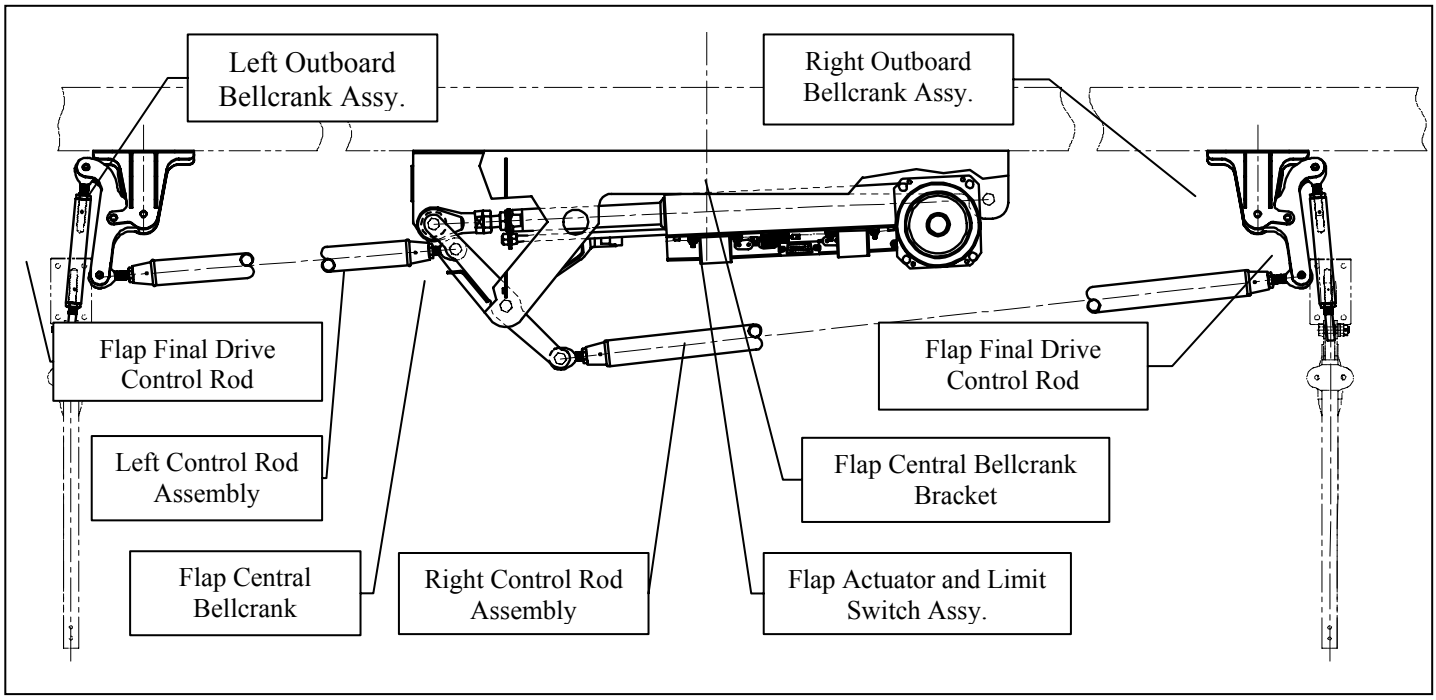


Figure 27 - 27 Flap Rigging Overview

1. With the flap control rods disconnected from the outboard bellcranks, extend the actuator (flap switch up/cruise position).
2. By adjusting the limit switch rod (see Figure 27 - 29) and the actuator rod end if necessary, adjust the flap up position so a 0.063 in. spacer fits between the flap central bellcrank and the web on the central bellcrank bracket as shown in Figure 27 - 28. Adjustments can easily be made with the actuator retracted (landing position). Torque the limit switch jam nuts 72 to 84 in.-lbs. and the flap actuator rod end jam nut 30 to 36 ft.-lbs. A minimum of $1^{1}/_{2}$ diameter thread engagement is required on the actuator rod end.

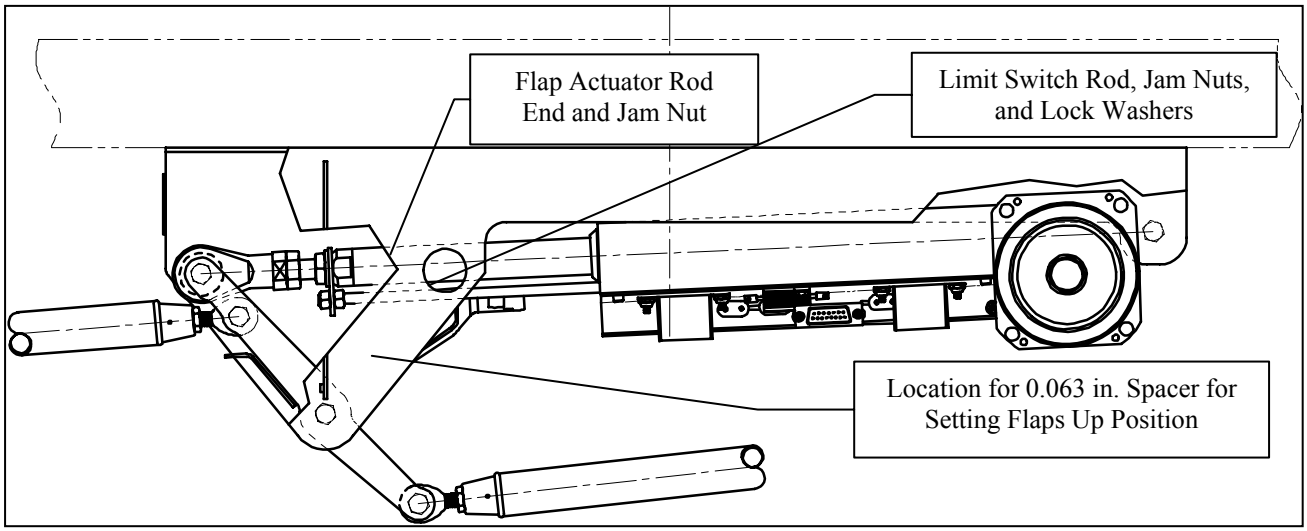


Figure 27 - 28 Flap Central Bellcrank and Flap Actuator

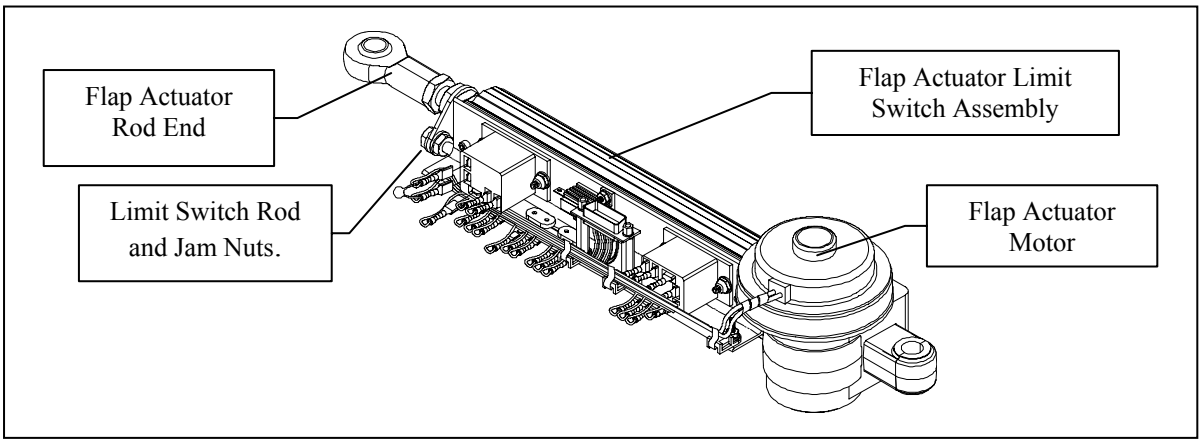


Figure 27 - 29 Flap Actuator Detail – Top View

3. Pin the outboard bellcranks with a 0.25 in. diameter bolt or equivalent, and adjust the flap control rods with the actuator set in the up position (see Figure 27 - 29).
4. Adjust the left and right control rod assemblies to length between the flap central bellcrank in the up position and the pinned outboard bellcranks. After adjusting the control rod assemblies, torque the jam nuts 140 to 180 in.-lbs., and verify that the minimum thread engagement is achieved by checking that the threads extend past the inspection hole in the control rods. Torque bolts through the bellcrank and control rods 72 to 84 in.-lbs., and install the cotter pin in the castellated nut (see Figure 27 - 29). Note the use of washers between the bellcrank and both sides of the rod end. If replacement of these washers is necessary, they may be replaced with either AN960-416 washers.

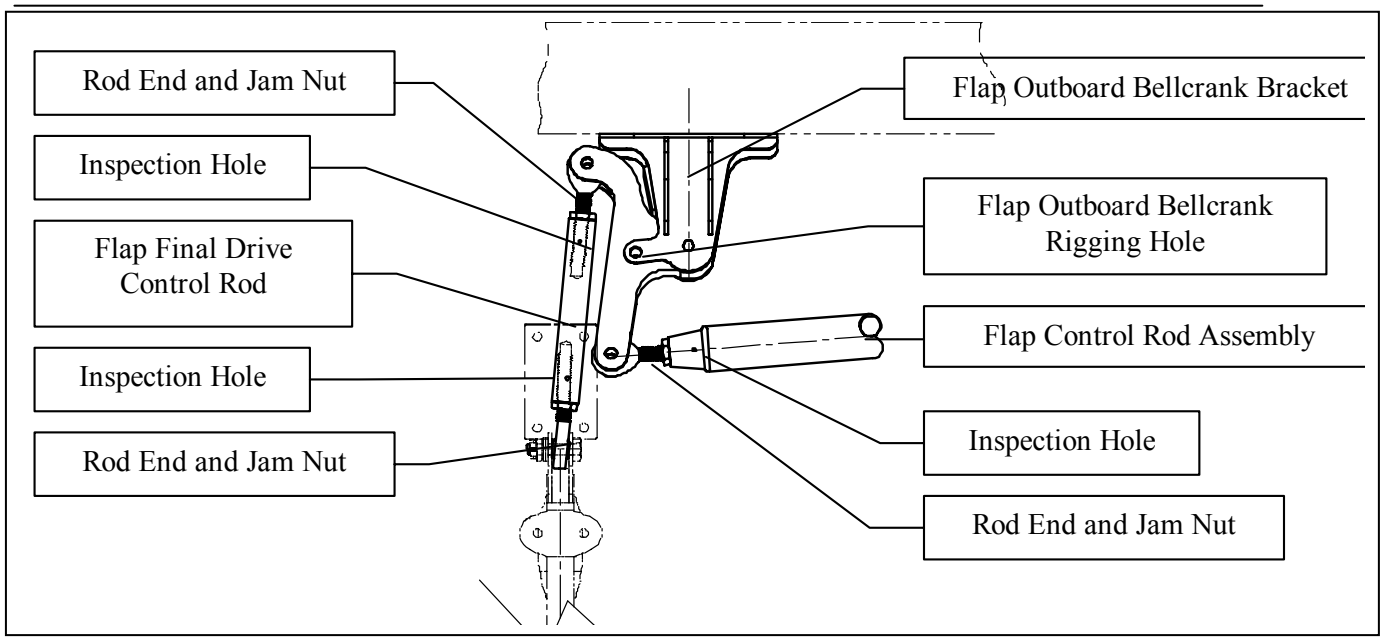


Figure 27 - 30 Flap Outboard Bellcrank and Final Drive

5. After adjusting the left and right control rod assemblies, adjust the final drive length to achieve a 0° ± 1° deflection with the flaps in the up position. After adjusting the correct length, torque the jam nuts 140 to 180 in.-lbs., and verify that the minimum thread engagement is achieved by checking that the threads extend past the inspection hole in the control rods (see Figure 27 - 30). Torque bolts through the bellcrank and control rods 72 to 84 in.-lbs., and install the cotter pin in the castellated nut. Note the use of washers between the bellcrank and both sides of the rod end. If replacement of these washers is necessary, they may be replaced with AN960-416 washers.

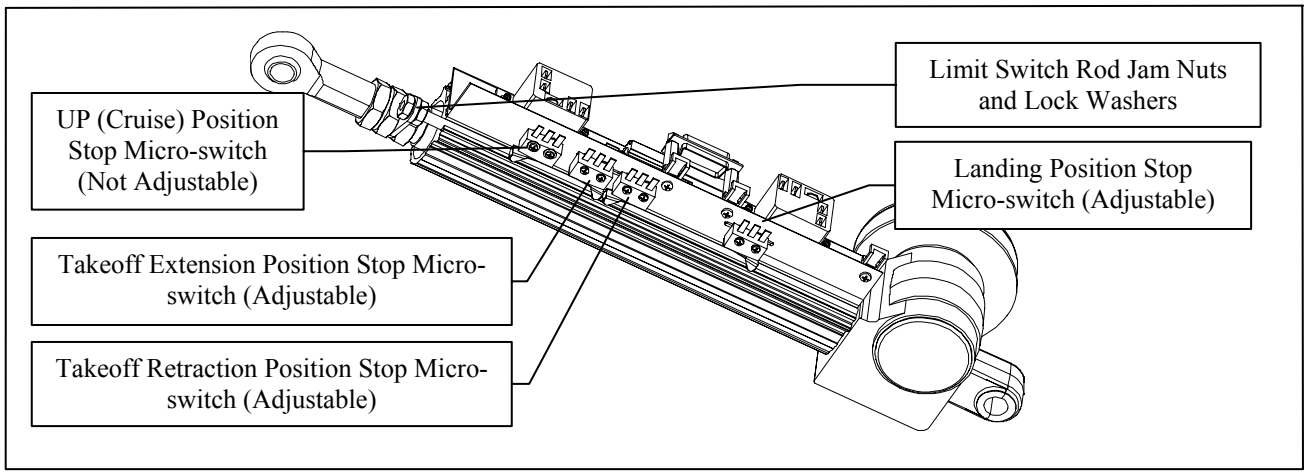


Figure 27 - 31 Flap Actuator and Limit Switch Detail

6. Using the flap/aileron template, part number TA57270000, lower the flaps to the takeoff position, and adjust the takeoff micro-switches to achieve 12° ± 1° deflection on both flaps. With the extension micro-switch adjusted, move the retraction micro-

switch as close as possible to the extension switch without interference. Note when adjusting the switches, always check deflection when the flaps are extended from the up position to the takeoff position.

7. Lower the flaps to the landing position, and adjust the landing micro-switch to achieve 40° ± 1° deflection on both flaps using the rigging template.
8. Ensure that the gap at the inboard edge of the aileron between the flap and the aileron (when the flaps are retracted) is 0.125 to 0.375 in. Ensure positive clearance when the flaps are extended.
9. Actuate the flap system, and check for binding or rubbing of components with the flap system. If problems are noted, re-check the rigging.

c. **Servicing and Maintenance**

The flaps do not require any service or maintenance. The flaps should be visually inspected annually for cracking at all hinge attachment locations. A tap test should be conducted on the flap at the inboard and outboard end caps to aid in detection of latent damage or disbonds. Liquid lubricants should not be used on the rod end bearings as this may cause the Teflon liner to swell creating excess friction in the bearing. Should the rod end need to be replaced, contact the manufacturer. Grease the wing flap actuator worm screw and gearbox as needed with MIL-PRF-23827 Type II (Aeroshell 7) grease.

27-23. FLAP ACTUATOR

The flap system is actuated by an electric linear actuator mounted to a bracket on the aft spar in the belly-section of the aircraft wing. This actuator assembly includes a motor, clutch, ball-screw, and limit switch assembly. The limit switch assembly provides the electronic flap stops for the three positions: cruise, takeoff, and landing.

a. **Removal and Installation**

1. Flaps must be in the cruise or landing position to remove the actuator. It is recommended to place the flaps in the cruise (up) position.
2. Access the flap actuator through the flap and aileron pushrod access panel.
3. Disconnect the power wire bundle to the actuator assembly (see Figure 27 - 32).
4. Tape or block the flaps to prevent them from moving when the flap actuator is disconnected from the central flap bellcrank.
5. Through the holes in the flap central bellcrank bracket, disconnect the flap actuator from the bellcrank as shown in Figure 27 - 32.
6. Remove the bolt holding the actuator to the bracket. Note the spacers and washers as shown in Figure 27 - 33 and Figure 27 - 34.
7. Remove the actuator from the airplane.
8. Reinstall in the reverse order.
9. If the limit switches have been moved, a recheck of the flap rigging is required and adjustments required as necessary.
10. Torque all 0.250 in. diameter bolts 72 to 84 in.-lbs. and all 0.375 in. diameter bolts 240 to 260 in.-lbs., and install the cotter pin.

b. **Maintenance**

1. During normal inspections and maintenance, there should be checks for binding, fretting, chafed wires, and slop in the actuator mounting. Check the groove for the limit switch assembly and the limit switch rod (see Figure 27 - 29).

2. If there is excess play in the actuator mounting, inspection and possible replacement of the spacers is required. To replace the spacers, follow the instructions for actuator removal and installation. Install new spacers, bolts, and washers as necessary.

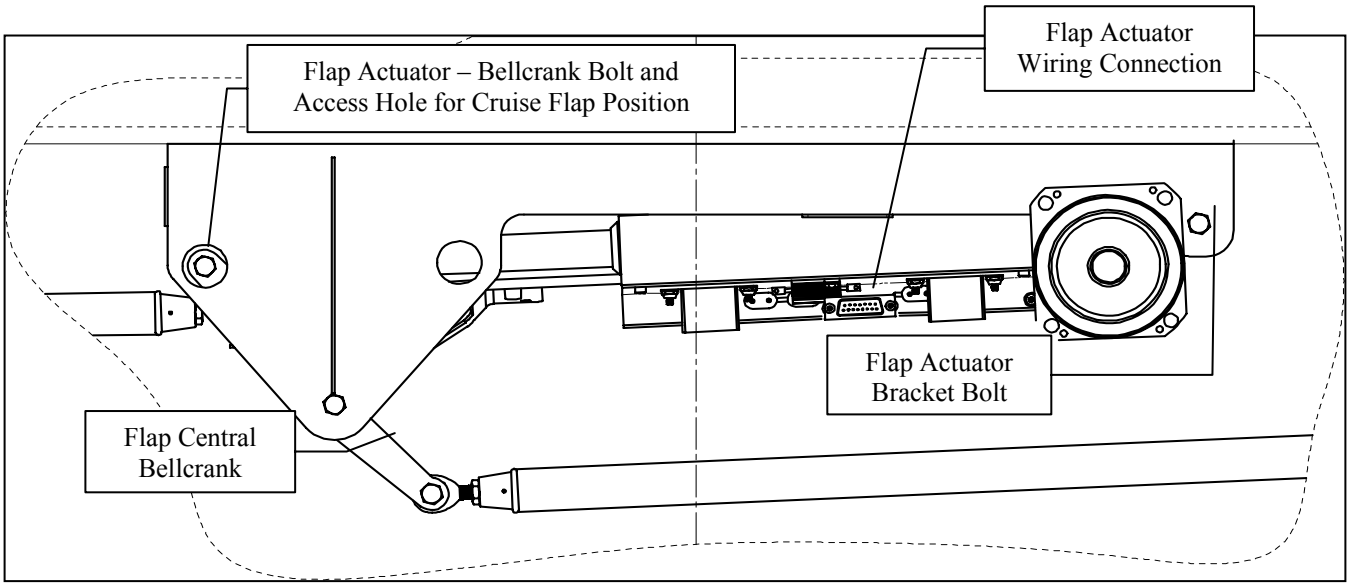


Figure 27 - 32 Flap Central Bellcrank and Actuator Assembly

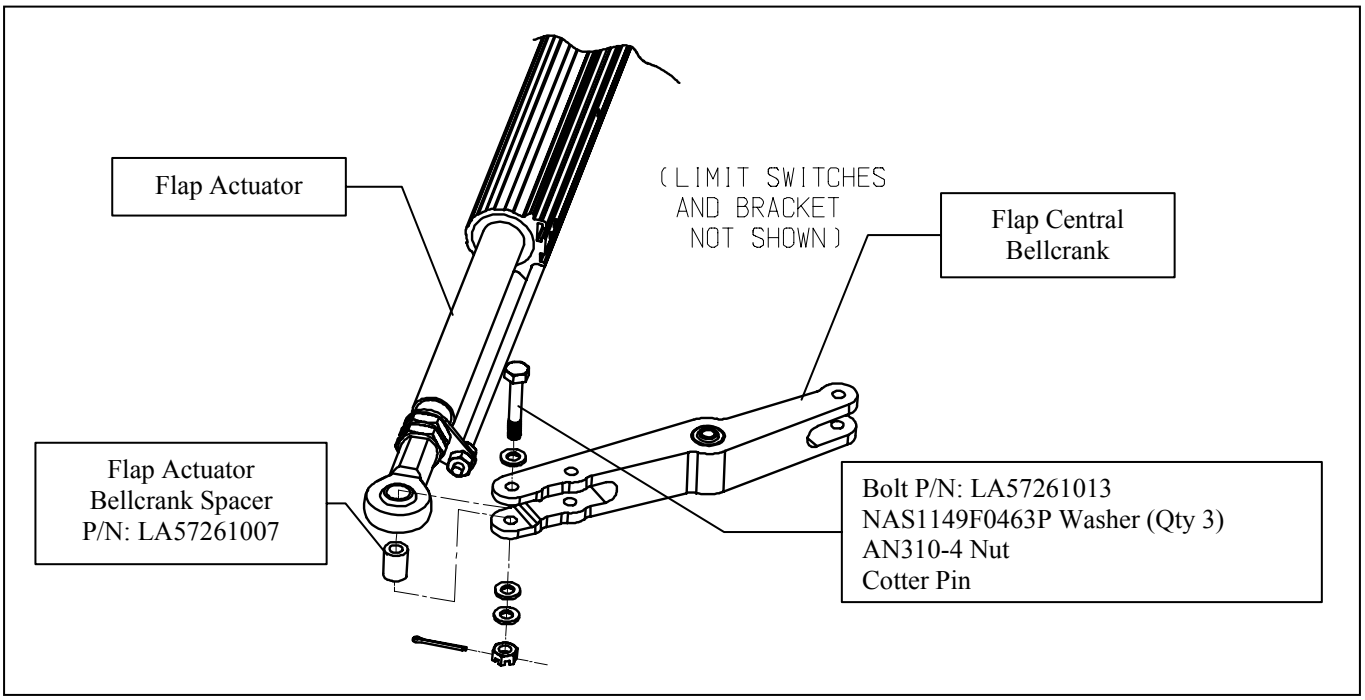


Figure 27 - 33 Flap Actuator to Bellcrank Bolt and Spacer Detail

3. Reinstall in the reverse order.
4. If the limit switches have not been moved, a recheck of the flap rigging is required and adjustments required as necessary.

5. Torque all 0.250 in. diameter bolts 72 to 84 in.-lbs. and all 0.375 in. diameter bolts 240 to 260 in.-lbs., and install the cotter pin.

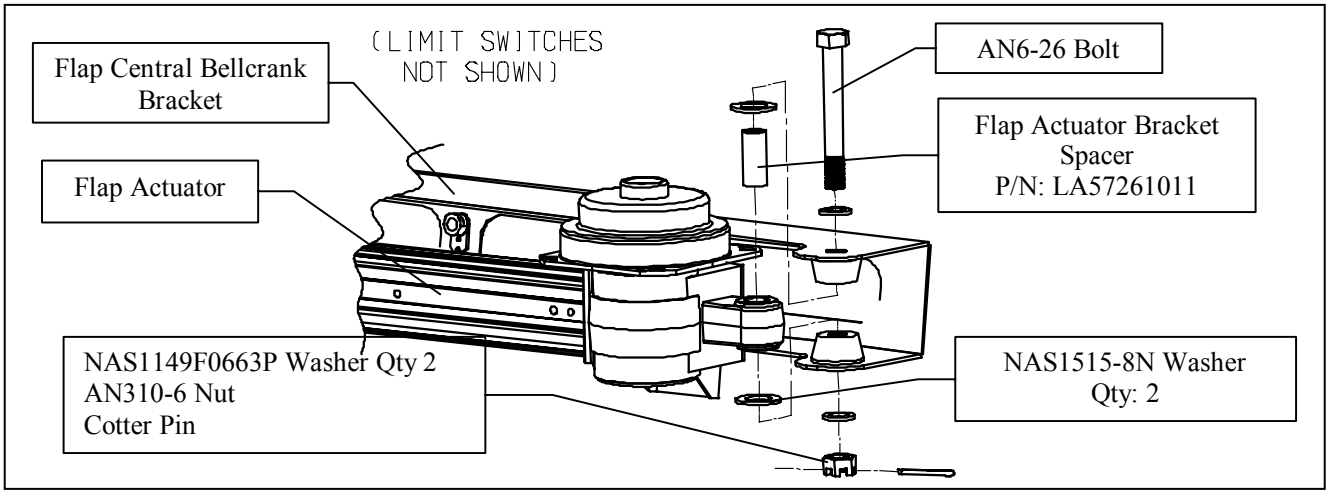


Figure 27 - 34 Flap Actuator to Bracket Bolt and Spacer Detail

6. If problems are found with the micro-switches, replacement is required. Contact the manufacturer for replacement micro-switches. Replace the micro-switch by removing wires and unscrewing the micro-switch. Replacement is a direct reversal of the removal. R-rigging is required for the replaced micro-switch.
7. If the internal clutch requires replacement, contact the manufacturer for a replacement actuator.
8. All the bearings and rod ends used in the flap system are Teflon ball bearings that require little maintenance other than cleaning during routine inspections. Do not use liquid lubricants as they cause the Teflon to swell creating excess friction in the bearing. If bearings or rod ends require replacement, contact the manufacturer for replacement parts.
9. During normal inspections, check control rods, bellcranks, hinges, and brackets for excessive wear, cracks, scratches, and corrosion. Replace worn or damaged components as necessary.
10. See Chapter 33 of this manual for detailed maintenance procedures of the flap electrical system.

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27-24. SPEEDBRAKE™ SYSTEM (OPTIONAL) - GENERAL

a. **System –** The Precise Flight SpeedBrake™ 2000 System provides expedited descents at low cruise power, glide path control on final approach, airspeed reduction, and an aid to the prevention of excessive engine cooling in descent. Extended SpeedBrakes™ will stow immediately upon application of the rudder limiter and require the SpeedBrake™ rocker switch be cycled OFF and then ON to re-extend the SpeedBrakes™. The SpeedBrakes™ will not automatically re-extend and must be recycled after the following conditions:

1. Circuit Breaker Pull
2. Automatic Stowage Due to Asymmetric Deployment or Low Voltage
3. Rudder Limiter Solenoid Engagement

b. **Components** (See Figure 27 - 35) **–** The Series 2000 SpeedBrake™ consists of wing mounted electric SpeedBrake™ Cartridges. A central logic-switching unit interconnects each SpeedBrake™ cartridge electronically and a panel mounted SpeedBrake™ actuator switch controls SpeedBrake™ deployment. The SpeedBrake™ cartridges receive electrical power from the aircraft electrical bus through a disconnect type circuit breaker.

1. Rocker Switch **–** The SpeedBrake™ rocker switch is located next to the throttle in the center of the instrument panel. The switch is positioned UP/ON to fully deploy and is positioned DOWN/OFF to retract the SpeedBrakes™.
2. Annunciator **–** The SpeedBrake™ Annunciator is located above and to the right of the airspeed indicator on the pilot instrument panel. The Annunciator will fully light after the SpeedBrake™ Switch is toggled ON and both brakes are in the up position. If one or both lights in the annunciator fails to light and both brakes do not extend after the switch is toggled on, it indicates a failure of one or both SpeedBrake™ cartridge(s) and the SpeedBrake™ switch should be toggled off. When the SpeedBrake™ Switch is toggled OFF, the annunciator will extinguish when both brakes are fully stowed in the wing.

NOTE

The SpeedBrake™ Cartridges and/or the Asymmetric Logic Control Unit must be returned to the manufacturer should repair or replacement be required. Even though only one cartridge may require repair or replacement, both cartridges must be returned to the manufacturer.

27-25. SPEEDBRAKE™

a. **SpeedBrake™ Cartridge Removal –** The SpeedBrake™ cartridges are located within the upper side of the wing and approximately 1/3 wing length from the tip of the wing. One SpeedBrake™ cartridge in each wing. See Figure 27 - 36 and Figure 27 - 37.

1. Remove the access panel on the underside of the left or right outboard wing**.**
2. Disconnect the electrical connection.
3. Remove the two 8-32 screws and remove the cartridge. Note the screw length and the hole from which it was removed. The screws may be of differing lengths and must be reinstalled in the holes from which they were removed.

4. A bottom mount bar is attached to the bottom of the SpeedBrake™ cartridge by two 8-32 round head screws. The bottom mount bar may be removed by removal of these screws.

b. **SpeedBrake™ Cartridge Installation** (See Figure 27 - 36 and Figure 27 - 37).
   1. Attach the bottom mount bar to the bottom of the SpeedBrake™ cartridge with two 8-32 round head screws. Torque the screws 6 to 8 in.-lb.
   2. Insert the speed brake cartridge into the wing.
   3. Connect the electrical connection.
   4. Install the 8-32 screws. Screws may be of differing lengths and must be reinstalled in the holes from which they were removed. Torque 15 to 18 in-lbs. Installed SpeedBrake™ cartridge should be flush with the wing surface within ± .020 inches. Bottom wing skin should have no more than .003 inch distortion after torquing.
   5. Replace the access panel.

c. **Asymmetric Logic Control Unit Removal–** The asymmetric logic control unit is located in the avionics shelf under the baggage compartment floor. See Figure 27 - 35.
   1. Remove the carpet and access panel from the baggage compartment floor per instructions in Chapter 25.
   2. Loosen the mounting screws that hold the unit to the avionics shelf.
   3. Disconnect all connectors to the unit.
   4. Remove the unit.

d. **Asymmetric Logic Control Unit Installation** (See Figure 27 - 35).
   1. Connect the connectors.
   2. Place the unit in its place on the avionics shelf.
   3. Install screws to connect the unit to the shelf and torque screws 15 in.-lbs.
   4. Install the access panel.
   5. Replace the carpet.

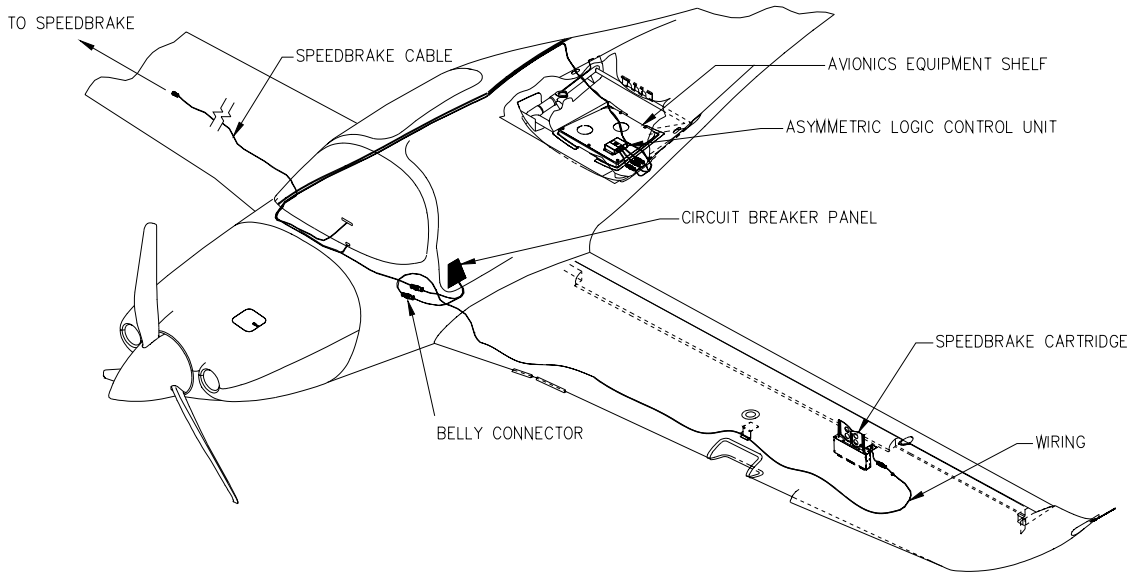

Figure 27 - 35 SpeedBrake™ System

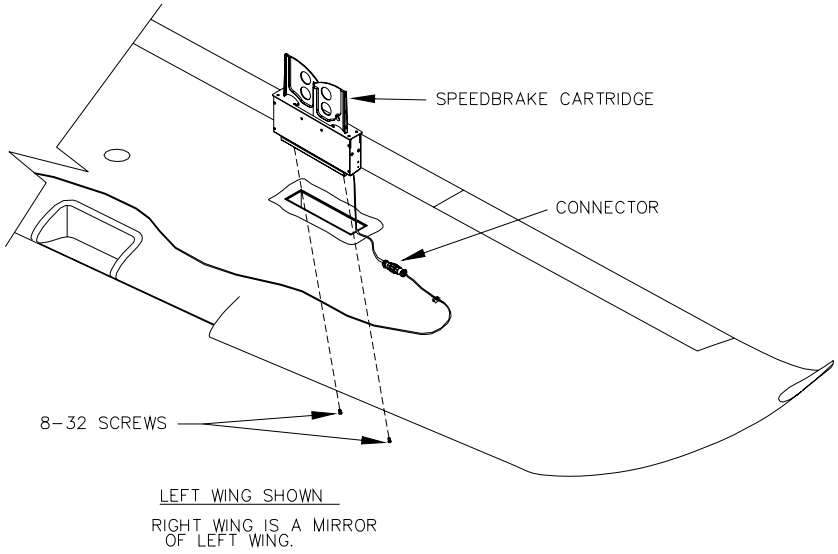

Figure 27 - 36 SpeedBrake™ Cartridge

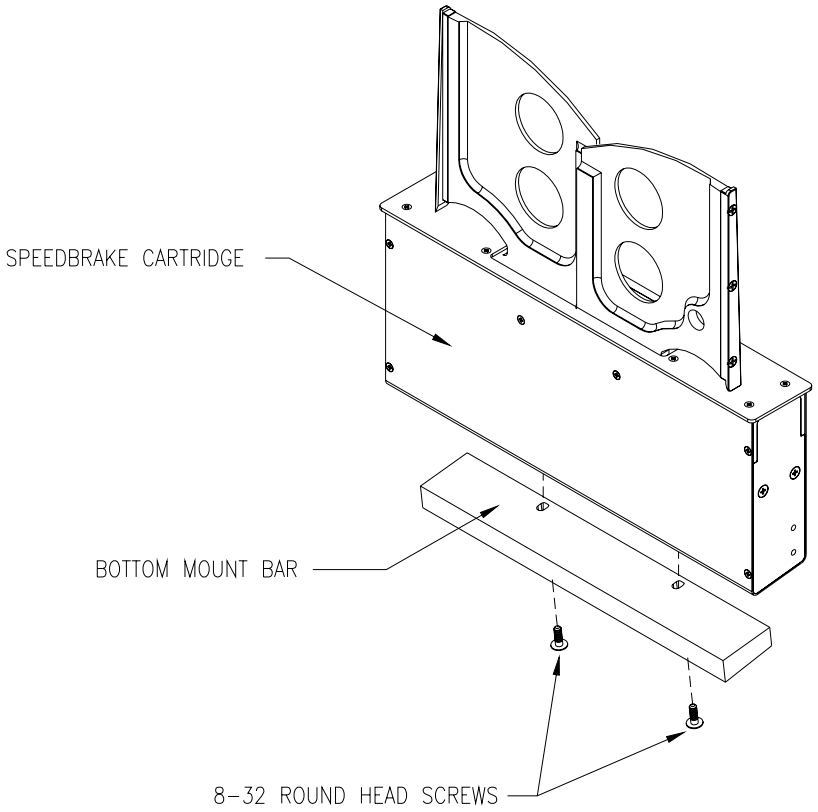

Figure 27 - 37 SpeedBrake™ Cartridge Detail

e. **Servicing and Maintenance**

1. Check SpeedBrake™ Cartridge cap strip cover screws and top attachment screws for security. If loose, remove screws and apply Loctite 242 and retorque to 8 in-lbs.
2. Annually remove the SpeedBrake™ Cartridge cover plate, clean and inspect for damage, corrosion, looseness and proper operation. Lubricate worm and worm gear with LUBRIPLATE.
3. Reinstall the cover plate by applying Loctite 242 to the screw threads and torque to 8 in-lbs.
4. Return the SpeedBrake™ Cartridge to the manufacturer every 1000 hours for clutch lubrication and spring replacement.
5. Return the SpeedBrake™ Cartridge to the manufacturer every 5000 hours for drive assembly replacement.

CHAPTER 28

FUEL

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Chapter 28

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28-1. GENERAL – FUEL SYSTEM

a. **Overview** – The fuel system has two tanks that gravity feed to a three position (Left, Right, and Off) fuel selector valve located in the wing and actuated from the selector knob in the forward part of the arm rest between the pilot and copilot seats. The fuel flows from the selected tank to the auxiliary fuel pump and then to the strainer. From this point, the fuel goes to the engine-driven pump where, under pressure, it is sent to the throttle/mixture control unit and then on to the fuel manifold valve for distribution to the cylinders. Unused fuel from the continuous flow fuel injector system is returned to the selected fuel tank. A pressure gauge on the metered side of the fuel manifold valve measures system pressure and displays both the fuel pressure and the equivalent fuel flow reading on the same gauge. See Figure 28 - 1 or Figure 28 - 2 for a general layout of the fuel system.

   Each fuel tank contains a slosh box near the fuel supply lines. A partial rib near the inboard section of the fuel tank creates a small containment area with a check valve that permits fuel flow into the box but restricts outflow. The slosh box is like a mini-fuel tank that is always full. The slosh box's function, in conjunction with the flapper valves, is to ensure short-term positive fuel flow during adverse flight attitudes, such as when the airplane is in an extended sideslip or subject to the bouncing of heavy turbulence.

b. **Fuel Quantity Indication –** The airplane has integral fuel tanks, commonly referred to as a "wet wing." Each wing has two internal, interconnected compartments that hold fuel. The wing's slope or dihedral produces different fuel levels in each compartment and requires two floats in each tank to accurately measure total quantity. The floats move up and down on a pivot point between the top and bottom of the compartment, and the position of each float is summed into a single indication for the left and right tanks. The positions of the floats depends on the fuel level; changes in the float position increases or decreases resistance in the sending circuit, and the change in resistance is reflected as a fuel quantity indication on either the gauge (Avidyne option) or the MFD System page (Garmin G1000 option). The indicators are powered by the airplane's electrical system, protected by a two-amp circuit breaker and will not operate with the master switch turned off.

   The pilot is reminded that the fuel calculation group of the Garmin G1000 system MFD System page provides approximate indications and are never substitutes for proper planning and pilot technique. Always verify the fuel onboard through a visual inspection, and compute the fuel used through time and established fuel flows.

c. **Fuel Selector** – The fuel tank selector handle is between the two front seats, at the forward part of the arm rest. The selector is movable to one of three positions: Left, Right, and Off. The fuel tank selector handle is connected to a drive shaft that moves the actual fuel valve assembly located in the wing saddle. Moving the fuel tank selector handle applies a twisting force to move the fuel selector valve.

d. When the fuel tank selector handle is moved to a particular position, positive engagement occurs when the fuel selector valve rests in one of the three available detents: Left, Right, and Off. The left and right tanks are changed by rotating the handle to the desired tank position; initially, a small amount of additional pressure is required to move the valve out of its detent. A spring-loaded release knob in the selector handle prevents inadvertent movement beyond the right and left tank positions. To move to the off position, pull up

on the fuel tank selector and rotate the handle until the pointer is in the off position and the fuel valve is seated in the detent. To move the handle from the off position to the left or right tank, pull up on the selector, and rotate the handle to the desired tank

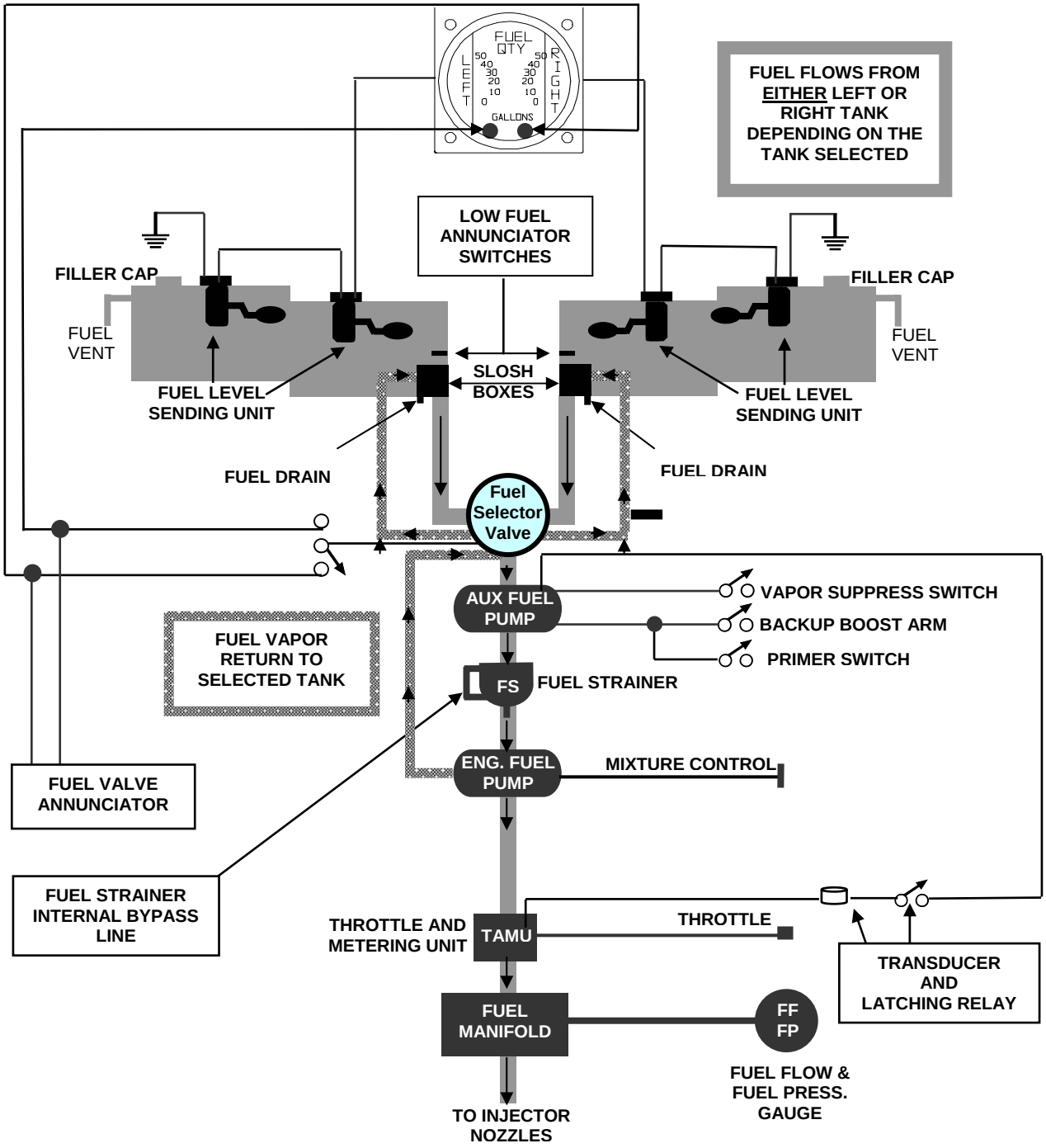

Figure 28 - 1 Fuel System Diagram (Basic or Avidyne Option)

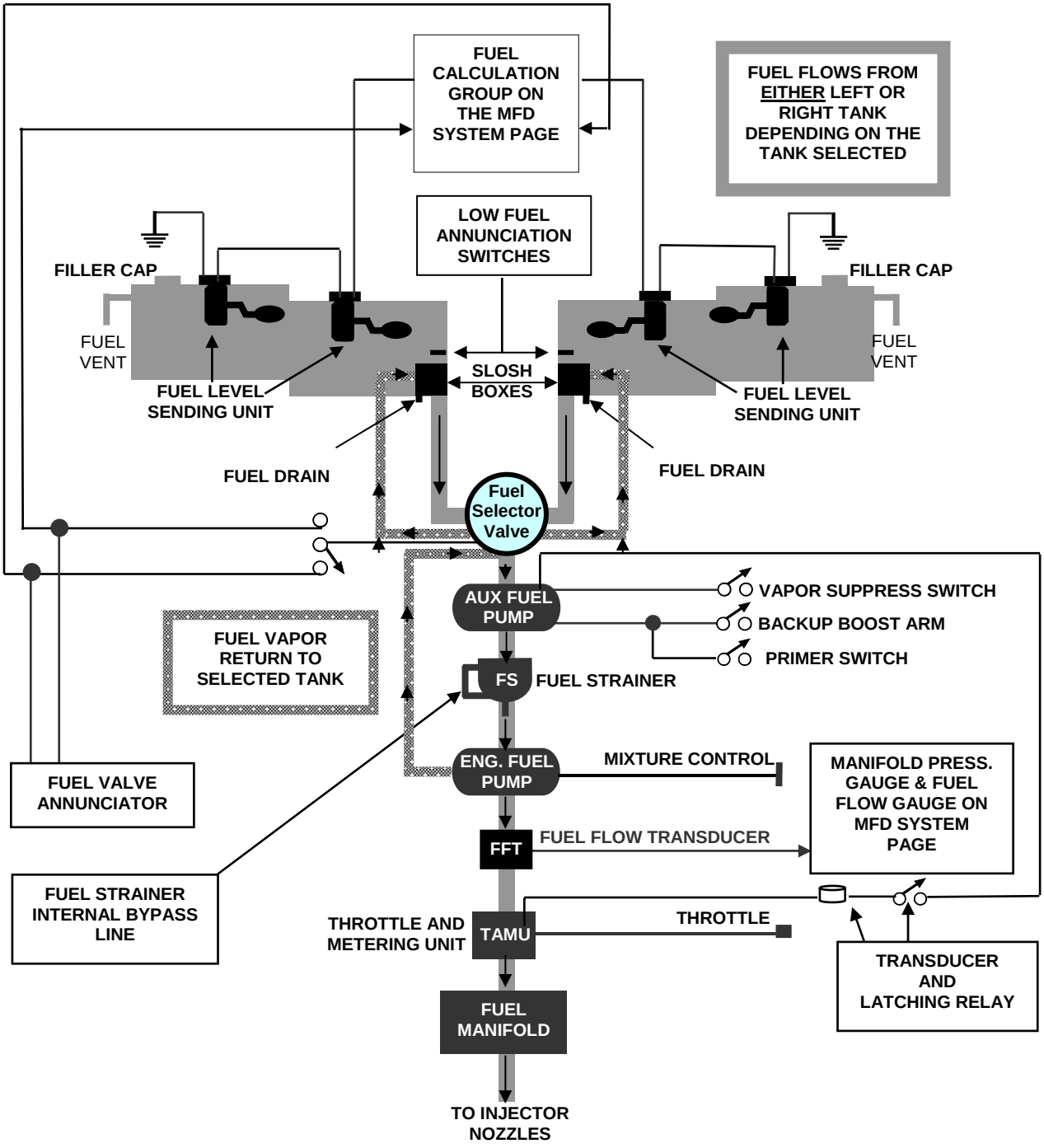

Figure 28 - 2 Fuel System Diagram (Garmin G1000 Option)

e. When a tank is selected and the selector is properly seated in its detent, one of two green lights on the left and right side of the fuel gauge (basic or Avidyne option) or one of two blue lights on the fuel calculation group on the MFD System page (Garmin G1000 option) will illuminate to indicate which tank is selected. If a tank is selected, and a light is not illuminated, then the selector handle is not properly seated in the detent. In addition, if the fuel selector is not positively seated in either the left or right detent, or is in the off position, a red FUEL VALVE light indication is displayed on the annunciator panel (basic or Avidyne option) or displayed in the PFD Annunciation window (Garmin G1000 option).

f. **Fuel Low Annunciators** – There is a separate system, independent of the fuel quantity indicators, which displays a low fuel state. A fuel level switch in each tank activates a L LOW FUEL or R LOW FUEL light on the annunciator panel (basic or Avidyne option) or displayed in the PFD Annunciation window (Garmin G1000 option) when there is less than 8 gallons (S/N 42001 to 42567), or 10 gallons (S/N 42568 and on), of usable fuel remaining in that tank. The fuel warning light has a 30 second delay switch, which limits false indications during flight in turbulent air conditions.

g. **Fuel Vents** – There is a ventilation source for the fuel tank in each wing. The vents are wedge-shaped recesses built into the access panel. They are located under the wing approximately five feet inboard from the wing tip and positioned to provide positive pressure to each tank. The vents should be open and free of dirt, mud and other types of clogging substances. When fuel expands beyond a tank's capacity, it is sent out the fuel vent if both tanks are full. An internal tank pressure of more than two to three psi will allow fuel to drain from the vents.

h. **Fuel Drains and Strainer** – The inboard section of each tank contains a fuel drain near the lowest point in each tank. The fuel drain can be opened intermittently for a small sample, or it can be locked open to remove a large quantity of fuel. The gascolator, or fuel strainer, is located under the fuselage, on the left side, near the wing saddle. Open the accessory door in this area for access to the gascolator. The gascolator is a conventional drain device that operates by pushing up on the valve stem. There is an internal bypass in the strainer that routes fuel around the filter if it becomes clogged.

28-2. SERVICING

Refer to Chapter 12 of this manual for detailed instructions.

28-3. FUEL ADDITIVES

Refer to Chapter 12 of this manual for detailed instructions.

28-4. STORAGE – Venting

a. The fuel tank vents are located in the outboard access panels. These panels have a molded scoop mounted flush with the lower surface of the wing with a bonded aluminum tube for the attachment of the fuel vent hose from the expansion area of the fuel tank. The fuel vent hose connects to the fuel vent near the top of the rib at WS 145.00. The hose is installed such that the lowest point in the line is the vent opening.

b. **Removal** – See Figure 28 - 3 or Figure 28 - 4.
   1. De-fuel the fuel tanks and open the outboard access panel/fuel vent and the outboard fuel bay access panel.
   2. Remove the hose clamp around the vent hose, and slide the hose off the vent tube bonded in the rib.
   3. If required for hose replacement, remove the hose clamp around the access panel vent tube, and slide off the tube.

c. **Installation** – See Figure 28 - 3 or Figure 28 - 4.
   1. Slide the vent hose over the vent tube bonded in the rib, and install the clamp. Torque 17 to 18 in.-lbs.
   2. Slide the vent hose over the access panel vent tube, and install the clamp. Torque 17 to 18 in.-lbs.
   3. Re-install access panels. Apply Loctite 222 or Permatex #2 and torque 12 to 15 in.-lbs.

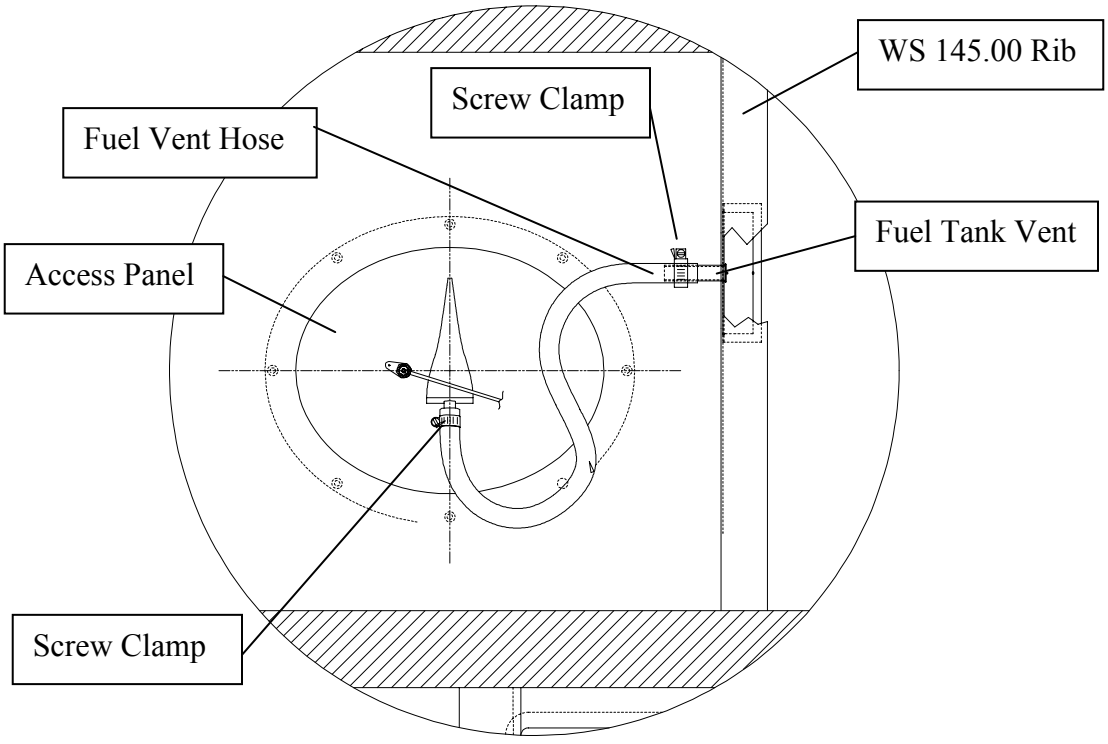

Figure 28 - 3 Fuel Tank Vent (Left Side Shown) (S/N 42001 to 42062)

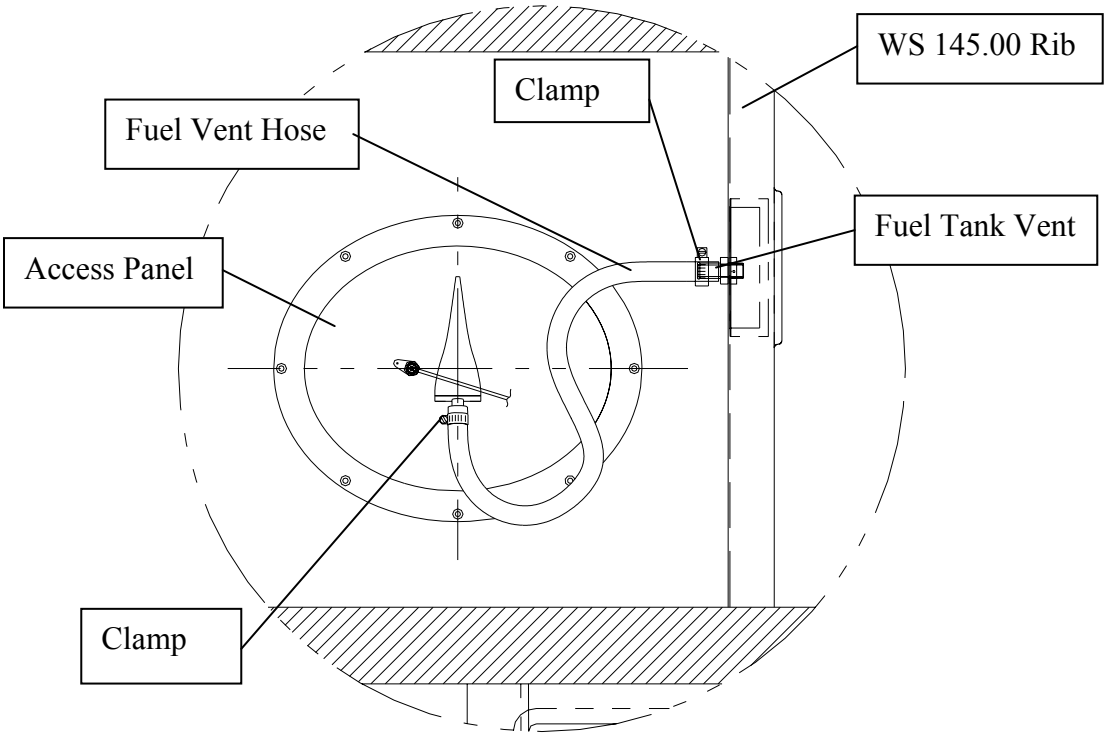

Figure 28 - 4 Fuel Tank Vent (Left Side Shown) (S/N 42063 and on)

28-5. FUEL LEVEL SENSORS

a. **General Description** – Each wing tank has two fuel level float sensors that indicate the quantity of fuel in the tank.

b. **Removal of the Fuel Level Sensors**

1. Remove the 15, 6-32 flat head screws from the fuel tank access panel nearest the fuel level sensor that requires access. (There are four fuel tank access panels, two on each wing. The access panel locations are shown in Chapter 57.) Slowly peel the panel away from the wing (the panel is also attached with sealant).
2. Remove four, 10-32 hex head bolts attaching the sensor to the bracket, noting the position of the grounding strap.
3. Remove the nut from the center fuel sensor bolt, noting the attachment of the grounding strap.
4. Remove the sensor.

c. **Installation of the Fuel Level Sensors**

1. Set the sensor on the bracket and align the four bolt holes, ensuring that the float on the sensor is pointed inboard (see Figure 28 - 5).
2. Install four, 10-32 hex head bolts with the bolt heads away from the bracket (only four of the five sensor holes will be bolted through) while reattaching the ground strap under the bolt it was initially removed from. Torque the bolts 30 to 36 in.-lbs.
3. Reattach the grounding strap under the center sensor bolt, and replace the nut (see Figure 28 - 6).

4. Replace the access panel by applying PR-1428 sealant to the side of the access panel that will contact the wing skin joggle. Replace the screws in the access panel. Torque the screws 12 to 20 in.-lbs.

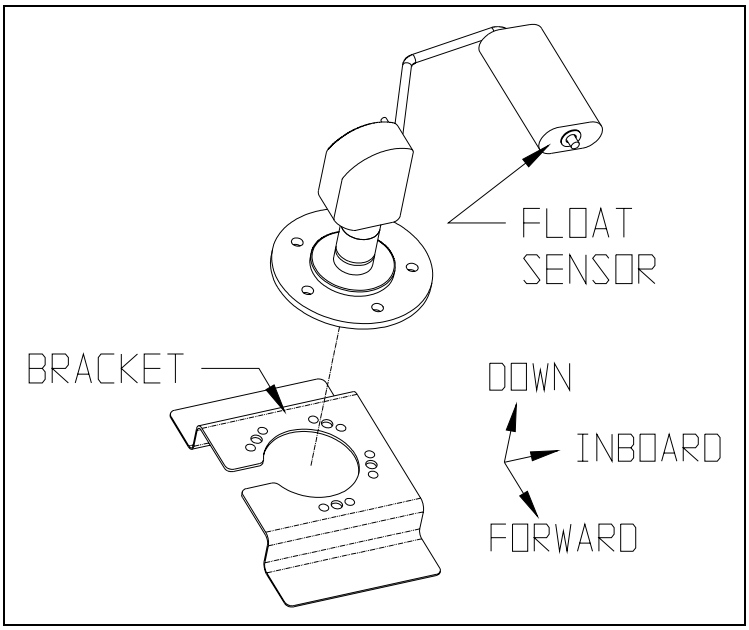

Figure 28 - 5 Outboard Fuel Level Sensor

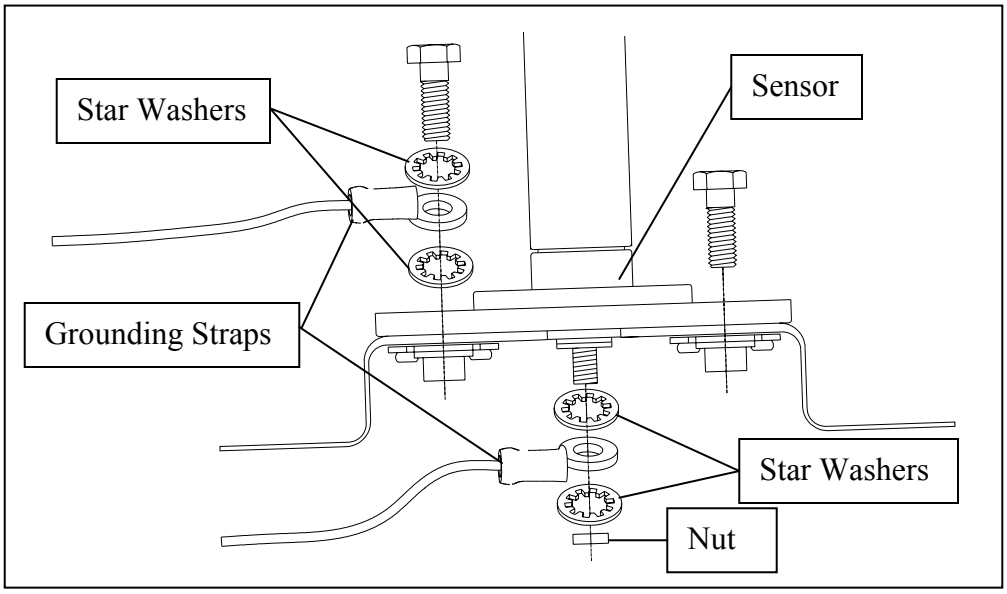

Figure 28 - 6 Grounding Strap Locations for Fuel Level Sensor

28-6. SLOSH BOXES, FUEL PICKUP, AND VAPOR RETURN

a. **General Description –** The slosh box is an internal sub-compartment mounted in the inboard section of the fuel tank. The purpose of the slosh box is to provide a compartment around the fuel pickup to hold fuel during low fuel level operation.

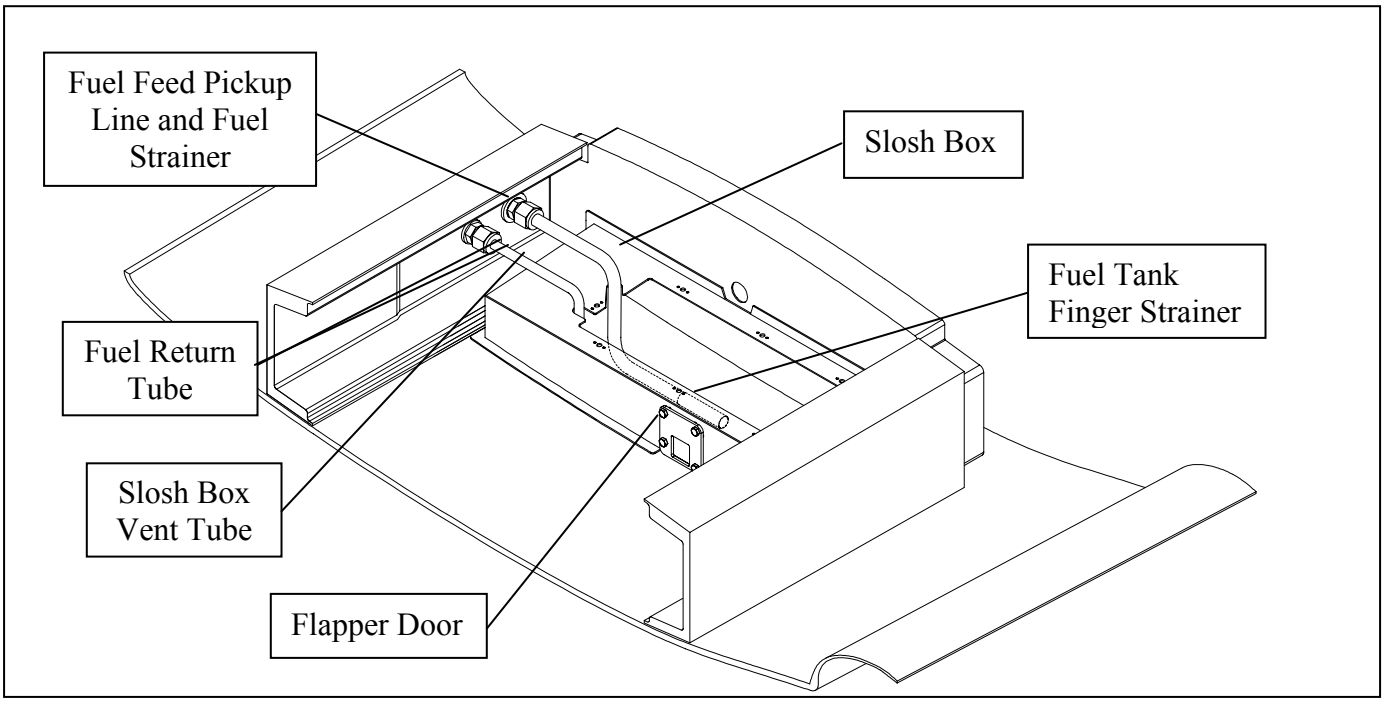

Figure 28 - 7 Slosh Box, Fuel Pickup, Fuel Return

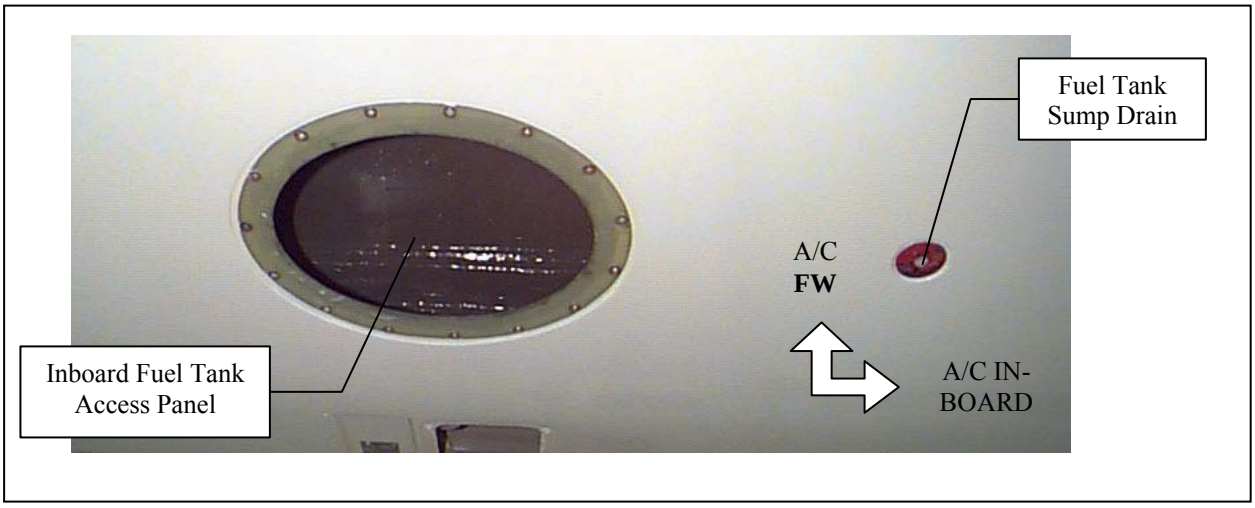

Figure 28 - 8 Inner Fuel Tank Access Panel and Fuel Drain

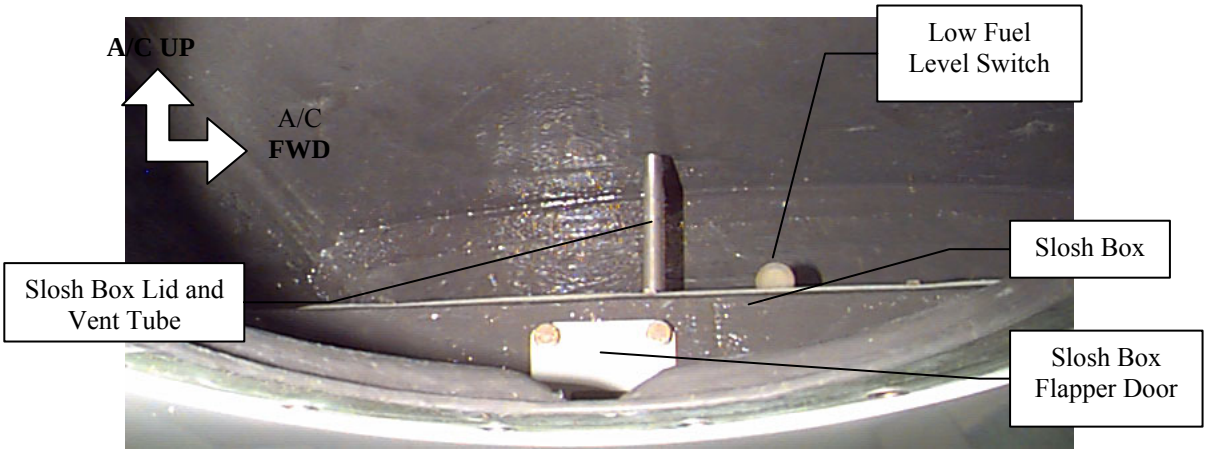

Figure 28 - 9 Slosh Box, Slosh Door, Low Level Float Switch (Facing Inboard)

b. **Replacing Vapor Return Lines**
   1. Contact the manufacturer when replacing the in-tank vapor return lines.

c. **Removal – Fuel Tank Strainer**
   1. Defuel the aircraft, then open the inboard fuel bay panel.
   2. Remove the slosh box lid to gain access to the fuel tank finger strainer. (See Figure 28 - 10.)
   3. Loosen the hose clamp, and remove the finger strainer.

d. **Installation – Fuel Tank Strainer**
   1. Installation is a direct reversal of the removal.
      a) Torque nut 17 to 18 in.-lbs.

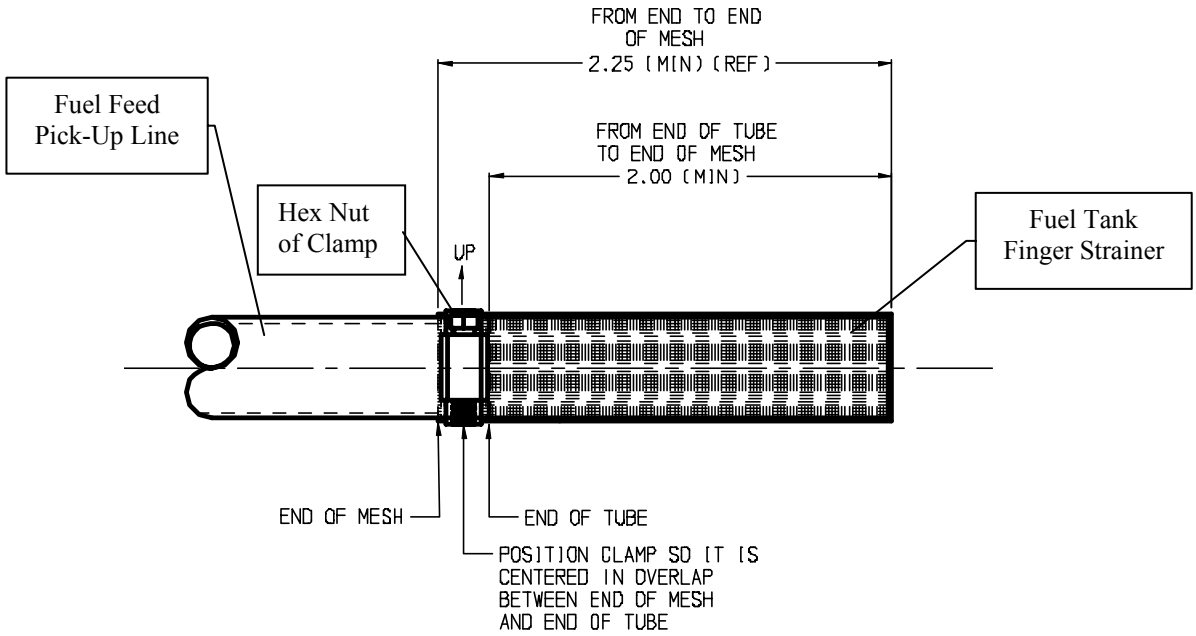

Figure 28 - 10 Finger Strainer Attachment Detail

e. **Replacing the Slosh Door**
During normal operation, the slosh door should not wear. During tank inspections verify smooth operation of the door without binding.

f. **Removing the Slosh Door**
1. Access the slosh door through the inboard fuel bay access panel.
2. Unbolt and remove the four mounting bolts that attach the slosh door to the slosh box.

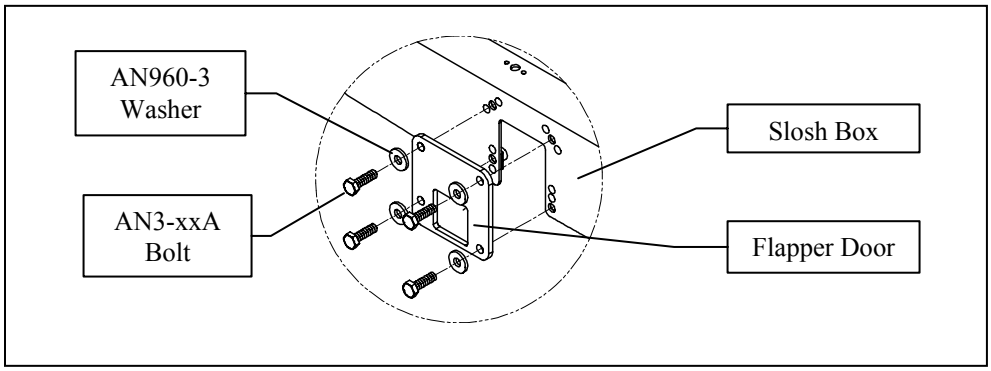

Figure 28 - 11 Slosh Door Installation/Removal

g. **Installing the Slosh Door**
1. Installation is a direct reversal of removal.

28-7. TESTING FUEL SYSTEM

a. **Procedures for Pressure Testing of Fuel Tank**
1. Two people are required to perform this test. One person is needed to observe the manometer, and one person is needed to pressurize the tanks.
2. Place the fuel selector in the off position.
3. Connect a 4 ft. length of plastic hose such as Tygon (OD $^{3}/_{8}$ in. and ID ¼ in.) to the left wing static air vent. Use sealant tape at the connection points to ensure a tight seal.
4. Connect a digital manometer that reads between 0 and 5 psi (in $^{1}/_{100}$ of an inch) with a T-fitting to the vent hose.
5. Pressurize the wing from the manometer until the manometer reads 1.00 psi ± 0.05 psi. Measurements can be taken with a low-pressure, hand-held pump, however, great care must be taken not to over-pressurize the tanks. The observer should call out manometer readings at each tenth of an inch.
6. Crimp or cap off the hose and wait for two minutes. The pressure cannot drop more than 0.10 of an inch.
7. While the fuel tank is still under pressure, apply a soapy solution to the following items to check for leak: All fittings and connectors on the fuel system from the spar to the fuel selector, around the left and right flap outboard bellcrank brackets, and the left and right fuel vent line fitting and attachment to the WS 145 rib.
8. Repeat the procedure for the right wing.

28-8. FUEL TANK DRAINS

a. Each of the two integral fuel tanks are equipped with a sump drain for the sampling of fuel and removal of moisture and residual fuel from the tanks. The fuel tank sump is located inside the slosh box as described in section 28-7 of this manual. Each drain is mounted to the

wing skin with a castellated nut. An O-ring and sealant seal the drain to the wing skin. Some features of the drains used on the Cessna 350 include a replaceable primary O-ring seal that can be replaced without draining the tanks and a locking position to aid in the defueling of the aircraft.

CAUTION

It is recommended to operate the drain in a well-ventilated area away from structures when working with fuel or fuel vapors.

CAUTION

When operating the fuel sump drain, care must be taken not to force the internal poppet into the drain, as damage to the drain may occur, and the drain may be stuck in the open position.

NOTE

The primary O-ring seal on the fuel tank sump drain is replaceable when the drain is installed without defueling the aircraft. See paragraph (d)(2) for detailed instructions on primary O-ring seal replacement.

NOTE

Contact the factory for replacement drains and access panels if required.

b. **Removal of Fuel Tank Sump Drain**

1. Defuel the fuel tank that will have the drain removed. It is recommended to at least partially defuel the other wing tank to prevent a large fuel imbalance.
2. Remove the inboard fuel tank access panel taking special care in the removal of the panel.
3. Remove the slosh box lid, as shown in Figure 28 - 12, and by removing the eight bolts (S/N 42001 to 42567) or 5 bolts (S/N 42568 and on) to gain access to the drain.

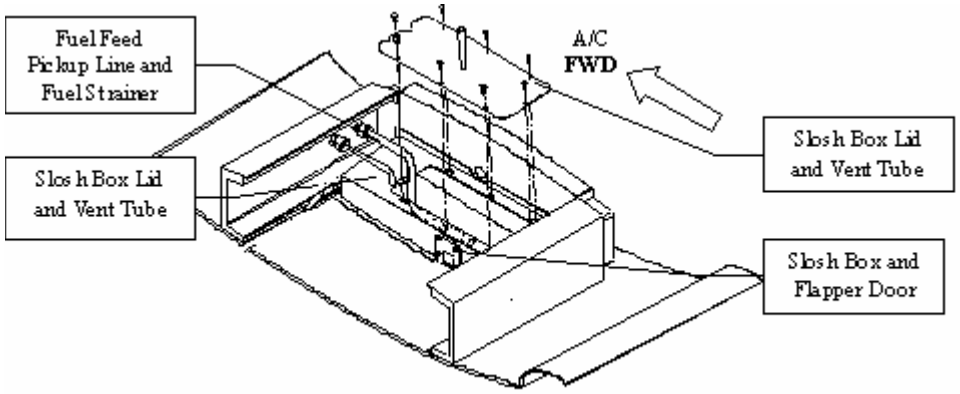

Figure 28 - 12 Slosh Box Lid Removal

4. Remove the castellated nut from the drain; use a two-prong spanner on the outside of the drain to prevent spinning during removal. Remove the drain valve. See Figure 28 - 13.

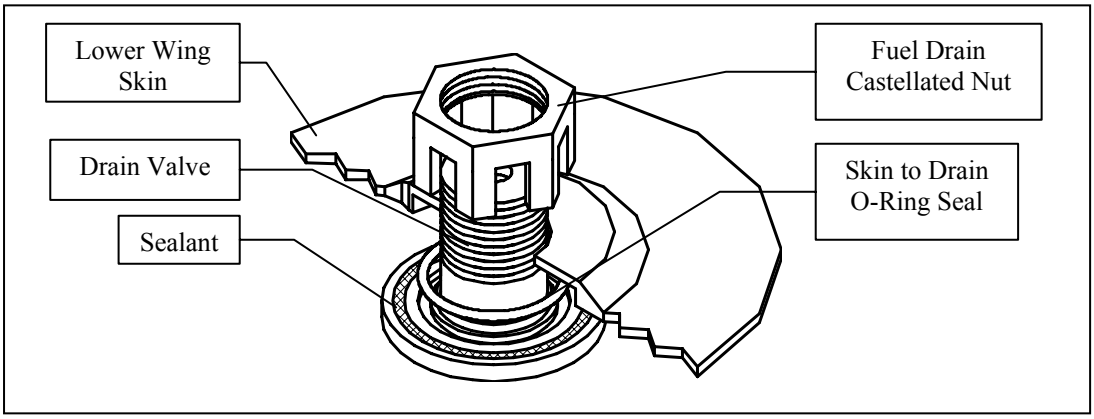

Figure 28 - 13 Fuel Tank Sump Drain Installation

c. **Installation of the Fuel Tank Sump Drain**
   1. Add a film of the fuel tank sealant discussed in section 28-12 to the upper surface of the valve assembly flange. Do not allow the sealant into the O-ring groove in the flange.
   2. Slip a new O-Ring, MS29513-021 or equivalent, over the drain, and insert in the wing.
   3. Torque castellated nut 48 to 60 in.-lbs.

d. **Maintenance Practices**
   1. Cleaning – Periodic cleaning of the bottom of the drain and primary O-ring seal is recommended.
      a) To open the drain to access the primary O-ring seal, turn the white poppet 55º clockwise using a Phillips screwdriver, and let the poppet drop down as shown in Figure 28 - 14. An internal seal will stop the flow of fuel so defueling of the tank is not necessary.
      b) Clean the O-ring and area surrounding the primary seal.
      c) Push gently up and rotate 55º counterclockwise to re-seal.
   2. Replacement of Primary O-Ring Seal
      a) To open the drain to access the primary O-ring seal, turn the white poppet 55º clockwise using a Phillips screwdriver, and let the poppet drop down. An internal seal will stop the flow of fuel so defueling of the tank is not necessary.
      b) Remove the worn seal, clean the O-ring groove and surrounding area.
      c) Replace the primary O-Ring seal, available through the manufacturer.
      d) Push gently up, and rotate 55º counterclockwise to re-seal.

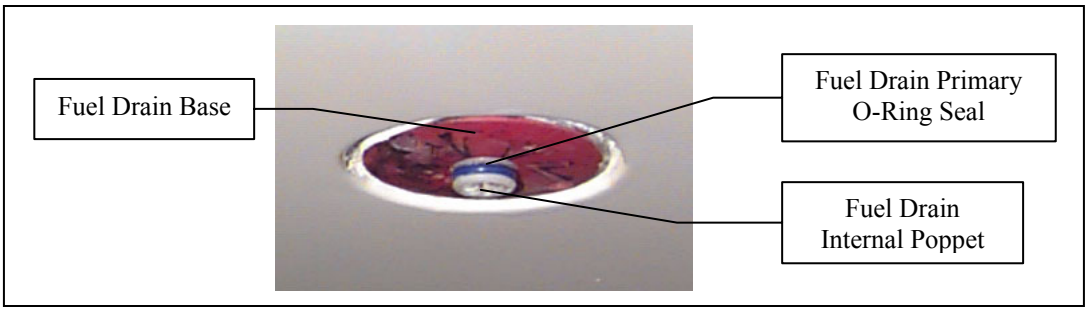

Figure 28 - 14 Fuel Drain Primary Seal Replacement

e. **Fuel Testing**

The fuel should be tested for contaminants before each flight by taking a fuel sample from the fuel drains. The operation of the fuel drain is similar to other aircraft. To operate, gently push up on the white poppet, and drain the required amount of fuel. Do not force the poppet past the initial stop, or damage may occur to the drain, and the poppet may not return to the closed position.

f. **Fuel Tank Draining**

1. To drain the fuel tanks, use a Phillips head screwdriver, and push gently up on the internal poppet of the valve to open the primary seal, and rotate counterclockwise approximately 15°.
2. Close the drain by reversing the process.

NOTE

Fuel will start flowing once the internal poppet is pushed up. Have a suitable container placed below the drain prior to opening the valve.

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28-9. FUEL TANK SEALING MAINTENANCE PRACTICES

This section discusses the general procedures and special tools and equipment for repairing leaks in the fuel tanks.

WARNING

The use of sealant is safe provided reasonable care and protective equipment is used. When working with fuel tank sealant, avoid ingestion, direct contact with skin, eyes, open breaks in the skin, etc. A well-ventilated area is required when working with sealant. Wash hands before eating or smoking. If sealant contacts skin, flush area with warm water.

NOTE

Approved fuel system materials may be obtained from the manufacturer.

a. **General Procedures**
   1. When working with fuel tanks and the fuel system, take all applicable safety precautions, and perform work in a well-ventilated area.
   2. The materials used for the Cessna 350 are selected for long life under normal conditions expected in the life of the aircraft. This section covers minor maintenance of the fuel tanks and fuel tank sealing. If a problem is outside the scope of this section, contact the manufacturer.

b. **Special Tools and Equipment**
   1. Gloves – Polyethylene
   2. Cheese cloth
   3. Brush – if brushable sealants are used.
   4. Soft plastic spatula
   5. Mixing cup or container
   6. Mixing sticks
   7. MEK and/or Acetone for surface preparation

28-10. DISCUSSION OF TERMS AND TECHNIQUES

Application Life: See Pot-Life

Pot-Life: Time after mixing a material until it becomes difficult to work with and apply.

MEK: Methyl Ethyl Ketone

MSDS: Material Safety Data Sheet

28-11. CLASSIFICATION OF LEAKS

a. It is required to inspect for leaks both inside and outside the aircraft on a periodic basis. Note that leaks can form inside the wing away from the slipstream and can be inspected for through the access panels and holes on the wing. (See section 57-7 for access panel locations.)

b. Fuel Leak Classification
   1. Stain – A slow fuel leak that evaporates soon after it is exposed to the air, leaving a blue or green stain (blue for 100 LL, green for 100).
   2. Seep – A slow fuel leak that reappears shortly after the area is cleaned.

3. Slow to Heavy Seeps – A seeping occurring in open and closed areas that do not constitute a flight hazard.
4. Running Leaks – A leak occurring in open areas, and must be repaired prior to flight. This is usually an indication of other damage, and the manufacturer should be contacted.

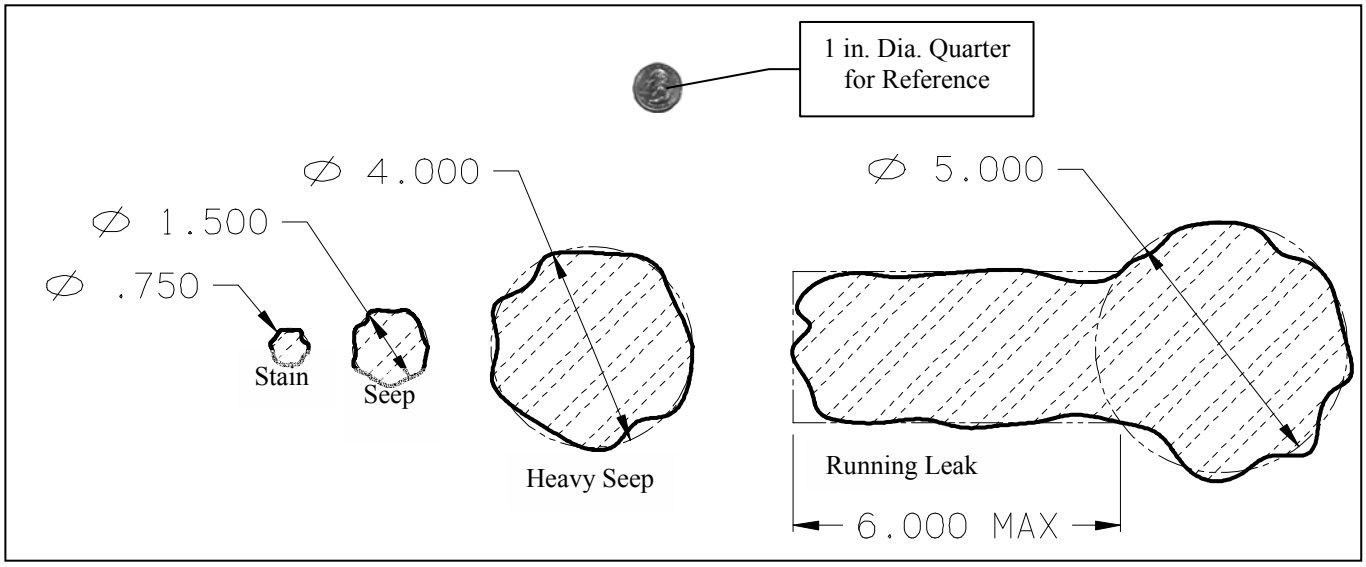

Figure 28 - 15 Fuel Leak Classification

28-12. SEALANTS FOR PANELS, VALVES, FITTINGS, AND SWITCHES

The same sealant is used on the fuel bay access panels, float valve, fittings through the forward spar, and low level float switches.

a. **Approved Sealants:**
   1. PRC PR-1428 Class B
      a) PR-1428 Class B is a two-part manganese cured, polysulfide based sealant. When mixed, it can be applied by spatula or extrusion gun.
      b) The sealant is available from the manufacturer or directly from the sealant manufacturer.
      c) Manufacturer:
         PRC Aerospace Sealants
         PRC-DeSoto International, Inc.
         5454 San Fernando Road, PO Box 1800
         Glendale, California, USA 91209
         Phone: (818) 240-2060
         Fax: (818) 549-7771
      d) Health Precautions:
         1) Refer to sealant manufacturer's warnings and MSDS.
         2) Uncured PR-1428 may produce irritation following contact with skin. When handling PR-1428 Class B, avoid contact with the body, especially contact with open breaks in the skin and ingestion. Wash hands before eating or smoking after working with any sealants or chemicals. Obtain medical

attention in case of extreme exposure or ingestion. For further information, see the MSDS for the product.

e) Cure rates available:

Sealant Number	Tack Free Time @ 77°F (25°C)	Cure Time @ 77°F (25°C)
PR-1428 B-1/2	4 Hours	8 Hours
PR-1428 B-2	8 Hours	20 Hours

Figure 28 - 16 PR-1428 Available Cure Rates

NOTE

PRC PR-1428 B1/2 is recommended because of its workability and pot-life.

f) Mixing Instructions:
   1) Mix per manufacturer's instructions.

g) Storage Life:
   1) The storage life of PR-1428 Class B is approximately nine months when stored at temperatures below 80°F (27°C) in original unopened containers. Check date of sealant prior to use.

28-13. SEALANT REPAIR

a. **Minor Sealant Repairs:**
   1. Clean around the damaged area to remove any fuel residue.
   2. Clean around the damaged area, scuffing the sealant, and cleaning with MEK and Acetone.
   3. Dry the area prior to applying fuel tank sealant.
   4. Apply fuel tank sealant with a brush or spatula avoiding blobs, stalactites, or stalagmites. Any flakes or splashes also must be removed to prevent blockage of the fuel strainers and filters.
   5. Allow sealant to completely cure prior to closing tanks and returning aircraft to service.

b. **Major Sealant Repairs:**
   1. Contact the manufacturer for repair procedures.

c. **Approved Sealants:**
   1. AC-236
      a) AC-236 is a two-component, liquid polysulfide polymer system for fuel tank sealing. When mixed it can be applied by brush, spatula, extrusion, injection gun, or roller depending on class of sealant.
      b) The sealant is available from the manufacturer or directly from the sealant manufacturer.
      c) Manufacturer:
         Advanced Chemistry and Technology (AC-Tech)
         7341 Anaconda Avenue
         Garden Grove, California, USA 92841
         Phone: (714) 373-2837

Fax: (714) 373-1913

d) Health Precautions:
   1) Refer to sealant manufacturer's warnings and MSDS.
   2) The sealant is considered safe when handled with reasonable care. Avoid contact with the body, especially contact with open breaks in the skin and ingestion.
   3) Wash hands before eating or smoking after working with any sealants or chemicals. Obtain medical attention in case of extreme exposure or ingestion. If accelerator contacts the skin, immediately wipe off and wash with waterless hand cleaner followed by soap and water. For further information, see the sealant manufacturer's MSDS for the product.

e) Cure rates available:

Sealant Number	Application Life @ 77°F (25°C) & 50%RH	Tack Free Time @ 77°F (25°C) & 50%RH	Cure Time @ 77°F (25°C) & 50%RH
AC-236 A-1/2	½ Hour	7 Hours	24 Hours
AC-236 A-1	1 Hour	16 Hours	30 Hours
AC-236 A-2	2 Hours	24 Hours	48 Hours
AC-236 B-1/2	½ Hour	8 Hours	24 Hours
AC-236 B-1	1 Hour	16 Hours	40 Hours
AC-236 B-2	2 Hours	24 Hours	48 Hours
AC-236 B-4	4 Hours	30 Hours	90 Hours
AC-236 B-6	6 Hours	48 Hours	120 Hours
AC-236 C-20	20 Hours	48 Hours	Not Available
AC-236 C-80	80 Hours	96 Hours	Not Available

Figure 28 - 17 AC-236 Available Cure Rates

NOTE

AC-Tech AC-236 A1/2 and B1/2 is recommended because of its workability and pot-life. The longer pot life and cure times are not recommended for field repair.

f) Mixing Instructions:
   1) Mix per manufacturer's instructions.

g) Curing of Sealant:
   1) Cure sealant per sealant manufacturer's recommendations prior to fueling the aircraft.

h) Storage Life:
   1) The storage life of AC-236 is nine months after the date of shipment when stored at temperatures below 80°F (27°C) in original unopened containers. Storage at lower temperatures increases shelf life. Contact sealant manufacturer for more information. Check date of sealant prior to use.

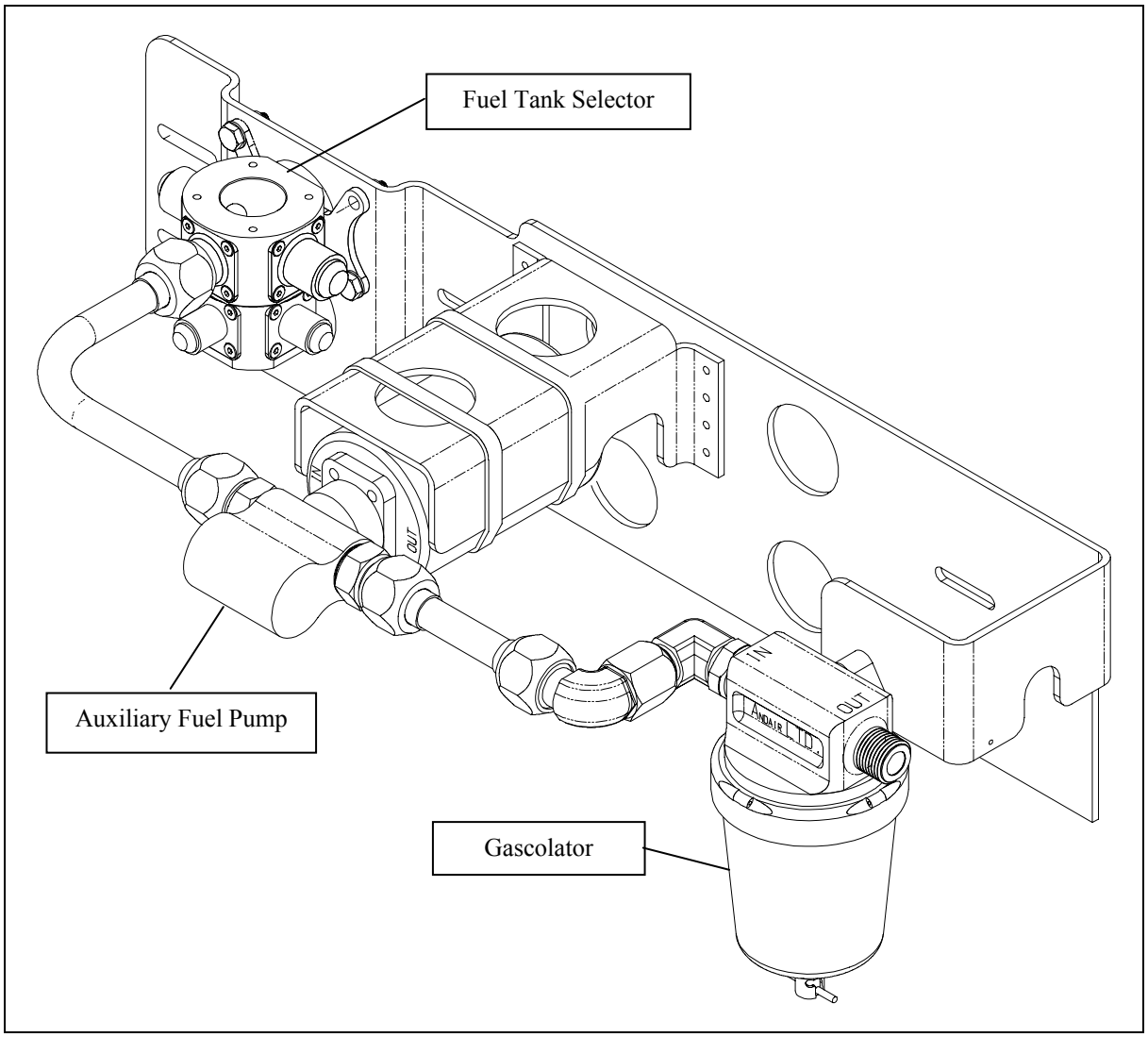

Figure 28 - 18 Fuel Selector, Pump, and Gascolator Module (S/N 42003 to 42040)

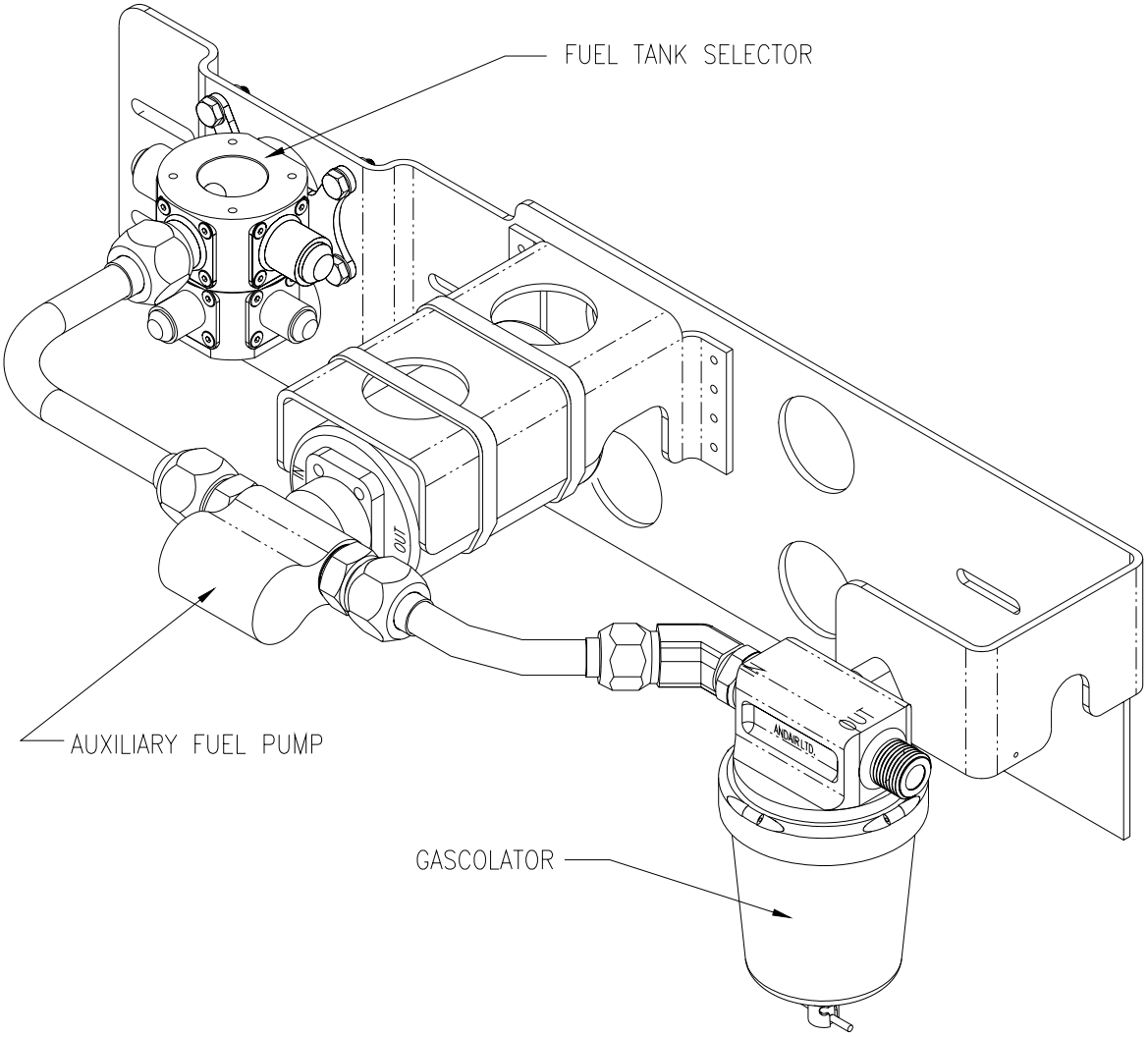

Figure 28 - 19 Fuel Selector, Pump, and Gascolator Module (S/N 42041 to 42062)

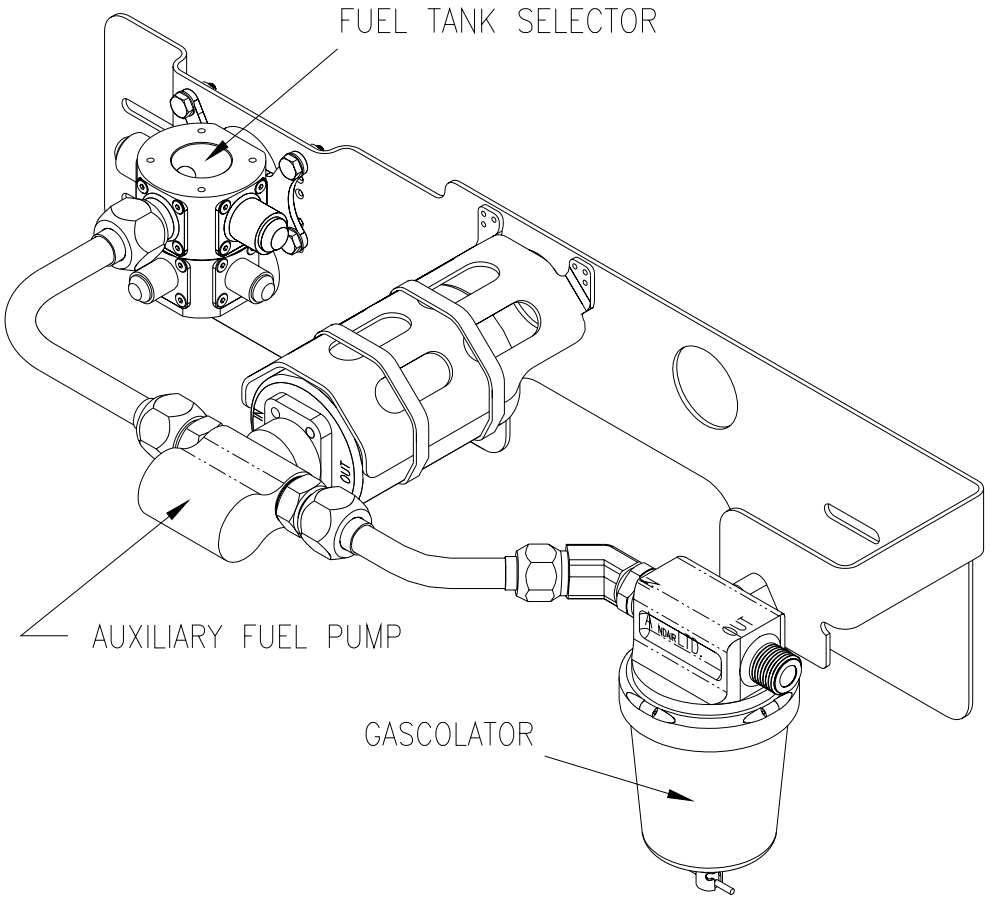

Figure 28 - 20 Fuel Selector, Pump, and Gascolator Module (S/N 42063 and on)

28-14. FUEL LINES

a. **General Practices for Fuel Line Assembly**

1. Re-bending, unbending, or tweaking tube assemblies to accomplish proper thread alignment and engagement is permissible on assembly with all established clearances, tie-downs, high points, and/or low points maintained.

   Prior to assembly, blow tubes clean and verify the ends are not damaged prior to installation. If the tube is not going to be installed after cleaning, cap ends to prevent contamination.

   As an aid for assembly and to prevent galling of mating surfaces, lubricants (e.g. petroleum based) or anti-seize products may be used on the threaded portions of fittings, connectors, etc. Such lubrication is required for stainless steel-to-stainless steel fitting connections.

2. **AN Fittings** – When required or specified, standard AN flare (i.e. 37°) shall be added to the end(s) of tubing. In forming flares, cut the tube ends square per the cutting procedures in this section, and deburr the ends per the deburring procedures in this section. The tubing is then flared using the correct 37° aviation flare forming tool for the size of tubing and type of fitting. A double flare is used on soft aluminum tubing Ø3/8 (O.D.)

and under, and a single flare on all other tubing sizes. Use appropriate caution when tightening; tighten per paragraph 5 of this section. Over tightening will damage the tube or fitting, which may cause a failure. Under-tightening may cause leakage which could result in system failure.

After flaring, tubes formed with the “Standard AN flare” (i.e. 37°) will conform to the angles shown in Figure 28 - 21 and dimensions shown in Figure 28 - 22.

In addition, the flares must meet the following requirements:

- The flare shall be square with the centerline of the tube within 0.5 in. for the distance covered by the length of the sleeve.
- The circular runout between the inner and outer surfaces of the flare shall not exceed 0.005 full indicator movement (F.I.M.) with the tubing O.D.
- The sealing surface shall be free of pit marks, radial or longitudinal scratches and indentations. Flare sealing surface shall not exceed 32 microinches (√)
- If necessary, it is recommended that the flared end of the tube be burnished during installation.

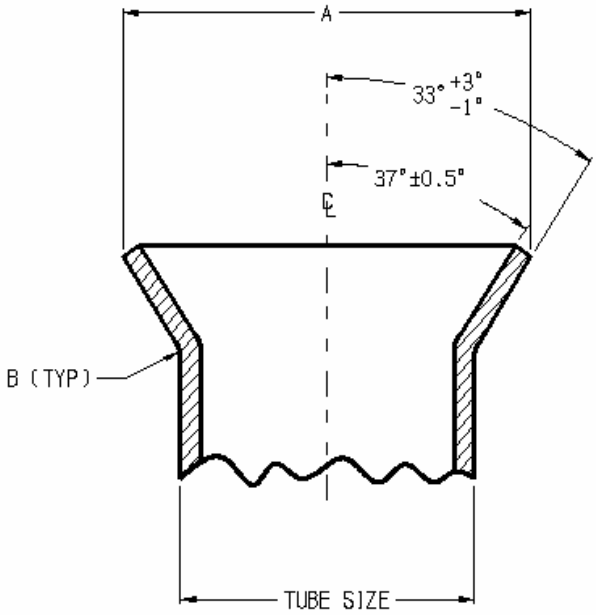

Figure 28 - 21 AN Flared Tube Angles

TUBE SIZE NOMINAL O.D. (in.)	"A" DIAMETER (in.)		"B" RADIUS ±.010 (in.)
0.125	.200	+.000 -.010	.032
0.188	.302		
0.250	.359		
0.313	.421		
0.375	.484		.048
0.500	.656		.062
0.625	.781		
0.750	.937		.078
1.000	1.187	+.000 -.015	.093
1.250	1.500		
1.500	1.721		.109
1.750	2.106		
2.000	2.356		
2.500	2.856		
3.000	3.356		

Figure 28 - 22 AN Flared Tube Dimensions

3. **Tube Cutting –** This section is provided as a guideline for tube cutting prior to installation of fuel lines. It is assumed that personnel performing this have had basic training on the use of the tools involved in correctly cutting tubing and preparing the tubing for installation.
   a) Measure and mark the tubing in the correct location for cutting/trimming. Note that based on the method of deburring, the user may have to compensate for the removal of material after cutting.

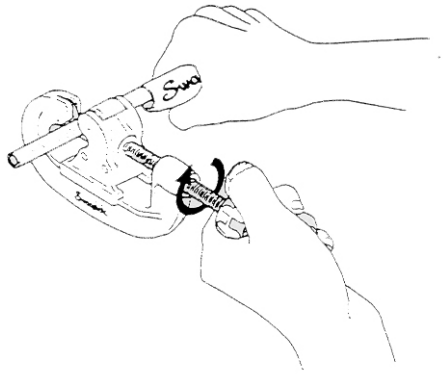

Figure 28 - 23 Inserting the Tubing into the Tube Cutter and Adjusting the Cutter

b) Maintain pressure on the cutter wheel by gradually tightening the handle as you rotate the cutter around the tube. (See Figure 28 - 23 and Figure 28 - 24.)

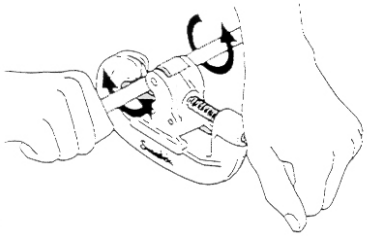

Figure 28 - 24 Cutting the Tube with Tube Cutter

c) Cutting the tube can be accomplished by either rotating the tube, the cutter completely around the tube, or by rocking it back and forth as shown in Figure 28 - 24.

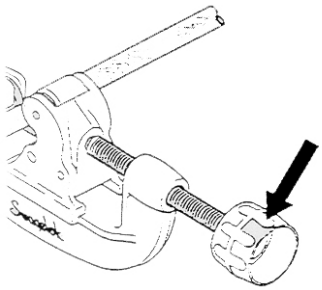

Figure 28 - 25 Adjusting the Tubing Cutter to Cut Deeper

d) To cut deeper, give a 1/8 turn of the knob for each two revolutions around the tube as shown in Figure 28 - 25.

NOTE

For a good quality cut, do not rush the process and always use a sharp cutting wheel and replace the wheel when it becomes dull.

e) Deburr the ends of the tubing per paragraph 4 of this section.

4. **Tube Deburring** – To ensure proper sealing of tubing with the fittings outlined in this specification, the ends of the tubing must be free of burrs, scratches, etc. This section describes some methods of deburring the ends of the tubing to provide the best surface for sealing and for proper system operation. Tube cutters burr into the ID of the tube and this must be removed prior to installation of the tube. Cutting tubing with a hacksaw will burr both the OD and the ID of the tubing. These also must be removed prior to tube installation.

   Small burrs on the OD may prevent the tubing from being fully inserted into the fitting. ID burrs, or chips, may cause the fittings to leak or cause damage and contamination to the system.

Small scratches in the outside surface of the tubing may be removed by carefully polishing out the defect using fine grit sandpaper, Scotch-Brite™, or other equivalent. Following any deburring or polishing, the tube must be cleaned inside and out to prevent introducing foreign material into the systems.

Figure 28 - 26 shows the proper way to deburr the OD of a cut tube using a file. Note that the file is at a shallow angle to chamfer the end slightly. If there is a raised portion of the tube from cutting, remove this also while deburring. Be careful only to remove the least amount of material from the tube to remove the burr. Removing too much material during deburring at this point will weaken the tube when installed.

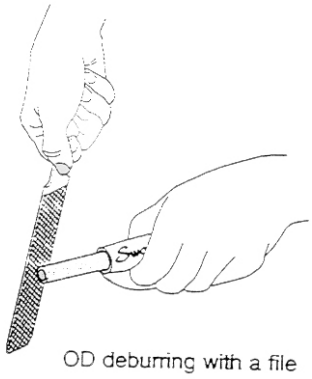

Figure 28 - 26 OD Deburring using a File

Figure 28 - 27 shows the use of a deburring tool to deburr the ID. As in deburring the OD, be careful not to remove excessive material, and only remove the burrs from the tube. There should be a slight chamfer on the ID end of the tube, and it must be free of nicks, protrusions, etc., as these affect the system operation and the strength of the tube.

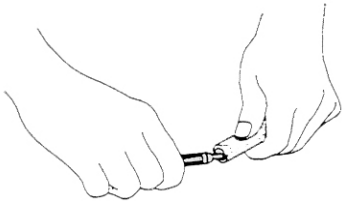

Figure 28 - 27 ID Deburring using a Deburring Tool

Figure 28 - 28 shows the use of a tubing deburring tool that can deburr both the OD and ID of the tubing. As with the above noted methods, be careful not to remove excessive material from the tube as this will weaken the tube when installed.

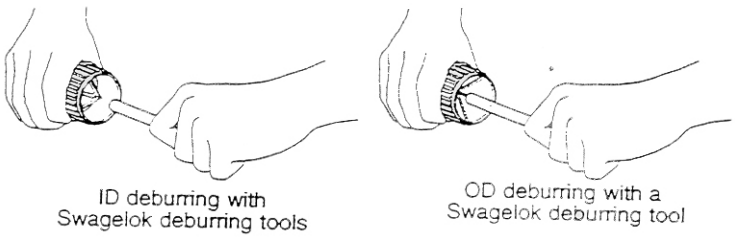

Figure 28 - 28 OD and ID Deburring using a Tube Deburring Tool

5. **Torque Values for Flared Tube Fittings –** The following torque values must be used when tightening flared tube fittings, except the torque on -6 aluminum tube must be 75 to 125 in.-lbs. and on -8 aluminum tube must be 150 to 250 in.-lbs. Over tightening will damage the tube or fitting, which may cause a failure. Under-tightening may cause leakage which could result in system failure.

Size	Tube O.D.	Material and Torque Specification			
		Tube Material	Fitting Material	*Inch Pounds	*Foot Pounds
- 2	1/8	Steel	Steel	75 - 85	6.3 – 7.1
		Aluminum	Steel/Aluminum/Brass	20 - 30	1.7 – 2.5
		Plastic	Steel/Aluminum/Brass/Plastic	10 - 15	.8 – 1.3
- 3	3/16	Steel	Steel	95 - 105	7.9 – 8.8
		Aluminum	Steel/Aluminum/Brass	25 - 35	2.1 – 2.9
		Plastic	Steel/Aluminum/Brass/Plastic	15 - 20	1.3 – 1.7
- 4	1/4	Steel	Steel	135 - 150	11.3 – 12.5
		Aluminum	Steel/Aluminum/Brass	50 - 65	4.2 – 5.4
		Plastic	Steel/Aluminum/Brass/Plastic	25 - 30	2.1 – 2.5
- 5	5/16	Steel	Steel	170 - 200	14.2 – 16.7
		Aluminum	Steel/Aluminum/Brass	70 - 90	5.8 – 7.5
		Plastic	Steel/Aluminum/Brass/Plastic	35 - 45	2.9 – 3.8
- 6	3/8	Steel	Steel	270 - 300	22.5 – 25
		Aluminum	Steel/Aluminum/Brass	110 - 130	9.2 – 10.8
		Plastic	Steel/Aluminum/Brass/Plastic	55 - 65	4.6 – 5.4
- 8	1/2	Steel	Steel	450 - 500	37.5 – 41.7
		Aluminum	Steel/Aluminum/Brass	230 - 260	19.2 – 21.7
		Plastic	Steel/Aluminum/Brass/Plastic	115 - 130	9.6 – 10.8
-10	5/8	Steel	Steel	650 - 700	54.2 – 58.3
		Aluminum	Steel/Aluminum/Brass	330 - 360	27.5 – 30
		Plastic	Steel/Aluminum/Brass/Plastic	165 - 180	13.8 - 15
-12	3/4	Steel	Steel	900 - 1000	75 – 83.3
		Aluminum	Steel/Aluminum/Brass	460 - 500	38.3 – 41.2
		Plastic	Steel/Aluminum/Brass/Plastic	230 - 250	19.2 – 20.8

*Dry torque values shown, except as noted in General Notes below

General Notes

Use a steel flare nut when using steel tubing. Either steel, brass or aluminum flare nut may be used with aluminum or plastic tubing. Plastic flare nuts to be used with plastic tubing only.

Pre-Install tube flare fitting finger tight. If parts do not assemble finger tight, check for foreign material on threads and/or damaged threads. Do not force assemblies together.

Maintain cleanliness of tube fittings. Protect all fitting threads from damage and dirt contamination.

When attaching stainless steel to stainless steel or stainless steel to aluminum, use MIL-T-5544 Anti-seize/lubricant on fitting threads. To prevent contamination, do not apply Anti-seize to tapered seat. Use lower torque value when Anti-seize is used.

b. **Removal of Fuel Return & Fuel Feed Fuel/Fuel Pickup Line at WS 48.0 and 50.25**

1. Loosen the B-nut on both sides of the bulkhead fittings, and slide the nuts onto the fuel lines. Slide off the fuel lines from both sides of the bulkhead fitting.
2. Remove the nuts on both sides of the bulkhead fitting, and slide out the bulkhead fitting as shown in Figure 28 - 30 and Figure 28 - 31.

c. **Installation of Fuel Return & Fuel Feed Fuel/Fuel Pickup Line at WS 48.0 and 50.25**

1. Insert the bulkhead fitting through the O-ring and through the spar insert as shown in Figure 28 - 30 and Figure 28 - 31. Tighten the bulkhead nuts at WS 48.0, 18 to 20 ft.-lbs. and at WS 50.25, 10 to 12 ft.-lbs.
2. Slide the fuel line onto the bulkhead fitting, then slide the B-nut over the bulkhead fitting. Tighten the flared tube fittings at WS 48.0, 150 to 250 in.-lbs. and at WS 50.25, 75 to 125 in.-lbs.

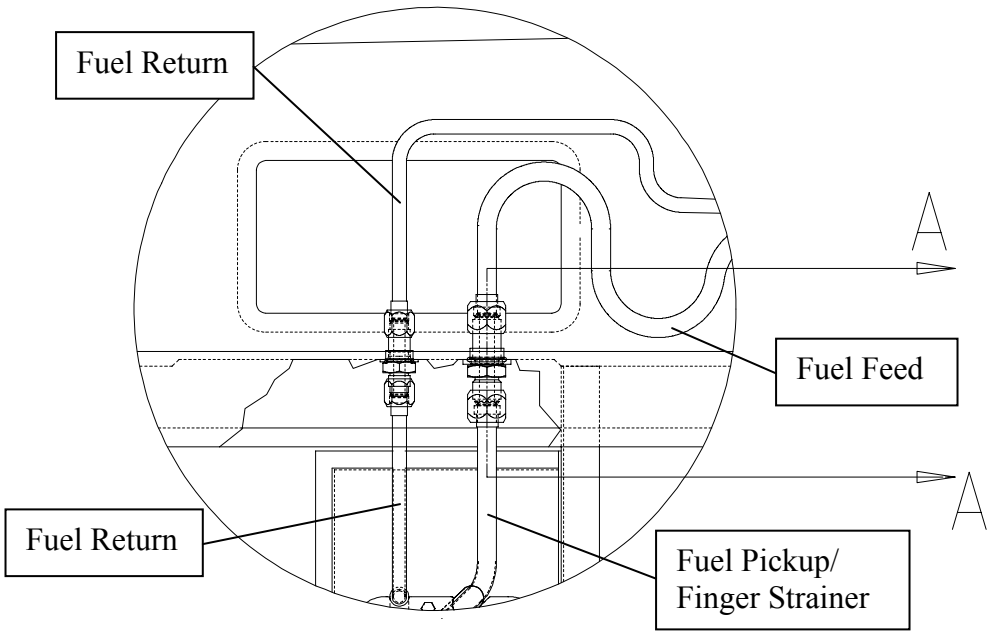

Figure 28 - 29 Fuel Return, Fuel Feed, and Fuel Pickup Lines (Left Wing Shown)

d. **General Leak Check for Each Line, Tube, or Fitting –** After any maintenance on the fuel lines or fittings, check each system for leaks with 90 psi air or nitrogen, and apply a soapy solution at each junction for 30 seconds. After a successful no-bubbles leak check, clean and dry the junctions. Apply a sentry stripe at nut to tube, nut to fitting, and fitting to mounting surface.

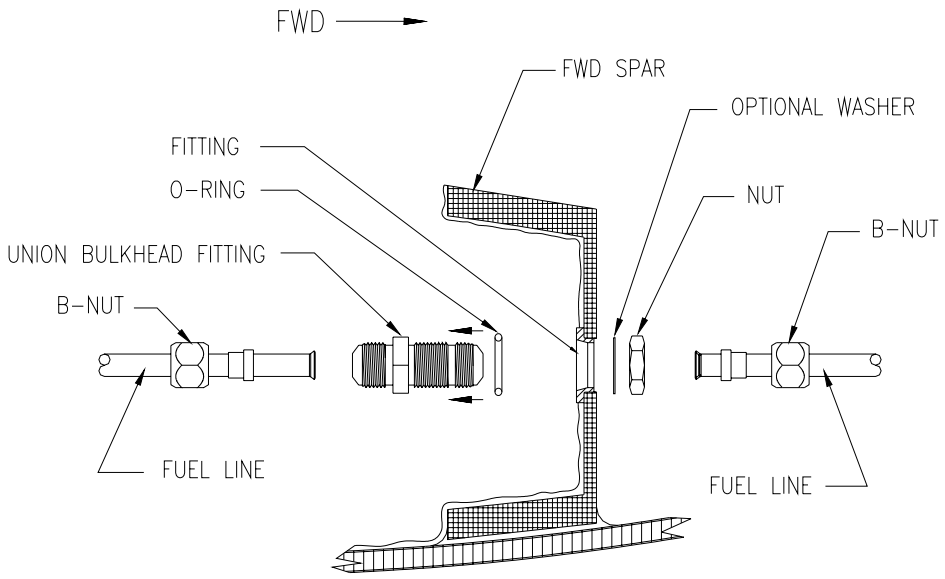

Figure 28 - 30 Installation of Fuel Line Fittings WS 48.0 & WS 50.25 (Section A-A)

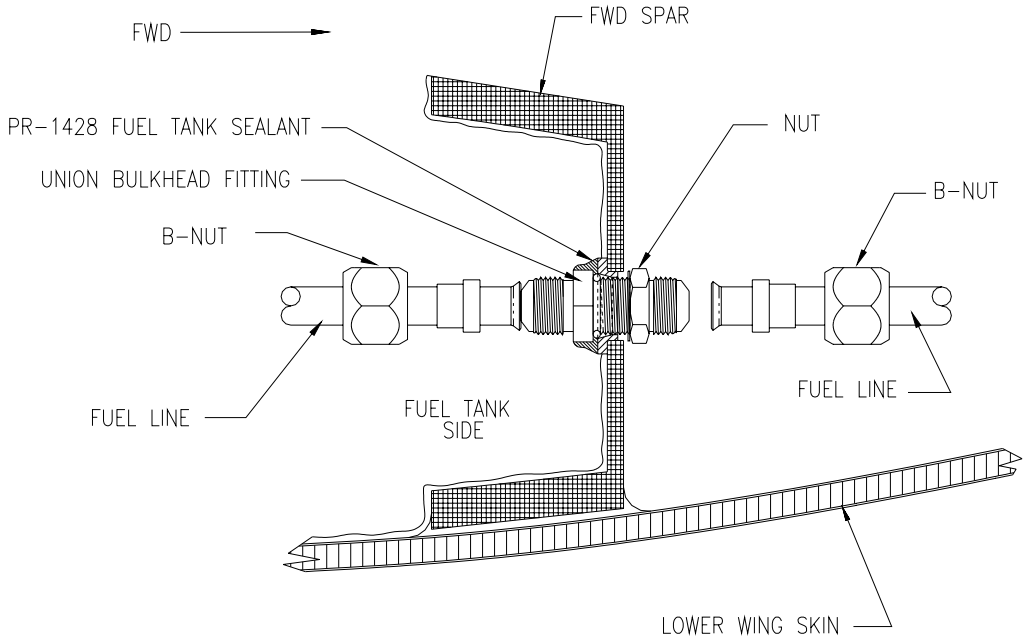

Figure 28 - 31 Installation of Fuel Line Fittings WS 48.0 & WS 50.25 (Section A-A)

28-15. FUEL SELECTOR

a. **Removal and installation of the Fuel Selector Knob**

1. Remove the co-pilot's side seat per instructions in Chapter 25.
2. Remove the center console access panels per instructions in Chapter 25.
3. Insert a Phillips screwdriver up through the inside of the center console through the access hole in the fuel selector plate, and remove the three fuel selector knob retaining screws; rotating the knob to gain access to each screw. Remove the knob. See Figure 28 - 35.
4. Inspect the knob screw holes for evidence of stripping. If stripped, contact Cessna for a replacement fuel selector knob.

5. Installation is the direct reverse of removal. Tighten the knob retaining screws snug.

b. **Removal of the Fuel Selector Valve**

CAUTION

Alignment of the shaft assembly from the fuel selector knob to the fuel selector is critical to ensure full and proper valve operation and indication. Follow step 5 below carefully.

1. Remove the co-pilot's side seat per instructions in Chapter 25.
2. Remove the carpet to gain access to the forward wing saddle access panel, and remove that panel per instructions in Chapter 25.
3. Remove the center console access panels per instructions in Chapter 25.
4. Disconnect all wiring to the selector, and mark and label connectors.
5. Note and record the position of the fuel selector knob in relation to the left or right tank detents. All the couplings and the splined shaft in this assembly have alignment markings comprised of either dimples or 1/8" tall letters. If the alignment markings are visible note and record their position. If the alignment markings are not visible mark each shaft connection so that alignment will be maintained during reassembly. See Figure 28 - 34.
6. Remove the four countersunk 6-32 screws, self-locking nuts and washers attaching the upper fuel selector plate to the center console.
7. Slide the upper portion of the fuel selector assembly off the splined shaft. See Figure 28 - 34.
8. Remove the two truss head screws and large area washers securing the vapor barrier and grommet to the floor. Slide the vapor barrier and grommet off of the spindle and splined shaft. See Figure 28 - 33.

c. **Installation of the Fuel Selector Valve**

1. Wire the fuel selector valve per instructions in the LC42-550FG Electrical Manual.
2. Rotate the spindle selector to the left fuel tank detent, and adjust the microswitch so that the roller face on the activated switch is 1.01 +0.03/-0.05 from the face of the spindle (see Figure 28 - 32). Ensure the left fuel tank annunciator illuminates, and tighten the microswitch attachment screws 3.8 to 4.5 in.-lbs.
3. Rotate the spindle selector to the right tank position, and repeat the previous step.
4. Rotate the selector back to the left tank position.

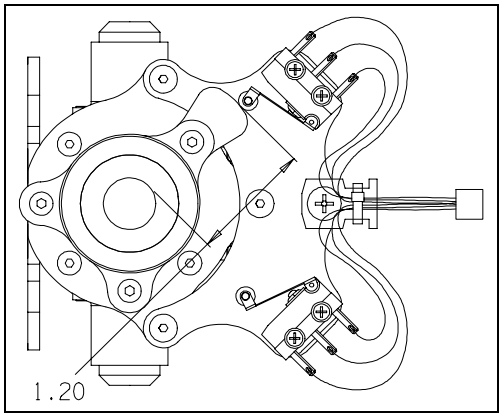

Figure 28 - 32 Fuel Selector Microswitch Position

5. Slide the vapor barrier plate and grommet over the top of the fuel selector spindle, and position the grommet so it is mounted in the vapor barrier plate and over the solid shaft of the selector. While sliding the grommet, it is acceptable to stretch the grommet over the lower universal joint. The vapor barrier plate should be flush with the top of the floor. See Figure 28 - 33 and Figure 28 - 34.
6. Install two 10-32 truss head screws and two large area washers in the two rivnuts installed in the floor as shown in Figure 28 - 33.
7. Slide the upper portion of the fuel selector coupling over the splined shaft, as shown in Figure 28 - 34. After assembly, all alignment marks (existing dimples, existing 1/8" letters, or marks applied during disassembly) must be aligned with the mating part. Alignment marks must also face the same side of the aircraft they faced before disassembly.
8. Attach the upper fuel selector plate to the center console by installing the four countersunk 6-32 screws, self-locking nuts, and washers in the upper fuel selector plate. Torque 4 to 6 in.-lbs.
9. Install the center console access panels per instructions in Chapter 25.
10. Install the co-pilot's side wing saddle access panel, carpet, and co-pilot's seat per instructions in Chapter 25.

d. **General Leak Check for Each Line, Tube, or Fitting –** After any maintenance on the fuel lines or fittings, check each system for leaks with 90 psi air or nitrogen, and apply a soapy solution at each junction for 30 seconds. After a successful no-bubbles leak check, clean and dry the junctions. Apply a sentry stripe at nut to tube, nut to fitting, and fitting to mounting surface.

e. **System Check –** Follow the instructions in section 28-7 to assure system integrity.

f. **Maintenance Practices**

1. Periodic inspection of the fuel lines, hoses, fittings, etc., is required to check for leaks, chafing, or damage.
2. Cleaning is required on top of and around the fuel selector linkage by vacuuming or wiping with a damp cloth to clean away dust and debris that may have worked into the area.

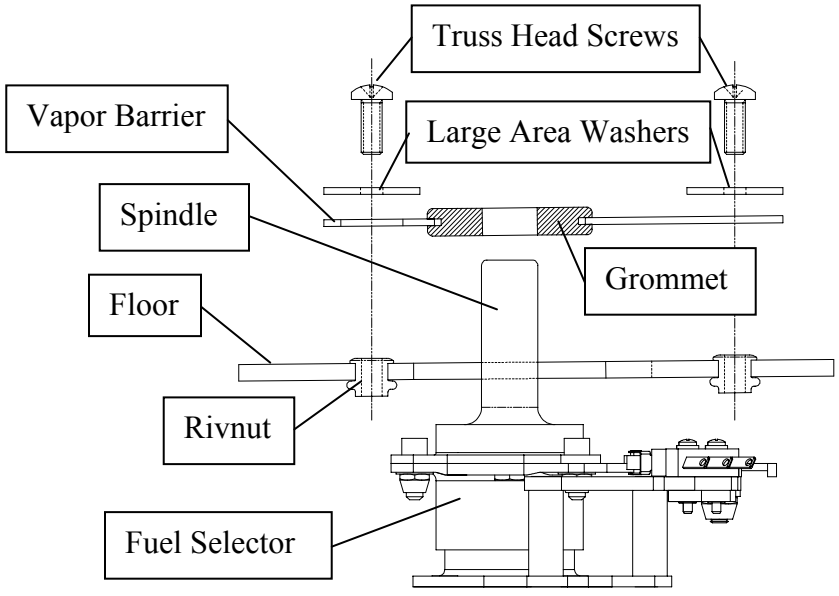

Figure 28 - 33 Fuel Selector Assembly Viewed from Passenger Seat

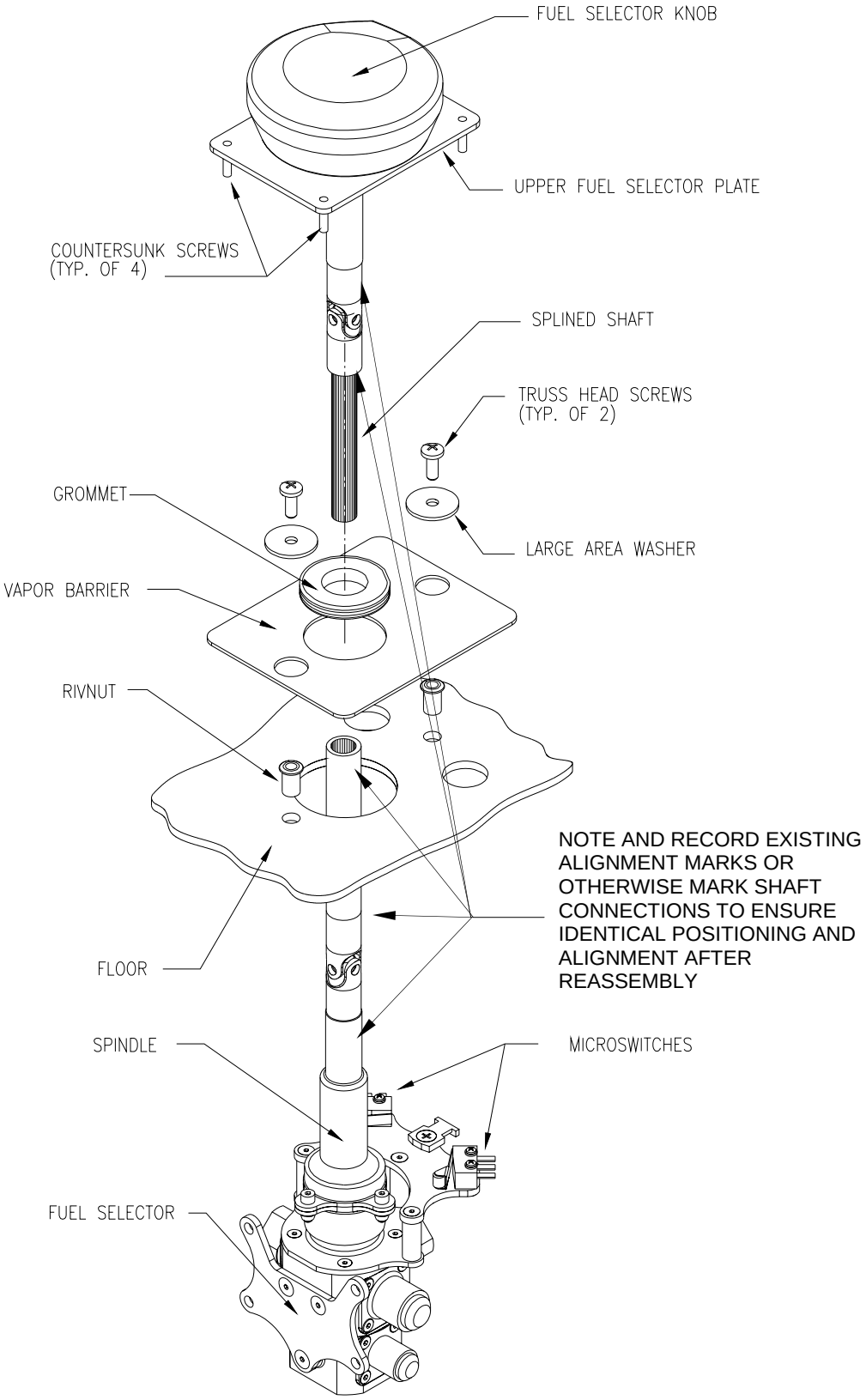

Figure 28 - 34 Fuel Selector Assembly

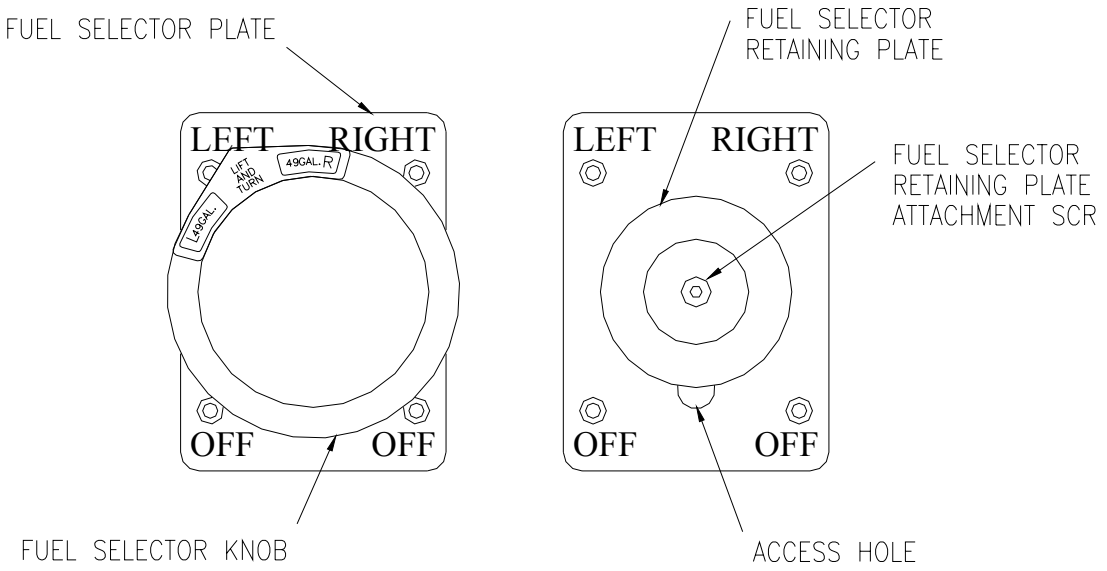

Figure 28 - 35 Fuel Selector Knob

g. **First 500 Hour Inspection and Every 1000 Hours Thereafter**
   1. Ensure valve rotates freely to all positions.
   2. With power applied to the aircraft, ensure that annunciators on instrument panel indicate correct selector position.
   3. Remove upper knob and ensure socket-head, countersunk screw is tight. If not, remove screw, apply a drop of thread-locking compound (Loctite 242 or equivalent), and reinstall. Tighten until snug.
   4. Inspect interconnect shaft between upper detent plate and the valve body. Check that roll pins are securely in place and that there is no sign of chafing or excessive wear.
   5. Check lobed plate on valve body and verify it actuates microswitches as selector is moved to the various positions. Replace broken or worn microswitches.
   6. Inspect valve body for any signs of leaking. If leakage is evident, check security of fuel line connections. Correct any loose connections. If no loose connections are found, remove valve body and repair or replace.

28-16. AUXILIARY FUEL PUMP

The Cessna 350 is equipped with an electric auxiliary fuel pump to provide both vapor suppression and backup boost pump operation. The pump is located on the fuel module mounted to the forward spar, below the front seats. Access is through the fuel system access panel located on the belly of the aircraft.

b. **Removal of Auxiliary Fuel Pump**
   1. Verify aircraft power is turned off.
   2. Turn the fuel selector to the off position.
   3. Disconnect the auxiliary fuel pump wire connector.
   4. Disconnect the fuel lines.
   5. Loosen the hose clamps, and move clamps to release the fuel pump.

b. **Installation of Auxiliary Fuel Pump**
   1. Installation is a direct reversal of the removal of the auxiliary fuel pump.

2. Hand tighten the flared tube fittings connecting the fuel lines to the fuel pump until the assembly is complete and then tighten 150-250 in.-lbs.
3. Prior to tightening the hose clamps around the fuel pump ensure that the fuel lines are not pre-stressed. Tighten hose clamps 10-15 in.-lbs.

28-17. PRIMER SWITCH

a. **Removal and Installation**
   1. See Chapter 33 of this manual for detailed instructions.

b. **Maintenance Practices**
   1. See Chapter 33 of this manual for detailed instructions.

28-18. AUTOMATIC FUEL BOOST PUMP SYSTEM

a. **System Overview** – The purpose of the automatic fuel boost pump system is to reduce pilot workload in case of a primary engine driven fuel pump failure during takeoff. The system is operated by arming the backup pump system prior to takeoff and during climb to cruise altitude. Once armed, the system will turn on the backup pump if it senses a pressure drop in the unmetered fuel line. The system provides at least 79% power, or 246 hp, within two seconds of an engine driven fuel pump failure as tested by Teledyne Continental Motors (TCM). Figure 28 - 36 Automatic Fuel Boost Pump System Overview, shows the overview of the automatic boost pump system and the relationship of the primer switch, the vapor suppression switch, and the boost pump arm switch to the boost pump.

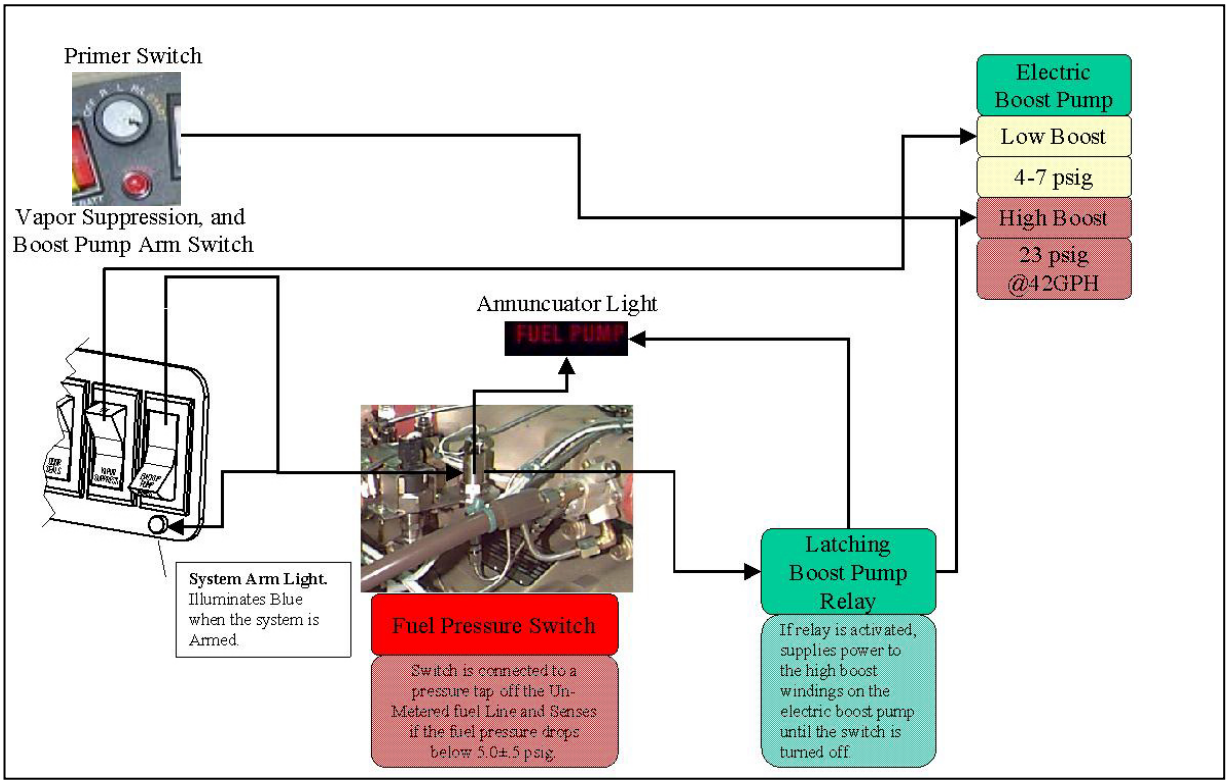

Figure 28 - 36 Automatic Fuel Boost Pump System Overview

The system includes a fuel line, boost pump pressure switch, a warning light, system-armed light, system-arming switch, and the wires and connectors to the auxiliary fuel pump. (See section 28-16.) The fuel line is located between the T-fitting on the TCM IO-550-N throttle metering unit (TMU) and the boost pump pressure switch. (See Figure 28 - 37 or Figure 28 - 38.)

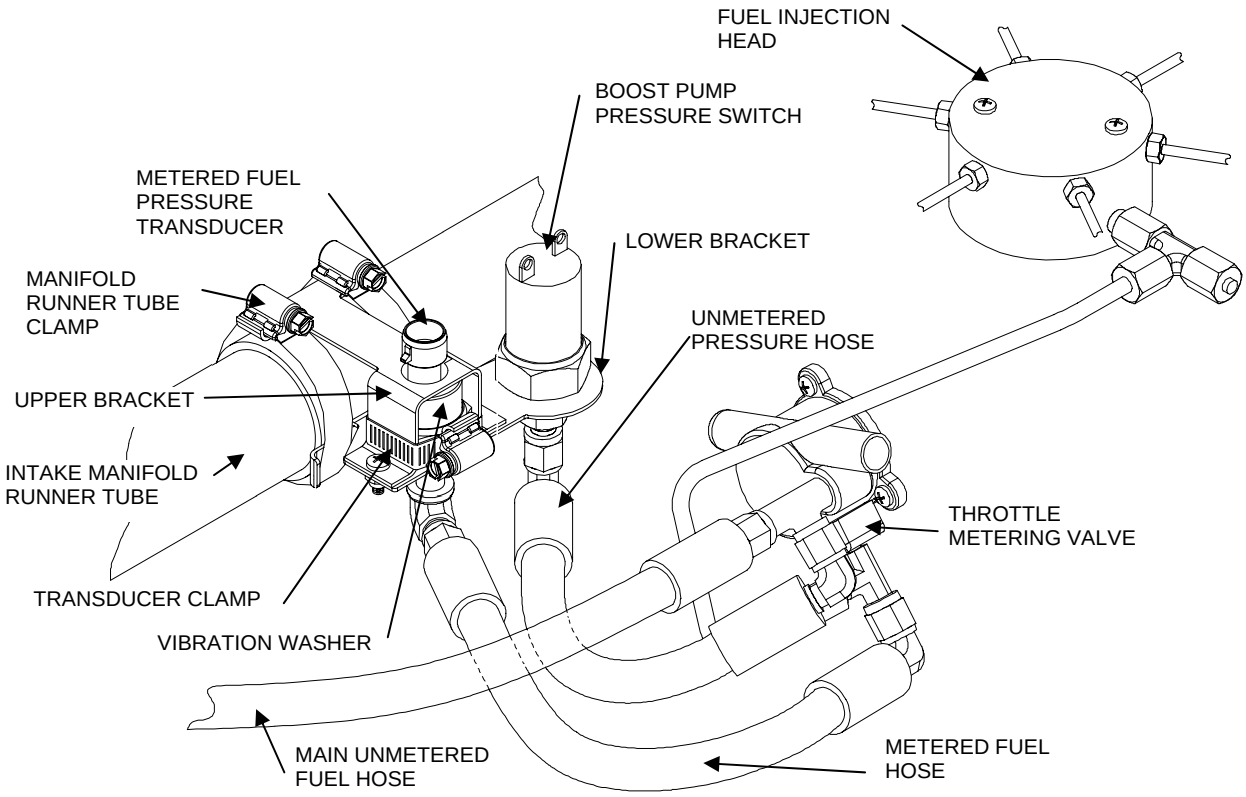

Figure 28 - 37 Boost Pump Pressure Switch Installation (Basic or Avidyne Option)

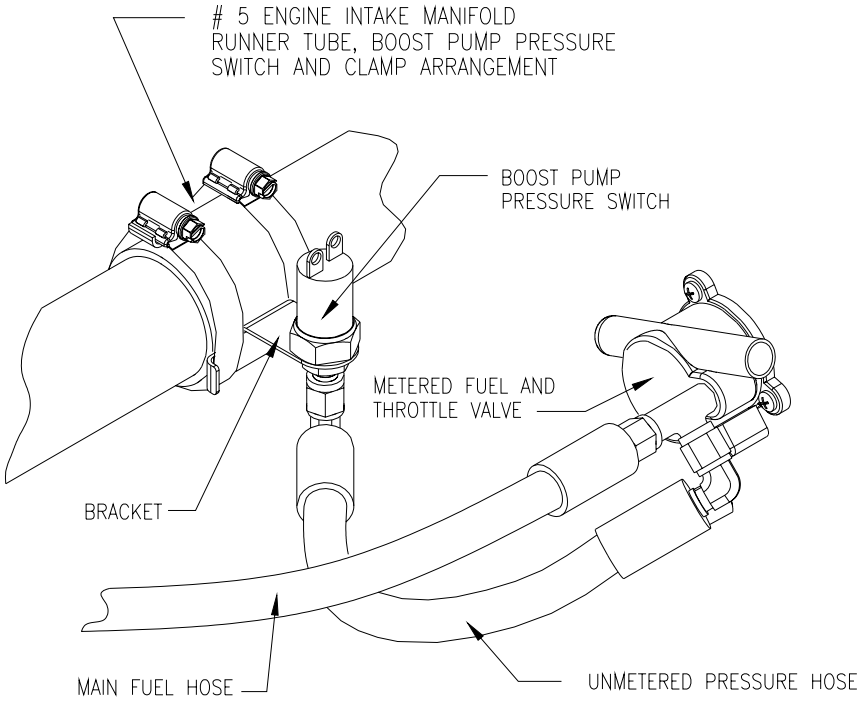

Figure 28 - 38 Boost Pump Pressure Switch Installation (Garmin G1000 Option)

The system activated by the back-up pump switch will illuminate the system-armed light located directly below the switch. The boost pump pressure switch senses unmetered fuel pressure coming from the engine's fuel injection system by means of a fuel line connected directly between the fuel pressure switch and the fuel injection metering valve on the unmetered fuel pressure side. When the fuel pressure drops below 5.5 ± 0.5 psi and the back-up pump switch is in the armed position, the switch operates a latching relay, turning on the boost pump and illuminating the fuel pump annunciator. If the system is off, the annunciator light will still illuminate when the unmetered fuel pressure is below 5 psig.

For the Avidyne option, Figure 28 - 39 shows the location of the cockpit controls for the fuel pump and automatic fuel pump system.

For the Garmin G1000 option the vapor suppression switch and backup pump arm switch are located in the flap panel. The primer button is located adjacent to the ignition switch.

Figure 28 - 39 Fuel System Cockpit Switches (Basic or Avidyne Option Only)

b. **Maintenance Practices**

1. System Check Out

a) Verification of Normal Operation

If the operation of the automatic boost pump system is in question, a ground test, at a minimum, should be conducted prior to the next flight. This includes an engine warm-up and run-up test to verify that the system is operating correctly. If the system fails the ground test, or the system is still in question, further system checks must be performed.

b) Ground Testing

For the ground tests, an indication different from what is defined as normal constitutes a test failure, and testing should be stopped. In case of a system failure, check the pressure switch, follow the electrical troubleshooting section, or contact the factory. This test requires starting the engine, and due to the nature of the testing, may cause flooding of the engine. Proper safety precautions must be taken to prevent fire and damage to personnel, the system, or the aircraft.

WARNING

Testing of the backup boost pump system requires engine runs, running the fuel boost pump, and priming the engine. As with any work around aircraft, the engine must be assumed to be ready to run and able to turn over at any time. Precautions must be taken to prevent personal injury and damage to aircraft and surroundings.

1) Turn the master switch on. With the engine not running, the FUEL PUMP annunciator light (basic or Avidyne option) or display in the PFD Annunciations window (Garmin G1000 option) must be illuminated.
2) With the throttle partially open and the mixture full rich, press the primer button until an indication is registered on the fuel flow/pressure gauge, on the Avidyne Entegra PFD, or Garmin G1000 MFD. When flow indicated is above 0 GPH, the FUEL PUMP annunciator light or Garmin PFD annunciation display must be extinguished.
3) Return the throttle and mixture to the full aft position, (mixture to idle cut-off and throttle to closed).
4) Turn the back-up pump switch to ARMED. The system armed light must be illuminated, the fuel pump must be running, and the FUEL PUMP annunciator light or Garmin PFD annunciation display must be illuminated. Turn the switch to off after verification of the above items.
5) Start the engine and allow the engine to warm up prior to performing any engine run-up tests. During startup, the FUEL PUMP annunciator light or Garmin PFD annunciation display must extinguish once the engine starts.

CAUTION

At this point in the backup boost pump system testing, the engine should be well primed, if not flooded. Care must be taken to prevent damage to personnel, the engine, and the aircraft during starting. Refer to the Pilot's Operating Handbook for starting the engine with a flooded engine.

6) With the engine running and with engine temperatures in the green, turn on the backup boost pump system. Only the system-armed light should illuminate, and the boost pump should not operate. This can be verified by watching the fuel pressure/flow gauge for any fluctuations when the system is turned on and off.
7) With the system armed, prop set at full forward (max RPM), and engine running at 2000 RPM, slowly pull the mixture back until the indicated fuel flow is approximately 10 GPH. Continue pulling the mixture back until the boost pump system turns on the boost pump. If the pressure drops below 0 GPH and/or the engine quits, the system is not operating correctly. If the mixture is at idle cut-off, the system did not engage as indicated by the FUEL PUMP annunciator light or Garmin PFD annunciation display, and the engine is still running, the engine fuel system must be re-calibrated per Chapter 73. Return the mixture to the previous setting when completed with this test point.

NOTE

If the mixture is pulled completely back to the idle cut-off position, the system may not be able to provide enough fuel pressure to run the engine.

8) With the system armed, perform a full power run-up with a quick retraction of the throttle to idle. The system should not operate as indicated by the

illumination of the FUEL PUMP annunciator light, or Garmin PFD annunciation display, or engaging of the electric boost pump.

9) Turn off the system, and shutdown the engine. If no failures are noted, the system is considered safe for flight. If the system is in question, re-test, or contact the manufacturer prior to flight. In case of a system failure, check the electrical system as documented in the following section.

2. Electrical Troubleshooting
   a) Pump does not turn on with zero fuel pressure and the back-up pump switch is armed.
      1) Fuel Pressure Sensor
         i) Verify with an ohmmeter that one of the terminals has a good ground (0-1 ohms) and mark it GND.
         ii) With the aircraft power on and the back-up pump switch armed, verify with a voltmeter that approximately 12 volts (S/N 42001 to 42500) or 24 volts (S/N 42501 and on) is on the other terminal.
         iii) Jumper the two terminals on the fuel pressure sensor, and immediately place the back-up pump switch in the arm position.
         iv) If the pump runs with the fuel pressure sensor terminals jumped, replace the fuel pressure sensor.

CAUTION

Do not run the pump more than 2-3 seconds.

      2) Back-up Pump Rocker Switch
         i) With aircraft power on, move the rocker switch to the armed position.

CAUTION

Do not run the pump more than 2-3 seconds.

         ii) If the light does not illuminate but the pump turns on, replace the "armed" light.
         iii) If the light does not illuminate and the pump does not turn on, depress the primed button briefly.
            1. If the pump turns on, replace the rocker switch.
            2. If the pump does not turn on, replace the circuit breaker.
      3) K-7 Latching Relay (located on right side of fuselage behind the right kidney panel on the instrument panel for the Avidyne option or located behind the instrument panel to the right of the MFD and LRUs for the Garmin G1000 option)
         i) With the fuel pressure sensor terminals jumpered, move the back-up pump switch to the armed position. With your hand on the relay, you should feel the relay actuate. If not, replace the relay.
      4) Boost Pump
         i) With an ohmmeter, verify that the pump has a good ground at pin 3 of the disconnect plug (0-1 ohms).
         ii) With a voltmeter, verify that 12 volts (S/N 42001 to 42500) or 24 volts (S/N 42501 and on) is on pin 2 of the disconnect plug while the vapor suppress switch in on.

iii) Verify that 12 volts (S/N 41002 to 41500) or 24 volts (S/N 41501 and on) is on pin 3 of the disconnect plug while the back-up switch is in the armed position.
iv) If all three above checks are OK, replace the pump. Refer to section 0 for installation and removal instructions.

5) Vapor Suppression is inoperative (Low Speed)
i) Check for operation of pump's low-speed operation with the back-up pump switch in the off and armed position.
1. If the low speed only works with the back-up pump switch off, check for voltage on (± 12 volts for S/N 42001 to 42500 or ± 24 volts for S/N 42501 and on) both sides of the power diode (located on the instrument panel under the terminal junction block). If there is only voltage on one side, replace the diode.
2. If the low speed only works with the back-up pump in the armed position replace the back-up pump switch.

c. **Removal and Installation**

1. Boost Pump Pressure Switch Removal (See Figure 28 - 37)
a) Verify aircraft power is turned off.
b) Verify the mixture is in the idle cutoff position and the engine is not primed.
c) Turn the fuel selector to off.
d) Remove the upper engine cowling.
e) Disconnect the two wire leads from the boost pump pressure switch.
f) Disconnect the pressure line from the boost pump pressure switch.
g) Remove the jam nut, lock washer, and pressure switch.

2. Boost Pump Pressure Switch Installation (See Figure 28 - 37)
a) Installation is a direct reversal of the removal of the boost pump pressure switch.
b) Torque the nut 240 to 280 in.-lbs.
c) Reconnect the unmetered pressure hose to the bottom of the switch. Torque the fitting 75 to 80 in.-lbs.
d) Reconnect the electrical connections to the top of the switch.

3. Fuel Pump Removal and Installation.
a) See section 28-16 for removal and installation instructions.

4. Electrical Components
a) Power Diode Removal (for the basic or Avidyne option see Figure 28 - 40 or for the Garmin G1000 option, the diode is located in the wiring cluster near the boost pump switch)
1) For the basic or Avidyne option, remove the glare shield and right side kidney panel. For the Garmin G1000 option, remove the flap panel.
2) Cut the wires from the diode, and mark the cut wires.
3) Remove the diode, and replace with a new diode.
b) Power Diode Replacement
1) Install diode.
2) Slide heat shrink over the wire, and using good solder practices, solder and heat shrink the wires.
3) For the basic or Avidyne option, install the right side kidney panel and glare shield. For the Garmin G1000 option, install the flap panel.

4) Test the system by turning on the vapor suppress (Low Speed), and the pump should stay on when the back-up pump is activated.

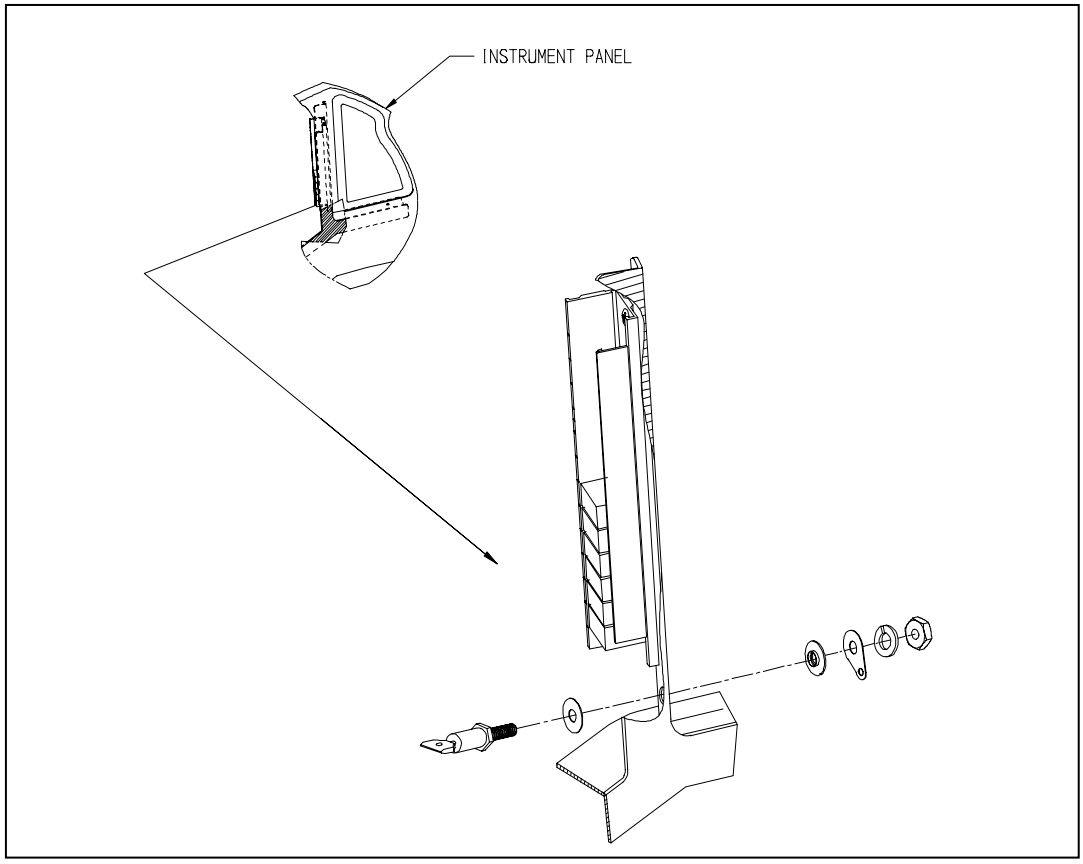

Figure 28 - 40 Power Diode Location and Installation

c) Arm Light Removal
   1) Remove the left side knee bolster.
   2) Loosen the switch plate, and pull out from the instrument panel 2-3 in.
   3) Remove the old light after cutting the wires.

d) Arm Light Installation
   1) Install a new light, and crimp the terminals on after inserting through the mounting hole.
   2) Re-install the switch plate and left knee bolster.
   3) Test the arm light operation by momentarily arming the back-up pump switch.

e) K-7 Latching Relay Removal
   1) For the basic or Avidyne option, remove the glare shield and right side kidney panel. For the Garmin G1000 option, access the relay from underneath the instrument panel.
   2) Carefully mark the wires to the relay.
   3) Remove the relay.

f) K-7 Latching Relay Installation
   1) Replace the relay, paying close attention to the orientation of the relay.

2) Re-attach the wires to the terminals.
3) Replace removed items and glare shield, as applicable.
4) Functional test the entire back-up pump system per section b.1.

g) Rocker Switches Removal
1) Remove the left side knee bolster.
2) Loosen the switch plate, and pull out from the instrument panel 2-3 inches.
3) Carefully mark the wires to the switch.
4) Remove the defective switch from the metal plate.

h) Rocker Switches Installation
1) Install the replacement switch, and connect the wires to the terminals.
2) Reinstall the switch plate and left side knee bolster.
3) Functional test the entire back-up pump system per section b.1.

28-19. FUEL GASCOLATOR

a. **Removal and Installation**

1. Normal use and maintenance does not require removal of the gascolator from its mounting. If removal and replacement of the whole gascolator is required, contact the manufacturer.
2. If maintenance practices do not maintain correct functioning of the gascolator, contact the manufacturer for removal and replacement instructions.

b. **Maintenance Practices**

1. Periodic cleaning of the filter is required for proper operation of the fuel system as indicated in Chapter 5 of this manual.

c. **Removal of the Gascolator Bowl and Filter**

1. Turn the fuel selector to the off position.
2. Drain any residual fuel from the gascolator.
3. Remove the safety wire securing the gascolator bowl mounting ring.

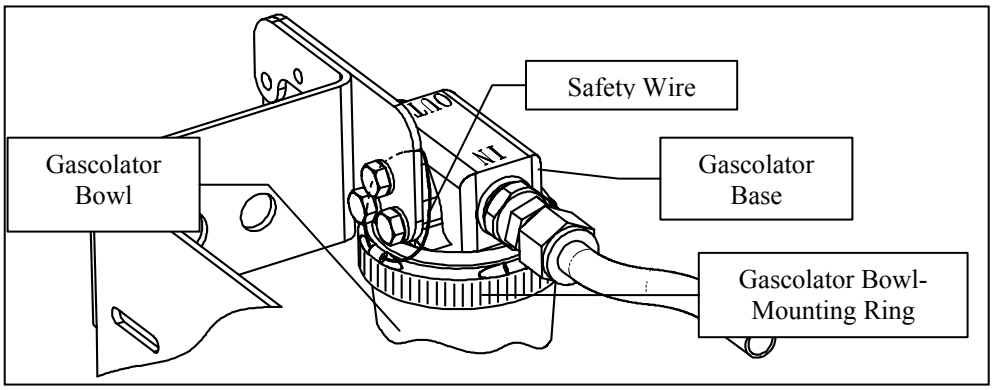

Figure 28 - 41 Gascolator Safety Wiring and Mounting

4. Turn the ring clockwise, as seen from the bottom, until the ring is unthreaded. Note the alignment of the arrow on the bowl to the line on the gascolator base.

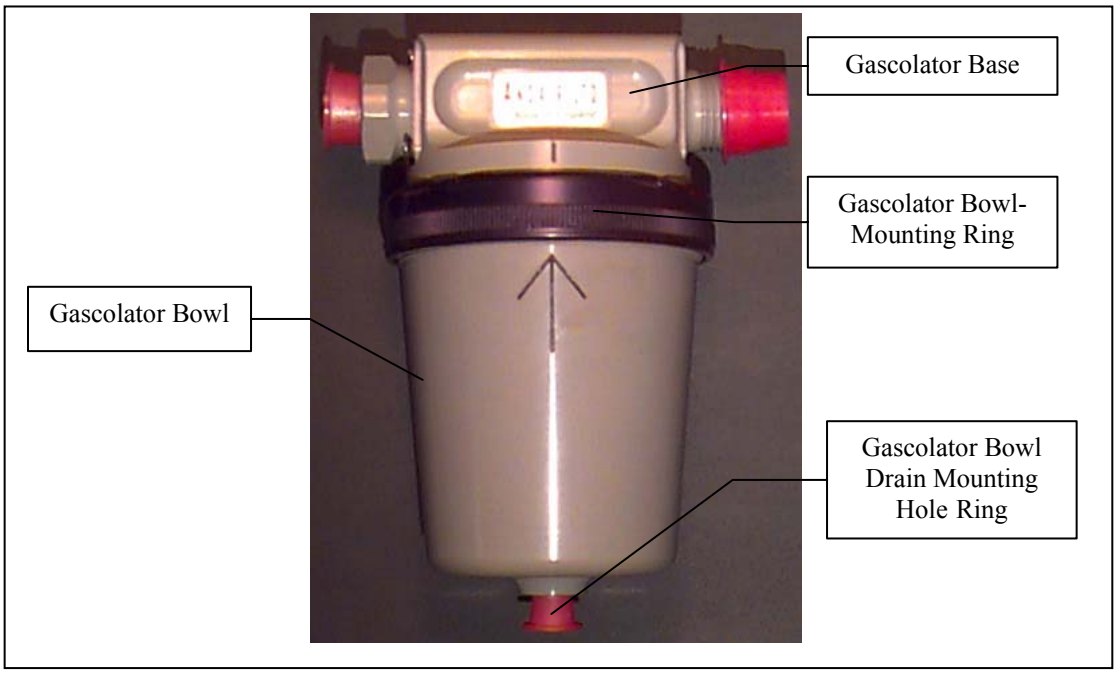

Figure 28 - 42 Gascolator Detail

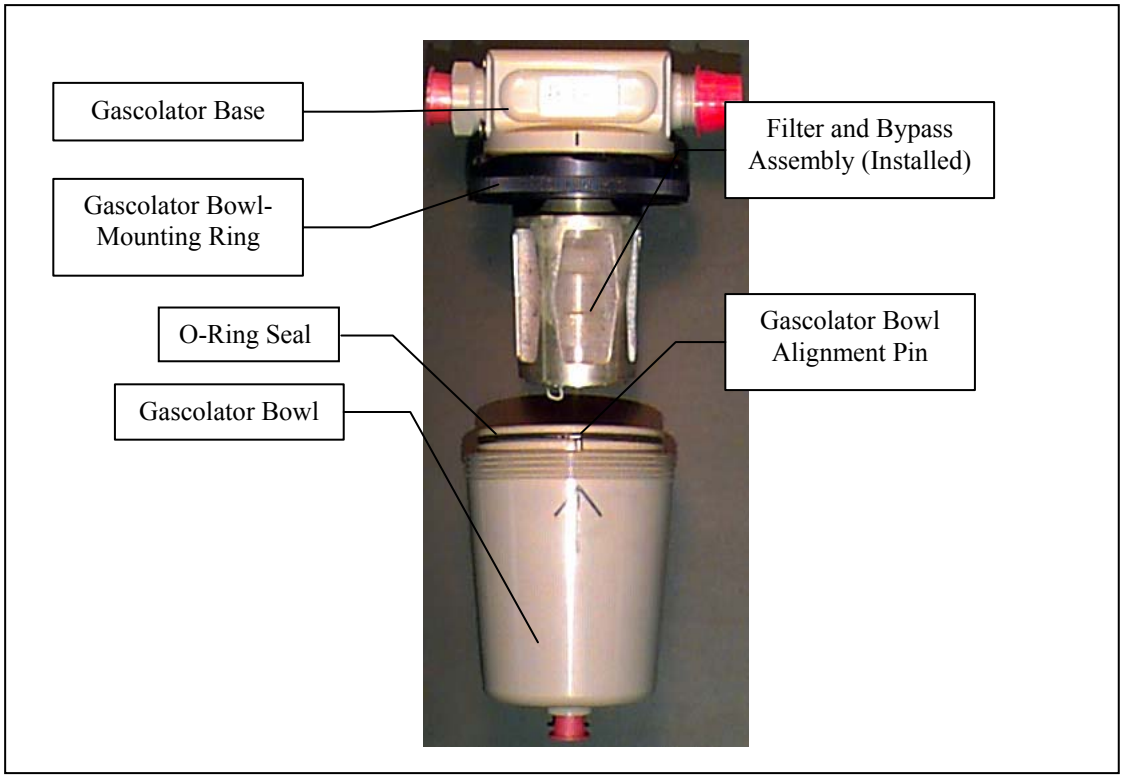

Figure 28 - 43 Gascolator Detail

5. Pull gently straight down on the bowl to remove and access the filter/bypass element.
6. Unscrew the filter/bypass element, and carefully clean the filter and bypass. Check for smooth operation of the bypass.
7. Clean the gascolator bowl, and inspect the O-ring seal. Replace the seal as necessary. Contact the manufacturer for replacement O-rings.
8. Reassembly is a direct reversal of removal. Note the pin on the top of the gascolator bowl must fit into the groove of the gascolator base.

d. **Fuel Strainer Internal Bypass and Filter Replacement**

1. If the internal bypass/filter requires replacement, contact the manufacturer.

28-20. INDICATING

a. **Basic or Avidyne Option –** The fuel system indicating system on the Cessna 350 includes a fuel level gauge, low fuel level warning lights, fuel tank selector lights located in the fuel level gauge, a fuel selector warning light, and fuel system information display on the Avidyne Entegra EX5000 Multi-Function Display (if installed on the aircraft). The fuel level is measured using two tank-mounted float level sensors, one in the inboard fuel tank bay, and one in the outboard fuel tank bay. The low fuel level sensors are float switches mounted approximately through the middle of the inboard fuel tank rib at WS 46.0. The fuel selector valve indicator lights use micro-switch sensors mounted on the top of the fuel selector.

b. **Garmin G1000 Option –** The fuel system indicating system displays on the Garmin G1000 MFD. The fuel level is measured using two tank-mounted float level sensors, one in the inboard fuel tank bay, and one in the outboard fuel tank bay. The low fuel level sensors are float switches mounted approximately through the middle of the inboard fuel tank rib at WS 46.0. The fuel selector valve indicator lights use microswitch sensors mounted on the top of the fuel selector.

28-21. FUEL QUANTITY GAUGES AND FUEL ANNUNCIATORS (Basic or Avidyne option only)

a. **Maintenance Practices**

1. See Chapter 33 of this manual for detailed instructions for electrical system maintenance practices.
2. Mechanical maintenance is not required unless errors in the system are suspected. Inspect and replace defective fuel sensors as necessary per instructions in section 28-5.

b. **Removal and Installation**

1. See Chapter 33 of this manual for detailed instructions.

28-22. FUEL VALVE INDICATOR LIGHTS

a. **Description and Operation**

1. The fuel valve is equipped with two micro-switches to sense the position of the fuel selector. The selected tank is displayed on the fuel level gauge with a green LED (basic or Avidyne option) or displayed by a blue light on the MFD (Garmin G1000 option) showing the selected fuel tank. If no green LED or blue light, as applicable, is illuminated, the selector is off the tank, in-between tanks, or in the off position. During these conditions, a FUEL VALVE warning light illuminates (basic or Avidyne option) or a warning annunciation displays (Garmin G1000 option).

b. **Testing**

1. Basic or Avidyne option
   a) With the master switches on (batteries only if the engine is not running) place the fuel selector in the left tank detent. The fuel warning light should be extinguished, and the green light on the fuel level gauge should be illuminated.
   b) Repeat for the right tank.
   c) Verify that both green lights, as applicable, are extinguished and the fuel warning light is illuminated when the valve is in between tanks or in the off position.
   d) Contact the factory if the indicator lights are not functioning properly.

2. Garmin G1000 option
    a) With the master switches on (batteries only if the engine is not running) place the fuel selector in the left tank detent. The fuel warning annunciation should clear, and the blue light on the MFD should be illuminated.
    b) Repeat for the right tank.
    c) Verify that both blue lights are extinguished and the fuel warning annunciation is displayed when the valve is in between tanks or in the off position.
    d) Contact the factory if the indicator lights are not functioning properly..

28-23. LOW FUEL ANNUNCIATORS (LEFT AND RIGHT TANKS)

a. **Removal and Installation** – Fuel Tank Low Fuel Level Floats
    1. If removal of the low fuel level switch is required, contact the manufacturer for further instructions.

b. **Testing Low Fuel Annunciator**
    1. With less than six gallons of usable fuel in the fuel tank, and the aircraft level, the low fuel warning annunciator should be illuminated for the tank with the low fuel quantity.
    2. During fueling, the low fuel level warning should extinguish prior to 10 gallons usable fuel being in the tank.
    3. If a low fuel level sensor is suspected, replace per instructions in section 28-5.
    4. Visually inspect the low fuel level float switch or visible damage and freedom of movement.
    5. If a faulty switch is suspected, see Chapter 33 of detailed instructions.

c. **Replacing Low Fuel Annunciator Lights**
    1. See Chapter 33 of this manual for detailed instructions.

CHAPTER 30

ICE AND RAIN PROTECTION

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Chapter 30

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30-1. PROPELLER HEAT/DE-ICE – GENERAL

The Propeller De-Ice system is intended to remove ice from the propeller blades of the aircraft. The system is intended as an aid during inadvertent flight in icing conditions. The system is not intended to allow flight into known icing conditions.

The Propeller De-Ice system is comprised of the propeller blade electric heater boots, slip ring assembly, propeller heat control module, brush block and bracket assembly, control switch with integral annunciator, 30 amp fuse on the power grid, and miscellaneous covers, hardware, and wire. See Figure 30 - 1.

The pilot controls the system by the Prop De-Ice Control Switch. After setting the Control Switch to the 'ON' position the controller will conduct a self test and then illuminate the annunciators 'ON' and 'PROP' in green. The system operates on a 90 second on, 90 second off, electrical cycle.

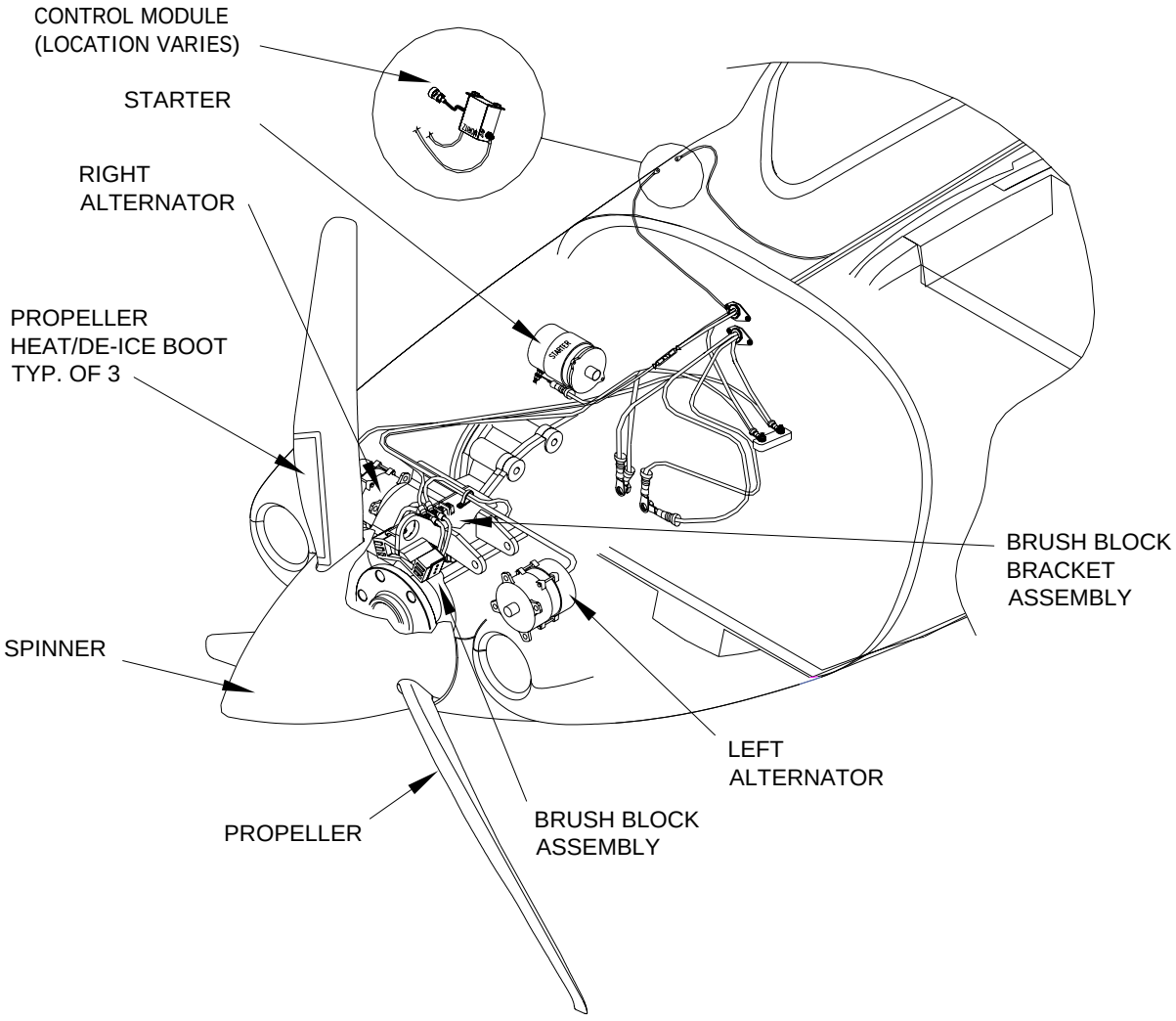

Figure 30 - 1 Propeller Heat/De-Ice System

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30-2. BRUSH BLOCK

a. **Brush Block Removal (See Figure 30 - 2)**

1. Remove the top engine cowling per Chapter 71.
2. Remove two insulated terminal screws and star washers, and disconnect the brush leads from the insulated terminal of the brush block bracket.
3. While holding the brush block, remove the two screws, washers, and spacers attaching the brush block to the brush block bracket.
4. Slowly pull the brush block away from the slip ring to relieve pressure on the brushes.
5. Remove the brush block taking care not to damage the brushes.

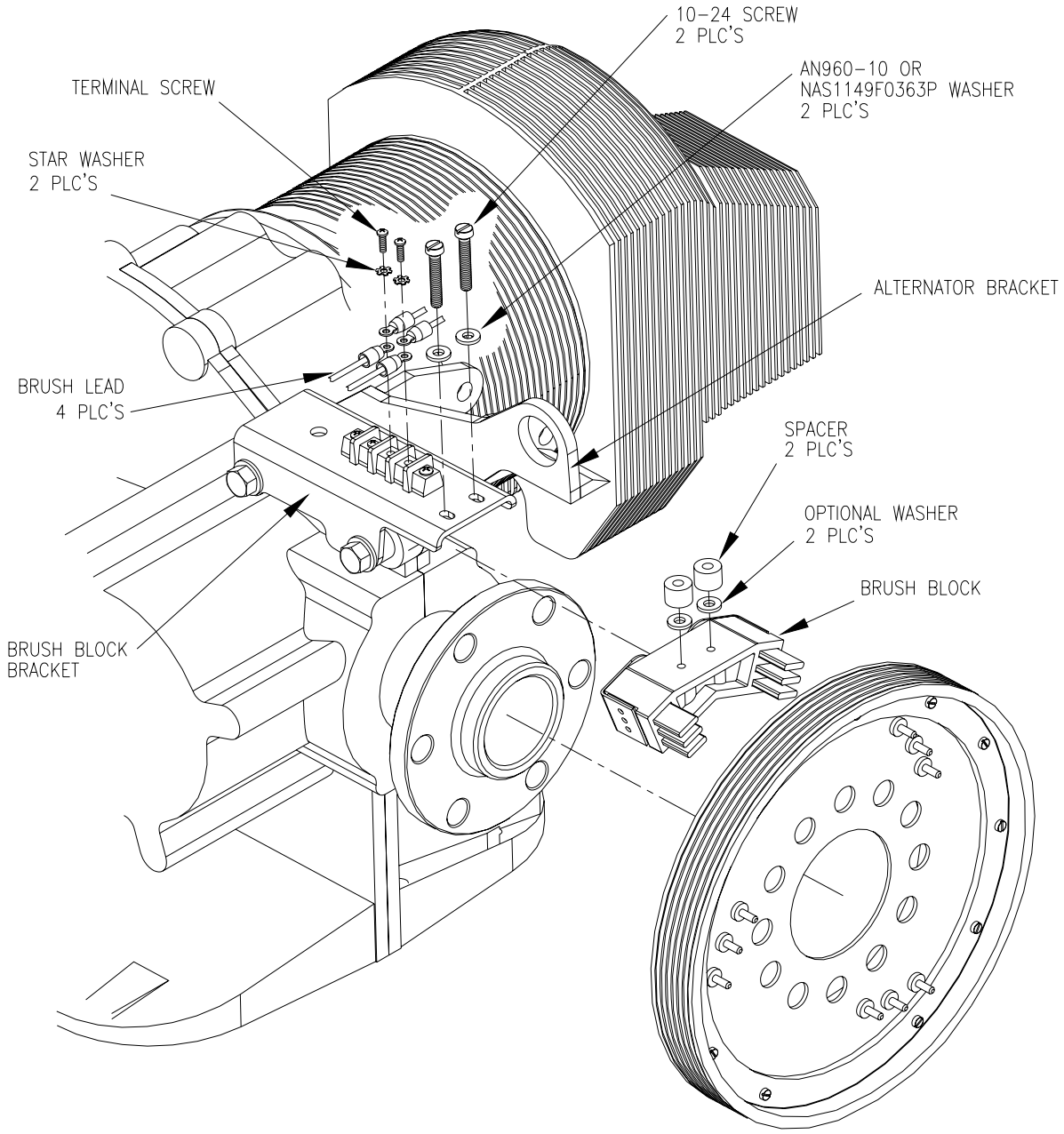

Figure 30 - 2 Brush Block

b. **Brush Block Installation (See Figure 30 - 2)**

1. Press the brushes into the brush block.

CAUTION

Use care when handling the brushes. Skin oils may contaminate the brush material and reduce brush service life.

2. With the brushes compressed into the brush block, wrap small tie straps around the brush block to hold the brushes in place. See Figure 30 - 3.

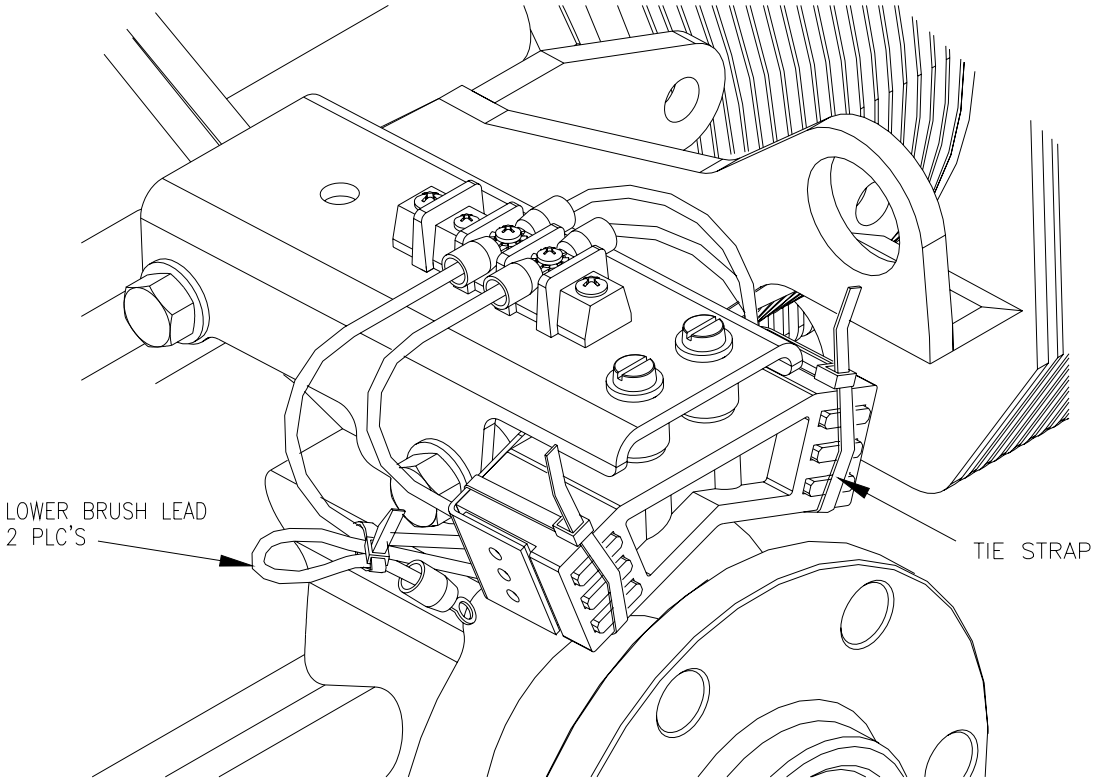

Figure 30 - 3 Brush Block Tie

3. Attach the brush block to the brush block bracket using two AN-503-10-20 screws, AN960-10 or NAS1149F0363P washers, and Northcoast DI0029 spacers. Leave the attachment screws just loose enough to allow movement of the brush block assembly fore and aft with minimal effort.
4. Attach the wire terminal ends for the top brushes to the forward insulated terminal screw and for the middle brushes to the middle insulated terminal screw. See Figure 30 - 4. Place star washers under the head of the screws and tighten snug. The bottom brush leads are not used and should be coiled and secured using tie straps.

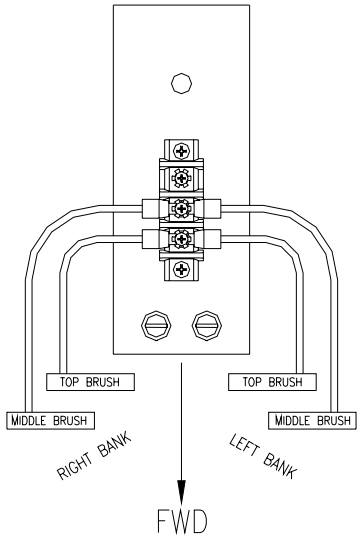

Figure 30 - 4 Brush Wire Routing

5. Clean the slip rings using MEK, acetone, or spray degreaser.
6. View the alignment of the brushes and the slip ring from the side. The brush should ride in the center of the slip ring. See Figure 30 - 5. If alignment is necessary, install NAS1149F0316, NAS1149F0332, or NAS1149F0363 washers between the spacer and brush block to achieve correct alignment. Different thickness washers may be stacked together.

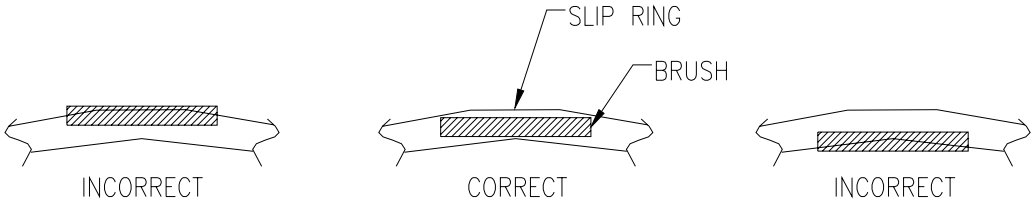

Figure 30 - 5 Brush Alignment on Slip Ring

7. Cut and discard the tie straps holding the brushes into the brush block to allow the brushes to contact the slip ring. Press the brush block up to the slip ring and tighten the mounting screws to allow the brush block to be adjusted with moderate force. Adjust the brush block so the left side has approximately .06 inch clearance and the right side has approximately .09 inch clearance when viewed from the front of the engine looking down. See Figure 30 - 6. Tighten the mounting screws snug and secure with .032 safety wire per MS33540.

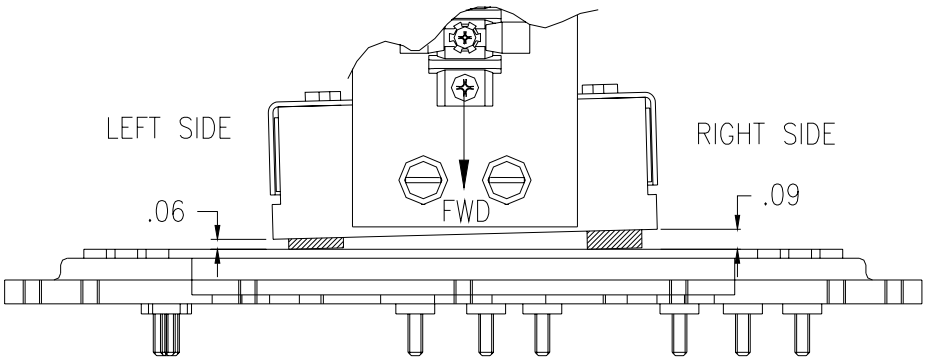

Figure 30 - 6 Brush Block Alignment

c. **Brush Block Maintenance** – Inspect the brush block during the 100 hr and/or annual inspections. There are no servicing requirements for the propeller de-ice system outside normal inspection intervals or during routine maintenance.
   1. Inspect for grease, fuel, oil, or carbon deposits on the brushes and brush block. Clean with methyl ethyl ketone (MEK) or replace components if unable to be cleaned.
   2. Measure brush wear. Replace the brushes if wear exceeds 50% or if less than 5/16 in. of brush material remains.
   3. Inspect for uneven wear, chipping, or broken brushes. Replace components as necessary.
   4. Verify brush to slip ring alignment per paragraph 30-2.

d. **Brush Block Repair** – The brushes are non-repairable and should be replaced if found to be defective.

30-3. SLIP RING

a. **Slip Ring Removal**
   1. Remove the upper a lower cowling per Chapter 71.
   2. Remove the brush block per paragraph 30-2.
   3. Remove the spinner, propeller, and left alternator drive pulley per Chapter 61.
   4. Remove the slip ring.

b. **Slip Ring Installation.**
   1. Place the slip ring on the engine crankshaft flange.
   2. Install the left alternator drive pulley, propeller and spinner per Chapter 61.

c. **Slip Ring Maintenance**
   1. Clean the slip ring with MEK or other suitable solvent.
   2. Visually inspect the slip ring for excessive wear, grooves, rough surfaces, cracks, burned or discolored areas, and oil or dirt deposits. Inspect for any debonding of the copper rings from the slip ring plate.
      a) If uneven wear or wobble is noted, verify that slip ring run-out is less than 0.005 in.
      b) Grooves should be less than 0.007 in. deep.
      c) Small defects can be removed with crocus cloth. The rings should have a minimum 32 micro finish.

d. **Slip Ring Repair –** The slip ring can be machined to achieve run out and surface finish requirements. The minimum slip ring total thickness is 0.437 in. The Maximum allowable slip ring run out is 0.005 in. Surface finish should be 32 micro-inch or less.

CAUTION

Excessive heat build-up during machining can cause the copper ring to delaminate from the epoxy. Use appropriate cutting fluid and feed rates to minimize heat build-up during any machining operation on the slip rings

30-4. BRUSH BLOCK BRACKET

a. **Brush Block Bracket Removal (See Figure 30 - 2)**
   1. Remove the top engine cowling per Chapter 71.
   2. Remove the No. 2 (left) alternator per Chapter 24.
   3. Remove the brush block per Section 30-2.a.
   4. Remove the two AN5-24 bolts, two MS21042L-5 nuts, four AN960-516 washers, and four Northcoast DI-00030 spacers attaching the No. 2 (left) alternator bracket and brush block bracket to the crankcase. See Figure 30 - 2.
   5. Remove the bracket.

b. **Brush Block Bracket Installation (See Figure 30 - 2)**
   1. Install the brush block bracket and No. 2 (left) alternator bracket using two AN5-24 bolts, two MS21042L-5 nuts, four AN960-516 washers, and four Northcoast DI-00030 spacers
   2. Torques nuts 180 to 220 in.-lb.
   3. Install the brush block per instructions in Section 30-2.b.
   4. Install the No. 2 (left) alternator per instructions in Chapter 24.
   5. Install the top engine cowling per instructions in Chapter 71.

c. **Brush Block Bracket Maintenance**– Inspect the brush block bracket for cracks during the 100 hr and/or annual inspections. If cracks are present, replace the bracket. There are no servicing requirements for the propeller de-ice system outside normal inspection intervals or during routine maintenance.

30-5. PROPELLER HEATER BOOT

Propeller heater boots installed on the aircraft are manufactured by SMR Technologies Inc. or Hartzell Propeller Inc. If you are uncertain of the manufacturer of the propeller heater boots installed on the aircraft contact Cessna for guidance.

Remove and install propeller heater boots per instructions in the latest revision of SMR Technologies Report SMR-97-33-013, Propeller De-Icer Installation and Maintenance Manual or Hartzell Propeller Manual 182 (61-12-82), Propeller Electrical De-ice Boot Removal and Installation Manual, as applicable.

Repair or service propeller heater boots per instructions in the latest revision of SMR Technologies Report SMR-97-33-013, Propeller De-Icer Installation and Maintenance Manual or Hartzell Propeller Manual 181 (30-60-81), Propeller Ice Protection System Component Maintenance Manual, as applicable.

Copies of these documents can be obtained or downloaded from the sources that follow:

SMR Technologies Inc.
93 Nettie Fenwick Rd.
Fenwick, WV 26202
Telephone: 304.846.6636
Toll Free: 800.767.6899
Fax: 304.846.6268
Website: www.Iceshield.com

Hartzell Propeller Inc.
Attn: Product support Administrative Assistant
One Propeller Place
Piqua, Ohio 45356-2634
Telephone: 937.778.4200
Fax: 937.778.4391
Website: www.hartzellprop.com

a. **Heater Boot Removal (See Figure 30 - 7 or Figure 30 - 8)**
   1. Remove the propeller spinner per instructions in Chapter 61.
   2. Remove the cowling per instructions in Chapter 71.
   3. Remove the propeller heater boots per the latest revision of SMR Technologies Report SMR-97-33-013 or Hartzell Propeller Manual 182 (61-12-82), as applicable. Make a record of lead wire routing and attachment hardware for reinstallation.

b. **Heater Boot Installation (See Figure 30 - 7 or Figure 30 - 8)**
   1. Inspect the heater boot leads for proper resistance. Resistance must measure 1.56 to 1.72 ohms (14 V) or 4.7 to 4.9 ohms (28 V) at 77º ± 5º F.
   2. Inspect the individual components for damage prior to installation.
   3. Install the propeller heater boots per the latest revision of SMR Technologies Report SMR-97-33-013 or Hartzell Propeller Manual 182 (61-12-82), as applicable.
   4. Ensure the SMR boots are positioned so the heat zone is located 1.7 ± 0.060 inches from the edge of the propeller hub; this will place the edge of the boot approximately 1.25 inches from the edge of the propeller hub. Ensure the Hartzell boots are positioned so the edge of the boot is 1.25 inches from the edge of the propeller hub.
   5. Route and attach the heater boot leads.
      a) SMR Boots
         (a) Route the heater boot leads between the propeller hub and the inside diameter edge of the spinner bulkhead.
         (b) Loosely install the lead clamps using screws, washers, and nuts over the heater boot leads and onto the forward side of the spinner bulkhead.
         (c) To relieve tension caused by centrifugal force, gently pull the heater boot leads up thought the lead clamps far enough that the lead rests against the inside surface of the spinner dome. Tighten the clamp hardware 20 to 25 in-lbs to secure the leads.
      b) Hartzell Boots
         (a) Route the heater boot leads between the propeller hub and the spinner bulkhead. Do not twist the boot lead wires or allow them to be twisted.

(b) Loosely install the loop clamps using screws, washers, and self-locking nuts.
(c) To relieve tension caused by centrifugal force, gently pull the heater boot leads up thought the loop clamps far enough that the lead rests against the inside surface of the spinner dome. Tighten the clamp hardware 22 to 25 in-lbs to secure the leads.
(d) Attach the boot leads and the slip ring wire harness to the bulkhead using bushings, washers, screws, and nuts. Tighten the nuts hand tight plus a quarter turn.

6. Connect the terminal ends of the propeller boot leads (SMR boots) or the slip ring wire harness (Hartzell boots) to the slip ring terminals with 6-32 washers and nuts. See Figure 30 - 9. For each heater lead use the two outer terminals only. The inner terminal is not used. Torque the retaining nuts 10 to 12 in.-lb.

CAUTION

Do not bend or damage the heater leads.

7. Verify that propeller pitch change over the allowable range will not cause binding or stress on the heater leads.
8. Install large tie straps over the three heater boot leads and around the propeller hub. Route the leads over the ridge in the hub. Ensure the leads are not pulled tight against the ridge to prevent chaffing. The leads should have a slight gap over the ridge edges of approximately 0.125 inch. Offset the ends of the tie straps 180 degrees to minimize any imbalance. Tighten the tie straps snug. See Figure 30 - 10 for clarification and for spacing of the tie straps.
9. Install the cowling per Chapter 71.
10. Install the propeller spinner per Chapter 61.

c. **Heater Boot Maintenance**– Inspect the heater boots during the 100 hr and/or annual inspections. There are no servicing requirements for the propeller de-ice system outside normal inspection intervals or during routine maintenance.
   1. Visually inspect the heater boots for wrinkles, loose or torn areas, and/or abrasions or cuts along the leading edge and on the thrust face (flat) of the propeller blades.
   2. Inspect the boot leads for signs of abrasion or damage. Inspect the lead clamps for tightness.
   3. Test the resistance of the heater boot elements. Disconnect the boot leads from the slip ring terminals. Place an ohmmeter across the boot leads. The ohmmeter should indicate 1.56 to 1.72 (14 V) or 4.7 to 4.9 ohms (28 V) of resistance at 77º ± 5º F. If the reading is not in this range replace the heater boot.

d. **Heater Boot Repair –** The heater boots should be serviced and repaired per SMR Technologies Report No. 97-33-013 or Hartzell Propeller Manual 181 (30-60-8812), as applicable.

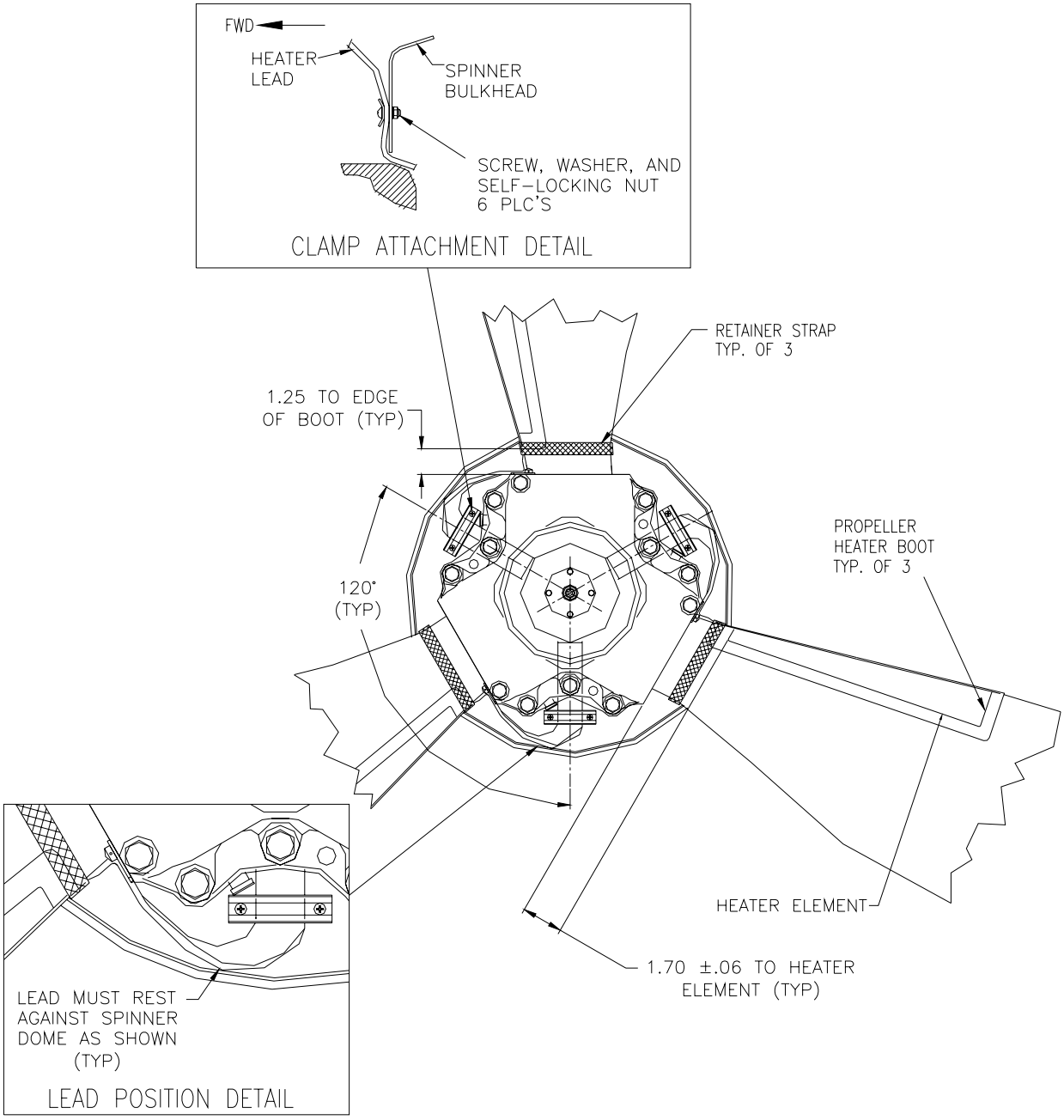

Figure 30 - 7 SMR Technologies Inc. Propeller Heater Boot

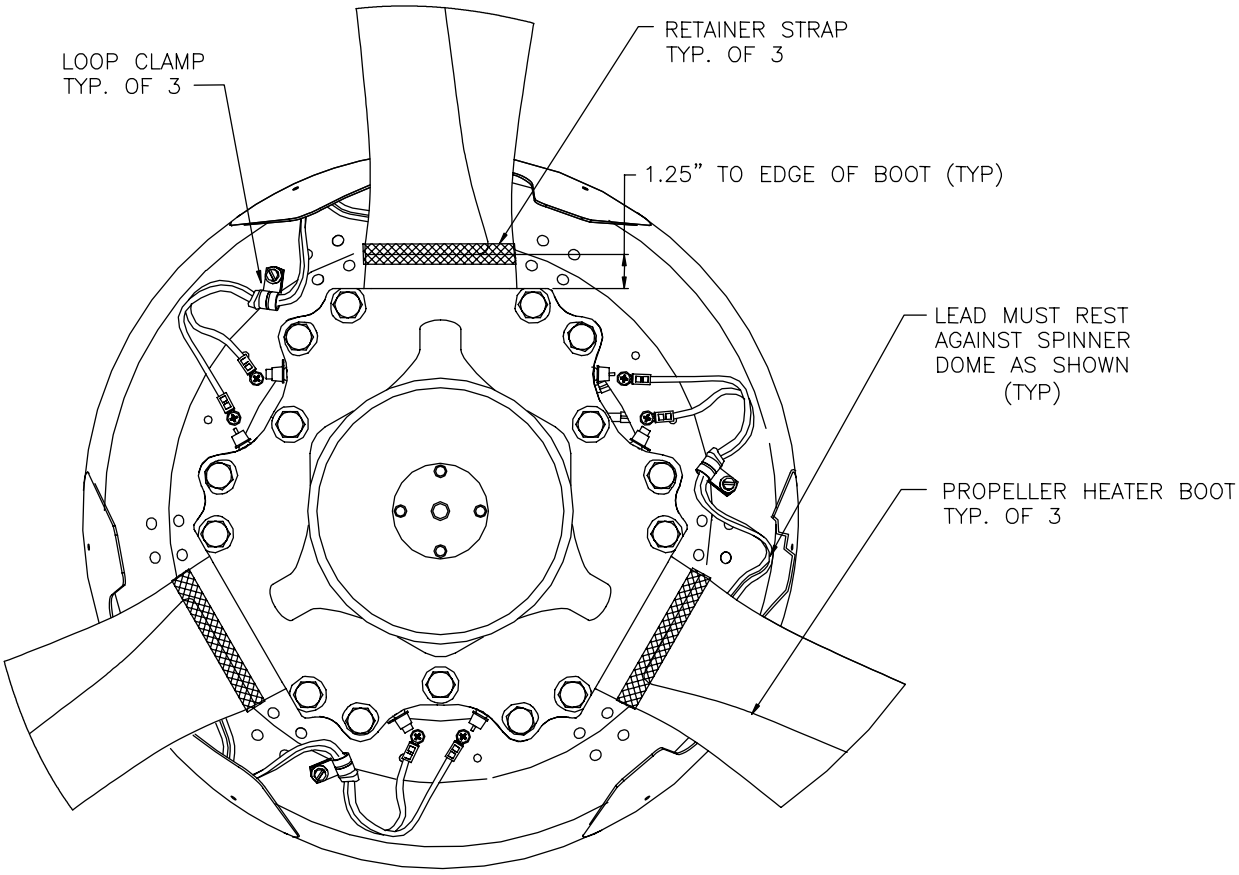

Figure 30 - 8 Hartzell Propeller Inc. Propeller Heater Boot

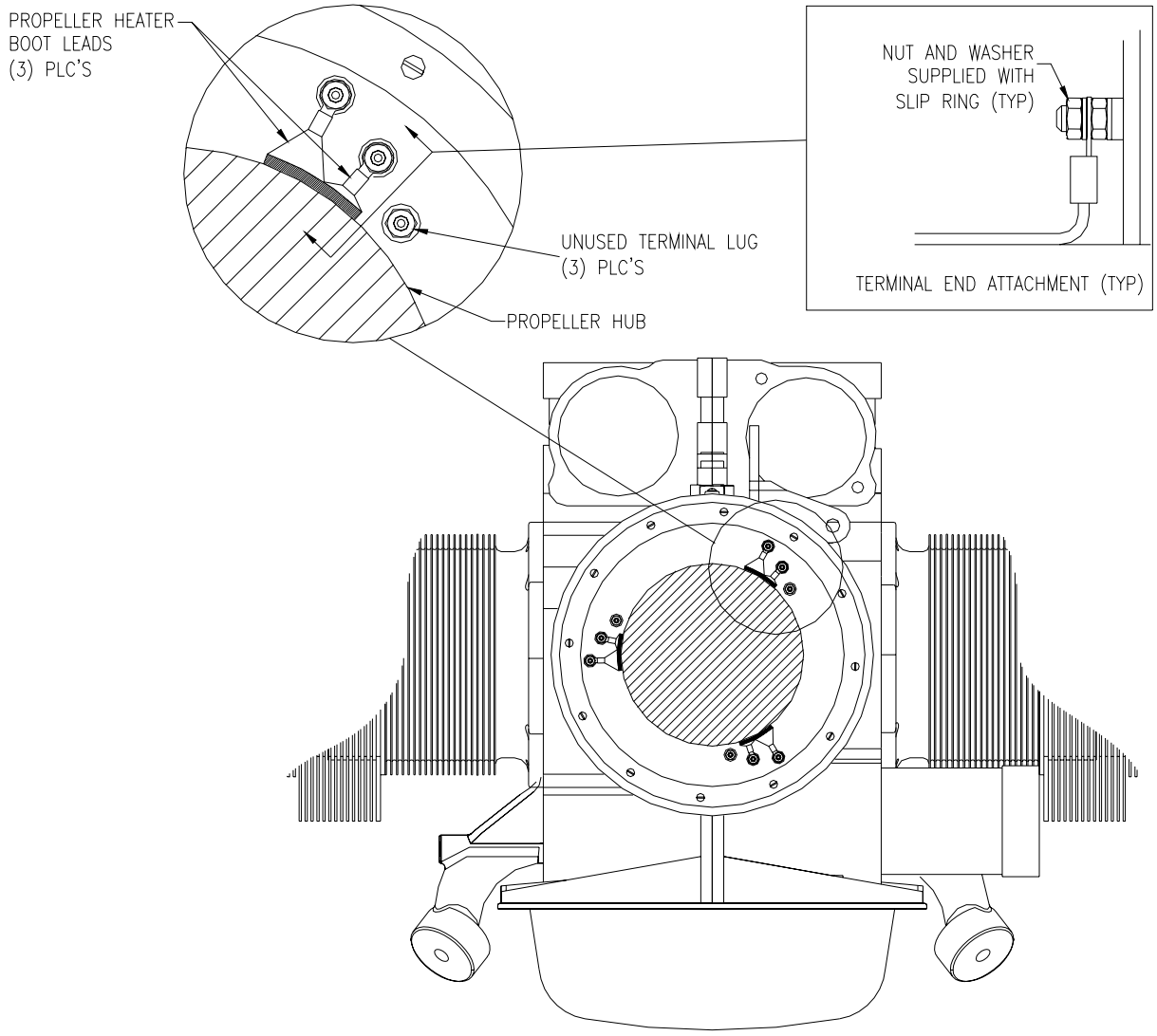

Figure 30 - 9 Heater Boot Lead Terminal Attachment

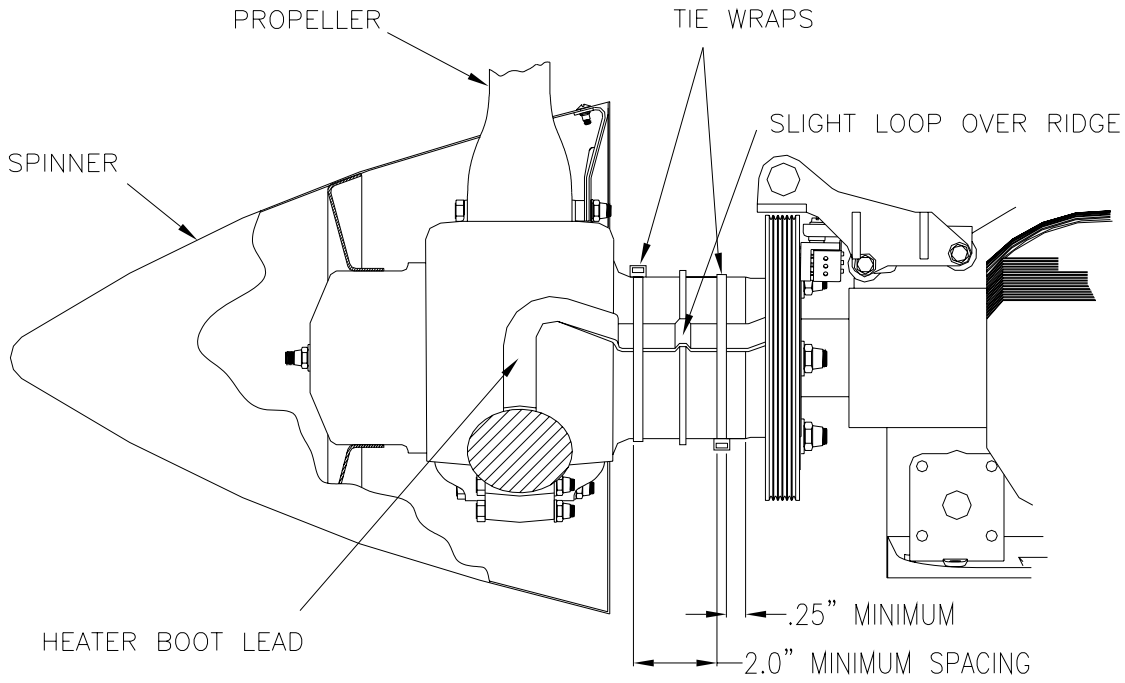

Figure 30 - 10 Heater Boot Lead Attachment

30-6. CONTROL MODULE – The control module is located in one of three places. If Avidyne avionics are installed the control module is attached to the standby battery bracket or to the right side instrument panel brace. If the Garmin G1000 system is installed the control module is located behind the instrument panel on the right hand side.

a. **Control Module (Basic or Avidyne Option) –** See Figure 30 - 11 or Figure 30 - 12.
   1. **Removal**
      a) Remove the co-pilot's side kidney panel per Chapter 25.
      b) Disconnect the cabling to the control module.
      c) Remove two screws, washers, and nuts attaching the control module to the mounting bracket.
   2. **Installation –** Installation is the direct reverse of removal.

b. **Control Module (Garmin G1000 Option) –** See Figure 30 - 13. For the Garmin G1000 option, the control module is located behind the instrument panel on the right hand side.
   1. **Removal**
      a) Remove the glare shield per Chapter 25.
      b) Disconnect the cabling to the control module.
      c) Remove two 8-32 pan head screws attaching the control module to the mounting bracket.
   2. **Installation –** Installation is the direct reverse of removal.

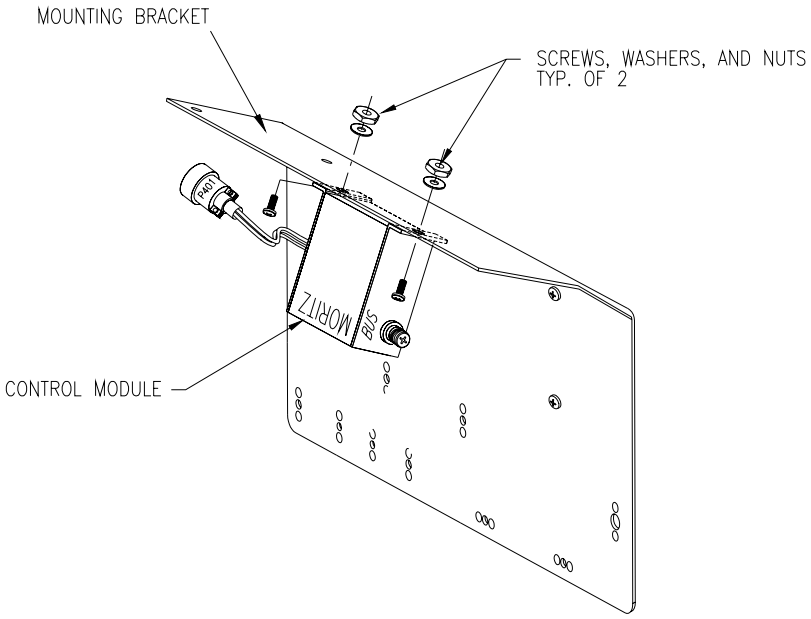

Figure 30 - 11 Control Module (Basic or Avidyne Option)

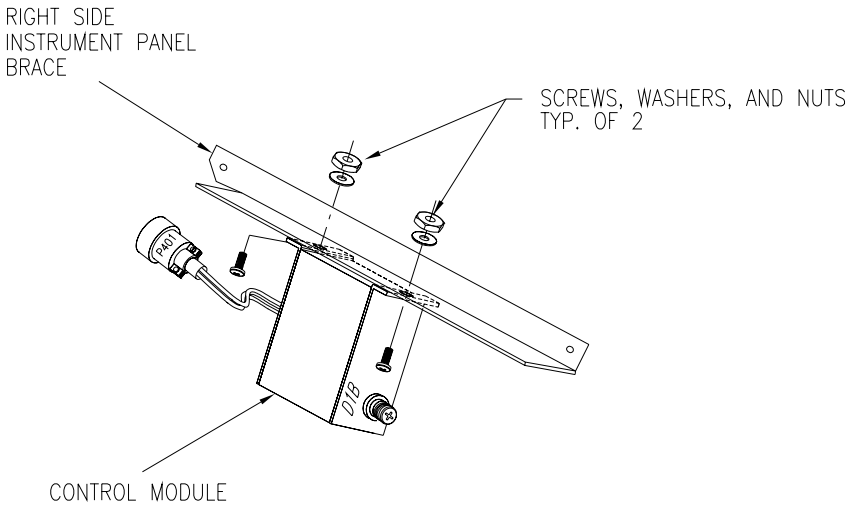

Figure 30 - 12 Control Module (Basic or Avidyne Option)

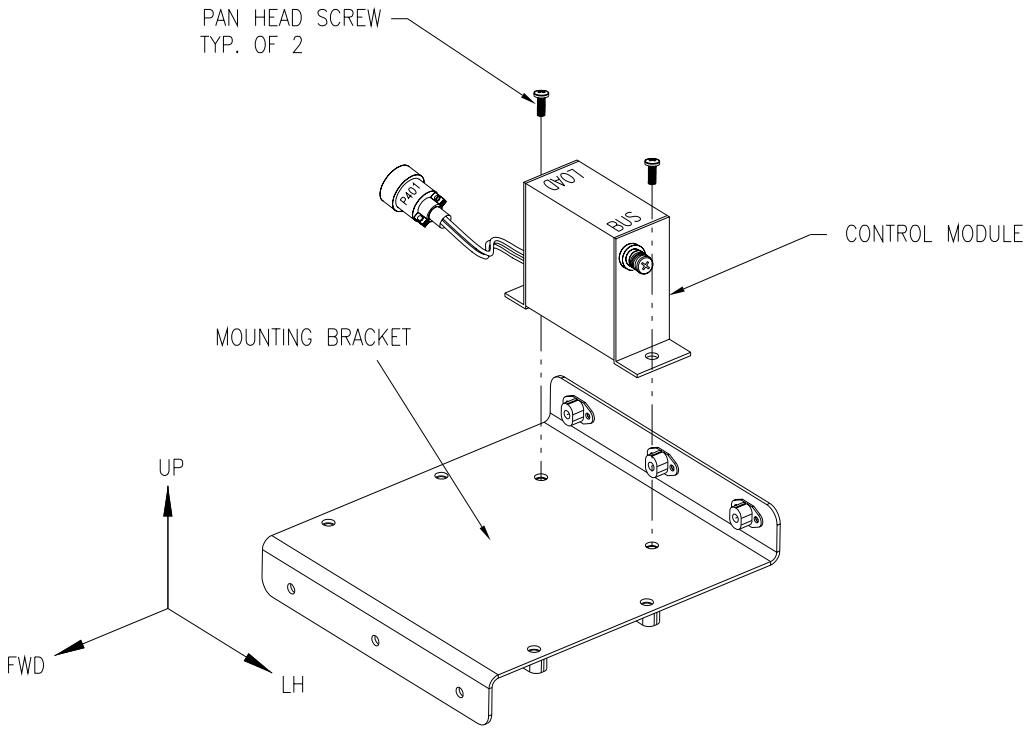

Figure 30 - 13 Control Module (Garmin G1000 Option)

c. **Control Module Maintenance –** Inspect the control module during the 100 hr and/or annual inspections. There are no servicing requirements for the propeller de-ice system outside normal inspection intervals or during routine maintenance. The propeller de-ice system has a high current draw; use ground power or start the engine.
   1. With aircraft power on, actuate the propeller de-ice system by pressing the de-ice control switch. Verify the power on self test performs correctly.
      a) When the Propeller de-ice control switch is powered on, the 'ON' and 'FAIL' annunciators will illuminate for one second and extinguish for one second. The on-off cycle will repeat during the six second self test. After the self test is completed the 'PROP' and 'ON' annunciators will illuminate steady for 30 seconds, indicating the DEICE is in the ON cycle. The 'Prop' annunciator will remain illuminate whenever the Propeller de-ice control switch is powered on.
   2. Inspect attachment of the control module to the support bracket for loose hardware.
   3. Inspect the wiring for damage or abrasion.
d. **Control Module Repair** – The control module is non-repairable and should be replaced if found to be defective.

30-7. TESTING

Before testing, assure that all electrical connections are secure and the propeller has been reinstalled.

When the Prop De-Ice Control Switch is powered on, the Prop De-Ice Control Module will perform a self-test. The 'ON' and 'FAIL' annunciators will illuminate for one second and

extinguish for one second during a six second cycle. The 'PROP DE-ICE' annunciator will remain illuminated whenever the prop de-Ice control switch is powered on.

Cycle testing should be performed with the cowling and spinner installed and the engine running.

CAUTION

Do not run the engine with the spinner removed. Damage to the propeller boot leads will occur.

a. **Ground Test –** The ground test is best accomplished with two persons. One person actuates the system from the cockpit and the second person verifies heater output at the propeller.

1. **Basic Operation Test** – The Basic Operation Test verifies the general electrical connections.
   a) Connect a source of external power to the aircraft.
   b) Power the right bus
   c) Actuate the prop de-ice switch, and wait for the controller set test to complete.
   d) Assure the 'ON' annunciator is illuminated.
   e) Verify the three propeller de-icers are generating heat by carefully touching the boots.
   f) After the system is verified to be heating, deactivate the prop de-ice switch.

CAUTION

Do not operate the Propeller De-Ice system for prolonged periods. Overheating and damage will occur.

2. **Cycle Time Test**
   Perform the Ground Cycle Test with the engine running to allow for proper cooling of the propeller boots.

   When the pilot presses the propeller de-ice switch, the prop de-ice system will perform a self-test for six seconds. After the self-test is completed the 'PROP' and 'ON' annunciators will illuminate steady for 30 seconds, indicating the propeller de-ice system is in the ON cycle.

   After 30 seconds the propeller de-ice 'ON' annunciator well flash, indicating the propeller de-ice is in the OFF cycle.

   Propeller de-ice 'ON' will continue to cycle 30 seconds steady state, and 30 seconds flashing until the pilot de-selects the propeller. de-ice switch
   a) Actuate the propeller de-ice switch and wait for the controller self test to complete.
   b) The 'ON' annunciator should remain steady for 30 seconds. Time the ON portion of the cycle with a stopwatch.

c) When the 'ON' annunciator begins to flash, the system is in the OFF portion of the cycle. Time this portion of the cycle with a stopwatch. The time should be 30 seconds. Verify the three propeller boots are cooling off by carefully touching the boots.
d) Repeat the cycle time test several times to verify consistent cycle times.

CAUTION

The propeller boot heaters will become extremely hot during operation. Exercise care when touching the propeller boots or adjacent areas.

3. **Fault Indications –** The Troubleshooting paragraph of this chapter charts the fault indications along with probable causes and remedies.
   a) The 'FAIL' light and the 'ON' light illuminate at the same time during the test – This situation indicates the Propeller De-Ice system is drawing less than 18 amps. Check for loose or open circuits, or heater or brush failure.
   b) The 'FAIL' light illuminates and the 'ON' light extinguishes during the test – This situation indicates the current draw is too high, above 30 amps. If this condition occurs, check for a short to ground in or around the brushes or power leads. Another cause could be a faulty propeller de-ice system control module, or no or low power at the bus terminal of the control module.

NOTE

The propeller heaters will be hotter at the root than at the outboard section; this is normal due to varying watt density in the heating element and is not cause for concern.

b. **In-Flight Testing**

It is recommended that the propeller de-ice system be operated during flight periodically to test the system. It is suggested that the system be turned off when not needed or when clearing the runway after landing.

The propeller de-ice system may be activated prior to take off. Assure that the current draw is between 19 and 25 amps.

a) Actuate the propeller de-ice switch and wait for the controller self test to complete.
b) The 'ON' annunciator should remain steady for 90 seconds. Time the ON portion of the cycle with a stopwatch.
c) When the 'ON' annunciator begins to flash, the system is in the OFF portion of the cycle. Time this portion of the cycle with a stopwatch. The time should be 90 seconds. Verify the three propeller boots are cooling off by carefully touching the boots.
d) Repeat the cycle time test several times to verify consistent cycle times.

30-8. PROPELLER HEAT/DE-ICE – TROUBLESHOOTING

Troubleshooting Chart				
Annunciator Lights	**Trouble Indication**	**Probable Cause**	**Inspection**	**Remedy**
ON – Steady On FAIL – Steady On	Low Current Draw (Under 18 A)	Heater boot open.	Electrically isolate each boot and check for continuity.	Replace boot if internal open.
				Tighten boot electrical connection at slip ring terminal.
		Brush failure.	Inspect brushes for condition and proper contact with slip ring.	Replace brushes.
				Align brushes.
		Open in power wiring.	Inspect wiring to verify condition. Inspect all connections for tightness.	Repair wiring as necessary. Tighten all connections.
ON – Off FAIL – Steady On	High Current Draw (Over 30 A)	Short to ground in power wiring.	Inspect wiring for shorts. Inspect all connections for tightness.	Repair wiring as necessary. Tighten all connections.
		Short to ground at brush block.	Inspect brushes for condition and short to ground	Replace brushes.
				Align brushes.
		Short between slip ring commutators.	Inspect slip ring assembly to verify electrical isolation	Replace slip ring assembly
	Low Voltage at Control Module	Loose connection at fuse or Control Module	Inspect wiring to verify condition. Inspect all connections for tightness.	Repair wiring as necessary. Tighten all connections.
		Faulty control module.	Test control module.	Replace control module.

Troubleshooting Chart				
Annunciator Lights	**Trouble Indication**	**Probable Cause**	**Inspection**	**Remedy**
	Radio Noise or interference when the De-Ice system is ON	Brushes arcing.	Inspect brushes for evidence of burning.	Replace brushes.
				Align brushes.
		Loose connection.	Inspect wiring to verify condition. Inspect all connections for tightness.	Repair wiring as necessary. Tighten all connections.
		Wire too close to radio equipment	Inspect wiring to verify location.	Reroute wiring to at least 8 in. from radio equipment and antenna leads.
		Defective fuse.	Place jumper across fuse.	If radio noise disappears, replace the fuse.
		Defective control switch.	Place jumper across pins 1 and 8 and pins 4 and 10 on the control switch.	If radio noise disappears, replace the control switch.
	Rapid or uneven brush wear or brush chipping	Brush assembly out of alignment.	Inspect brushes for condition and proper contact with slip ring.	Align brushes.
		Slip Ring run-out.	Verify slip ring run-out is less than 0.005 in.	Replace slip ring.
		Slip Ring surface defects.	Visually inspect surface condition.	Resurface slip ring if within limits.
				Replace slip ring.
		Oil leak at propeller hub or engine.	Visually inspect slip ring for oil.	Correct oil leak condition.
	System does not actuate when control switch is selected	Faulty control switch.	Bypass control switch and verify system operation.	Repair wiring as necessary. Tighten all connections.
		Faulty or blown fuse.	Replace fuse. If fuse blows again inspect wiring for shorts.	Replace fuse.

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CHAPTER 31

INDICATING/RECORDING SYSTEMS

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Chapter 31

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31-1. VOLTMETER/OAT/CLOCK – If the Avidyne Entegra Primary Flight Display (PFD) or the Garmin G1000 system is installed in the aircraft, the Voltmeter/OAT/Clock described by the following paragraphs will not be installed in the aircraft.

a. **Voltmeter/OAT/Clock** – This instrument contains three separate indications. The upper window is dedicated to voltmeter and outside air temperature readings, and the lower window is a multi-function timepiece. See Figure 31 - 1 for a drawing of the instrument. There are three control buttons on the face of the indicator. Pressing the top button, labeled O.A.T. and VOLTS, will cycle the reading in the upper Liquid Crystal Display (LCD) from a voltage reading, to an outside air temperature in °F, to an outside air temperature reading in °C. The lower two buttons, labeled SELECT and CONTROL, are for the multi-function clock and are discussed in Section 7 of the POH/AFM.

b. **The digital clock** displays four time modes: universal time (UT), sometimes referred to as Greenwich Mean Time (GMT), local time (LT), flight time (FT), and elapsed time (ET). The time mode selected is indicated in the lower digital display window on the left side with a flashing underscore below the selected time mode. In Figure 31 - 1 the ET time mode is selected. Pressing the SELECT button several times cycles the display through the four time modes in the order they are listed.

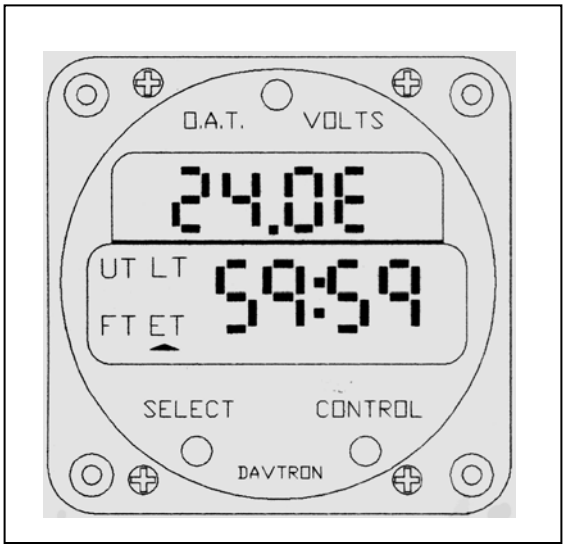

Figure 31 - 1 Voltmeter/OAT/Clock

c. **To test the clock**, hold the SELECT button in for three seconds. This will cause the display to read 88:88 with all four modes underscored. When electrical power is first turned on, the top display will indicate the system's voltage, and the clock will display the mode it was in when the electrical power was turned off. When the electrical system of the airplane is not on, the time and other settings such as flight time are held in the memory of the clock by a backup AA battery. The battery has a three-year life, but it must be replaced every two years with a good 1.5 volt battery. Incorrect time and mode readings during power up are indications of an expended or defective backup battery. If temperatures are below –20ºF, the numbers in the clock will not appear.

d. **Initial Set-up**

To Set UT (Universal Time[1]) use the following instructions:

1. Press the SEL button one or more times until UT is displayed on the left hand side of the lower window.
2. Press both the SEL and CTL buttons at the same time to enter the set mode. The tens of hours digit will start flashing.
3. Press the CTL button one or more times to change the tens of hours digit to the correct digit of Universal Time (see footnote 1).
4. Press the SEL button once to move to the single hour digit, which will start flashing.
5. Press the CTL button one or more times to change the single hour digit to the correct digit of Universal Time.
6. Press the CTL button one or more times to change the tens of minutes digit to the correct digit of Universal Time.
7. Press the SEL button once to move to the single minute digit, which will start flashing.
8. Press the CTL button one or more times to change the single minute digit to the correct digit of Universal Time.
9. Press the SEL button to exit the set mode. The lighted annunciator will resume its normal flashing indicating the UT clock is running.

To Set LT (Local Time[2]) use the following instructions:

1. Press the SEL button one or more times until LT is displayed on the left hand side of the lower window.
2. Press both the SEL and CTL buttons at the same time to enter the set mode. The tens of hours digit will start flashing.
3. Press the CTL button one or more times to change the tens of hours digit to the correct digit of Local Time (see footnote 2).
4. Press the SEL button once to move to the single hour digit, which will start flashing.
5. Press the CTL button one or more times to change the single hour digit to the correct digit of Local Time.
6. Press the SEL button to exit the set mode as the minutes are already synchronized with the UT clock and can not be set in Local Time.

31-2. HOUR METER – GENERAL

a. **Indicating** – Flight time is recorded by an hour meter – most frequently referred to as the *Hobbs.* If basic instruments only or Avidyne avionics are installed, the Hobbs is installed in the right knee bolster, next to the ELT remote switch. If the Garmin G1000 system is installed, the Hobbs is installed on the left side of the center console. The instrument is

[1] The times of various events, particularly astronomical and weather phenomena, are often given in "Universal Time" (abbreviated **UT**) which is sometimes referred to, now colloquially, as "Greenwich Mean Time" (abbreviated **GMT**). The two terms are often used loosely to refer to time kept on the Greenwich meridian (longitude zero), *five hours ahead of Eastern Standard Time*. Times given in UT are almost always given in terms of a 24-hour clock.

[2] Local Time is *eight hours later* (PST) or *seven hours later* (PDT) than the Universal Time just set.

DC powered. DC power for the hour meter bypasses the master switch and will operate with the master switch in the off position, provided sufficient battery power

is available. The system is protected by a fuse located in a fuse block next to the battery. The meter will record up to 9999.9 hours. Flight time is measured by either an oil pressure switch or an air switch, depending on which option is installed.

b. **Oil Pressure Switch** – There is an oil pressure switch next to the oil pressure transducer as shown in Figure 31 - 2. The switch is closed when oil pressure is introduced to the system. If the battery is adequately charged, a ground is established for the Hobbs (the other side of the Hobbs is powered from the 3 amp fuse on the power grid), and the time the engine is in operation is recorded and displayed on the meter.

c. **Air Switch –** There is an air switch located either under the standby battery bracket or next to the terminal junction rail behind the co-pilot's side of the instrument panel if Avidyne avionics are installed, or behind the instrument panel on the co-pilot's side if the Garmin G1000 system is installed, that activates at 2 in. of Hg or approximately 60 kts. This switch is plumbed into the pitot line for its activation.

NOTE

To disable the Hobbs during a pitot static check, remove the Hobbs fuse from the power grid. This will stop the accumulation of "false" time on the Hobbs. Replace the fuse after the test.

31-3. DOCUMENTATION OF HOBBS METER REPLACEMENT

a. **Placard** – If a Hobbs meter becomes inoperative, it must be replaced as soon as possible. The time shown on the defective meter plus the best estimate of the additional time in service must be placarded in an area near the Hobbs. For example, if the meter had 627.9 hours when it became inoperative and an additional 2.1 hours of time in service was flown before replacement, then the placarded time would be 630 hours. A statement like *630 hours must is the current Hobbs Meter reading* is an appropriate placard.

b. **Logbooks** – Similar statements like those in paragraph a, along with the date and time of the discrepancy must be adequately documented in the airframe and engine logbooks.

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31-4. REMOVAL, INSTALLATION, AND MAINTENANCE OF THE CLOCK

a. **Removal of the Clock**
   1. Remove the glare shield per instructions in Chapter 25, and remove the connector installed in the back of the clock.
   2. Unscrew the four panel screws surrounding the clock.
   3. Remove the clock from the backside of the flight instrument panel.

b. **Installation of the Clock**
   1. Install the connector into the back of the clock.
   2. Align the clock with the screw holes in the flight instrument panel, and install the four screws. The screws should be snug tight.
   3. Reinstall the glare shield.

c. **Maintenance of the Clock** – No special maintenance of the clock is necessary other than the biennial replacement of the backup battery.

NOTE

It is not necessary to remove the clock to replace the backup battery. The clock is in the top portion of the flight instrument panel, and when the glare shield is removed, the AA battery can be accessed.

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31-5. REMOVAL, INSTALLATION, AND MAINTENANCE OF THE HOUR METER

a. **Removal of the Hour Meter**
   1. If basic instruments only or Avidyne avionics are installed in the aircraft, remove the right knee bolster per instructions in Chapter 25. If the Garmin G1000 system is installed, remove the access panel on the right side of the center console.
   2. Remove the connector installed in the back of the hour meter. Be sure to label the location of each wire.
   3. Unscrew the two screws on the back of the knee bolster, and slide the unit out.

b. **Installation of the Hour Meter**
   1. With the hour meter partially inserted through the hole in the knee bolster or center console, as applicable, install the connectors (per labeling) into the back of the hour meter.
   2. Slide the unit into its slot flush with the bolster or center console, as applicable, and install the two screws. The screws should be snug tight.
   3. Reinstall the knee bolster per instructions in Chapter 25 or replace the center console access panel, as applicable.

c. **Hour Meter Maintenance** – No special maintenance of the hour meter is required.

31-6. REMOVAL AND INSTALLATION OF THE OIL PRESSURE SWITCH

a. **Switch Removal** – Use the following steps for removal of the oil pressure switch. See Figure 31 - 2.
   1. Remove the top engine cowling per instructions in Chapter 71.
   2. Remove the Packard electrical plug from the back of the switch.
   3. Use an open-end wrench to screw the $^{1}/_{8}$ in. pipe-threaded switch out of the engine case.

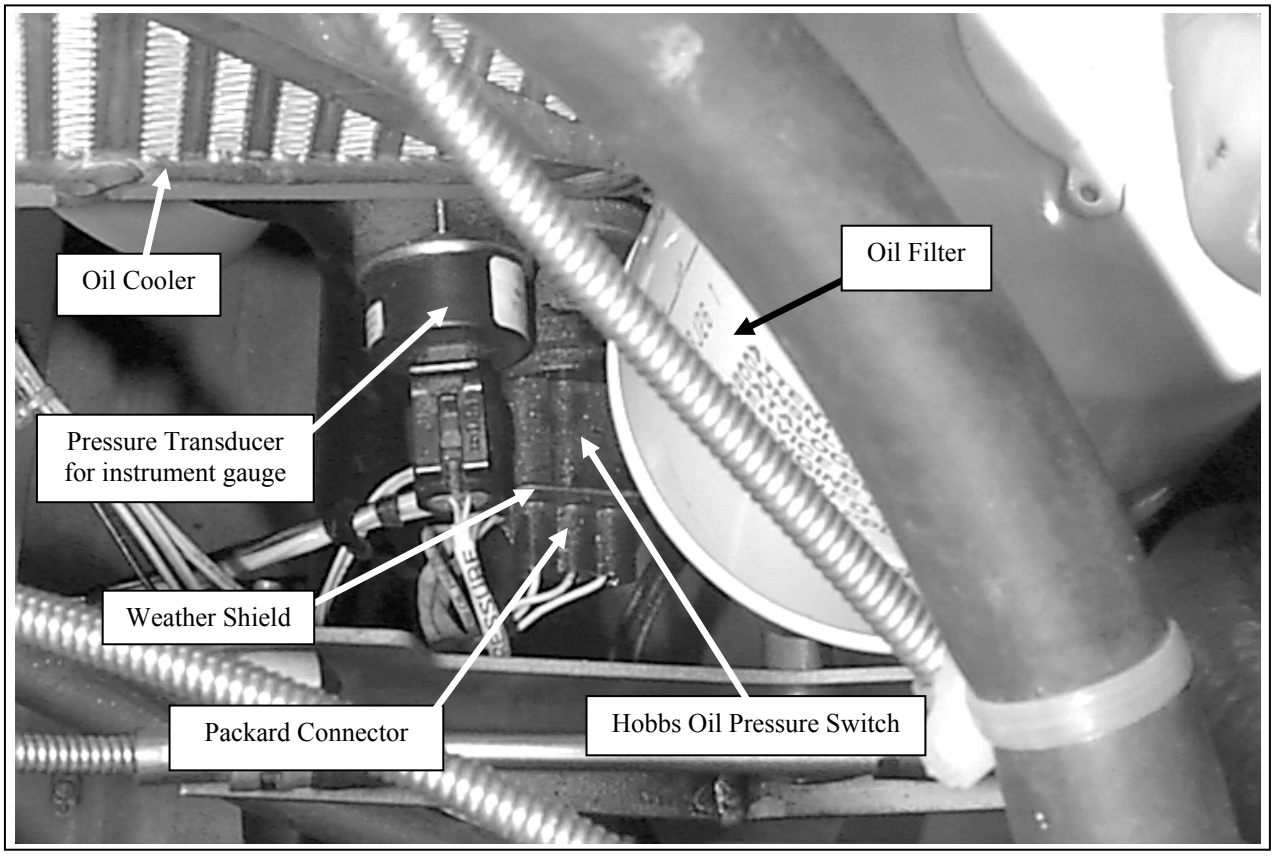

Figure 31 - 2 Hobbs Oil Pressure Switch

b. **Switch Installation** – Use the following steps for installation of the oil pressure switch.
   1. Install the $^{1}/_{8}$ in. pipe-thread pressure switch into the engine case. Torque switch 60 to 80 in.-lbs.
   2. Connect the Packard connector to the oil pressure switch. Ensure there is a red weather shield around the Packard switch.
   3. At an oil temperature reading of 75ºF or higher, run the engine at about 1700 RPM to pressure test the system.
   4. Shut down the engine and check for oil leaks.
   5. Reinstall the engine cowl.

31-7. REMOVAL AND INSTALLATION OF THE AIR SWITCH

a. **Removal of the Air Switch**
   1. If basic instruments only or Avidyne avionics are installed in the aircraft, remove the glare shield and right side kidney panel per Chapter 25. If the Garmin G1000 system is installed, remove the glare shield per Chapter 25.
   2. Remove the mounting hardware, wires, and pitot line.
   3. Cap the pitot line, and connect the wires together if the aircraft is going to be operated with the switch removed.

b. **Installation of the Air Switch**

1. Connect the pitot line, wires (to normally open and common), and mounting hardware.
2. Test the pitot system for leaks. The system cannot exceed 500 fpm at 1000 ft. above field elevation.
3. Replace the glare shield and kidney panel, as applicable, per instructions in Chapter 25.

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CHAPTER 32

LANDING GEAR AND BRAKES

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Chapter 32

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32-1. GENERAL – SCOPE AND DEFINITION

The Cessna 350 (LC42-550FG) has a fixed tricycle landing gear configuration. The main gear is constructed of high strength tubular steel and mounted to the fuselage via a welded steel gearbox. The main gearbox is bolted directly to the fuselage structure. The airplane has brakes located on the main wheels. The brakes are used for steering the airplane via differential braking. The nose gear is an oleo strut which casters for steering purposes. The oleo strut has a self-centering device to lock the nose wheel in a fixed position during flight. Aerodynamic fairings are used on the airplane to cover the landing gear and help reduce drag during flight.

This chapter covers the removal and installation of the main gear, nose gear, wheel pants, and fairings. Standard maintenance procedures are also covered.

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32-2. MAIN LANDING GEAR – GENERAL

The main landing gear consists of the gearbox (located inside the fuselage), the gear legs, the axle mounting foot, axle, wheel, tire and tube, upper fuselage fairing, gear leg fairing, wheel pant, and wheel pant access panel.

32-3. MAIN GEAR WHEEL PANTS AND FAIRINGS

a. **Removal and Installation –** The removal and installation instructions below are for the left main gear. Removal and installation practices for the right side are typical unless otherwise noted.

1. Remove the wheel pant access panel from the inside face of the wheel pant by removing the attachment screws.
2. Remove the screws attaching the leg fairing to the wheel pant.
3. Remove the two bolts attaching the wheel pant to the support bracket.
4. Remove the bolt and washer attaching the wheel pant to the standoff shown in Figure 32 - 1 and Figure 32 - 2.
5. Slide the wheel pant off by lifting and rotating it to clear the wheel and tire assembly.
6. Remove the brake caliper per the instructions in section 32-14a.
7. Jack the left side of the airplane such that adequate clearance is achieved for removing the main gear wheels and leg fairings.
8. Place a small pan or bucket under the brake caliper, and disconnect the brake line from the brake caliper. The pan will help catch any excess brake fluid that may leak from the brake line once it is removed. Once the brake line is removed, cap the end to prevent any excess brake fluid from draining out of the system.

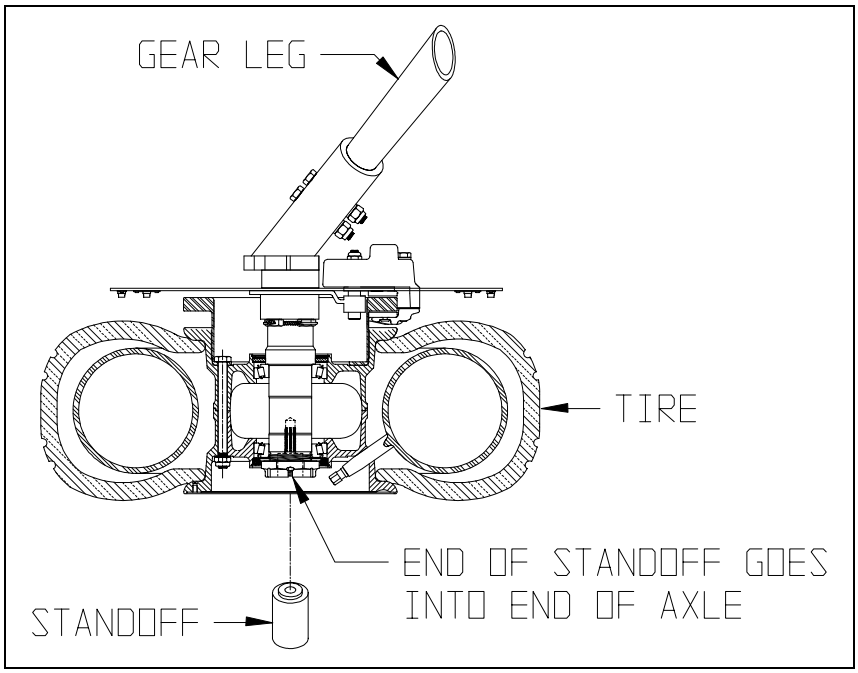


Figure 32 - 1 Wheel Pant Standoff Location

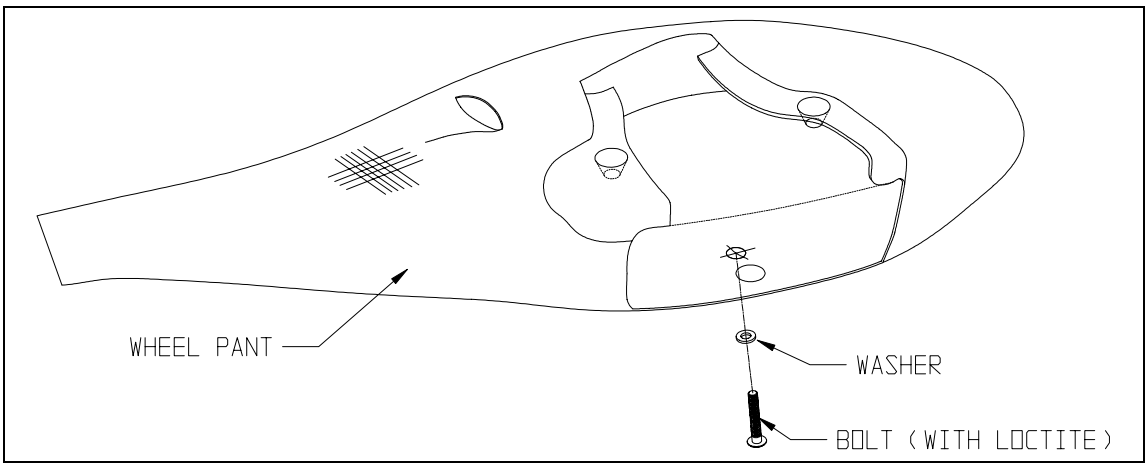


Figure 32 - 2 Hardware Attaching Wheel Pant to Standoff

9. Remove the main gear axle foot from the gear leg by removing the two AN5-22A bolts and two MS20365-524C all-metal lock nuts (see Figure 32 - 3) and sliding it straight down along the axis of the gear leg.
10. Slide the gear leg fairing straight down along the axis of the gear leg to remove it. Jack the airplane as necessary to remove the leg fairing from the leg.
11. Remove the attachment screws holding the upper leg fairing to the fuselage belly.
12. Slide the fairing straight down the gear leg to remove it.
13. Reinstall in the reverse order. Torque all fairing attachment screws 10 to 12 in.-lbs., and torque the MS20365 nuts 140 to 180 in.-lbs.
14. When installing the standoff and attaching hardware, seat the standoff in the end of the axle on the inside of the wheel pant as shown in Figure 32 - 1 and Figure 32 - 2. Up to two washers may be glued to the inside of the wheel pant where the pant attaches to the standoff to compensate for the variation in wheel pant size. The washer should not exceed a 1 in. diameter. Use Loctite 243 on the bolt and torque 23 ft.-lbs., using care not to cause compression damage to the pant. Torque two bolts attaching the pant to the support bracket 30 to 36 in.-lbs.

b. **Servicing and Maintenance –** The main gear wheel pants and fairings are a glass fiber composite construction and do not require any servicing or maintenance. The wheel pants and fairings should be inspected periodically for cracking, chips, etc. and repaired or replaced as necessary. See Chapter 57 for removal and installation of optional anti-chafe tape. Contact the manufacturer for replacement parts. The interior of the wheel pant should be cleaned of all foreign debris.

32-4. MAIN GEAR ASSEMBLY

a. **Removal and Installation –** Use the following procedures for removal and installation of the main gear assembly**.**

1. Remove the main gear wheel pant per the instructions in section 32-3a.
2. Raise the airplane on jacks such that adequate room is achieved for removing the main gear wheel assembly.
3. Remove the two AN5-22A bolts and two MS20365-524C lock nuts from the main gear axle foot, and slide the wheel assembly straight down along the axis of the gear leg (see Figure 32 - 3).

4. Disassemble the main wheel assembly by removing the cotter pin and axle nut from the outboard end of the axle (see Figure 32 - 4).

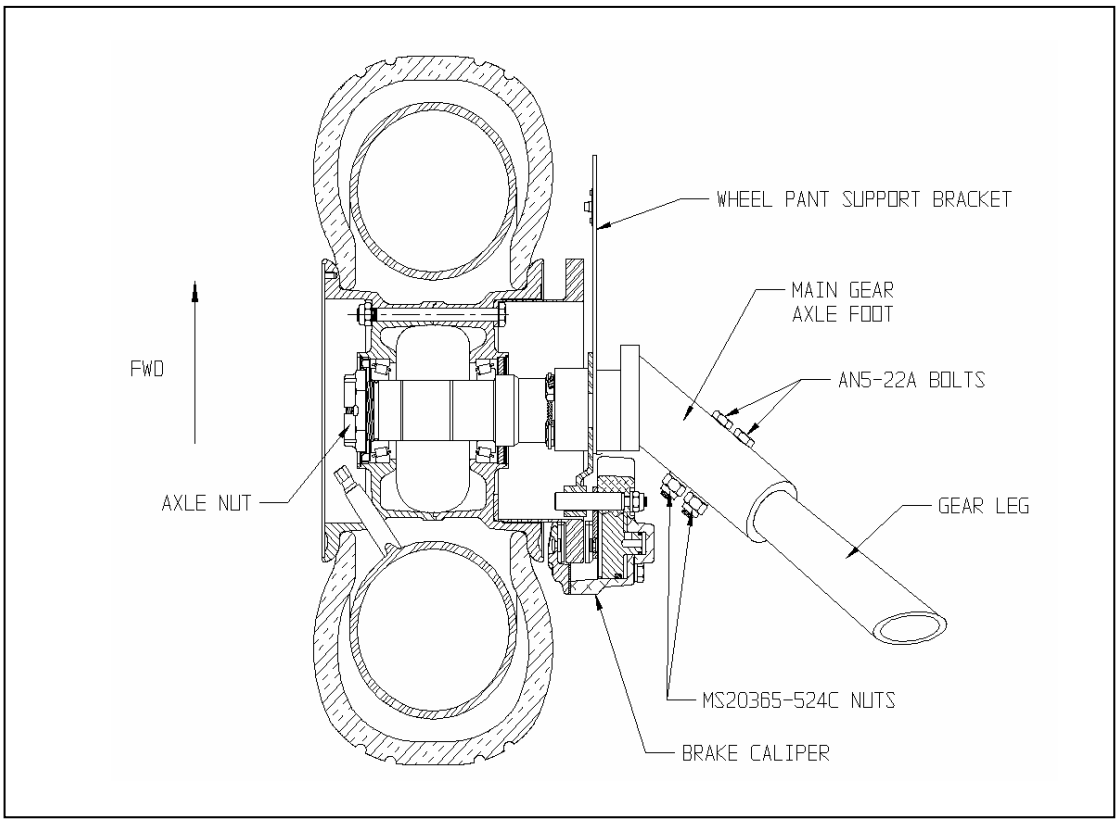


Figure 32 - 3 Main Gear Wheel Assembly

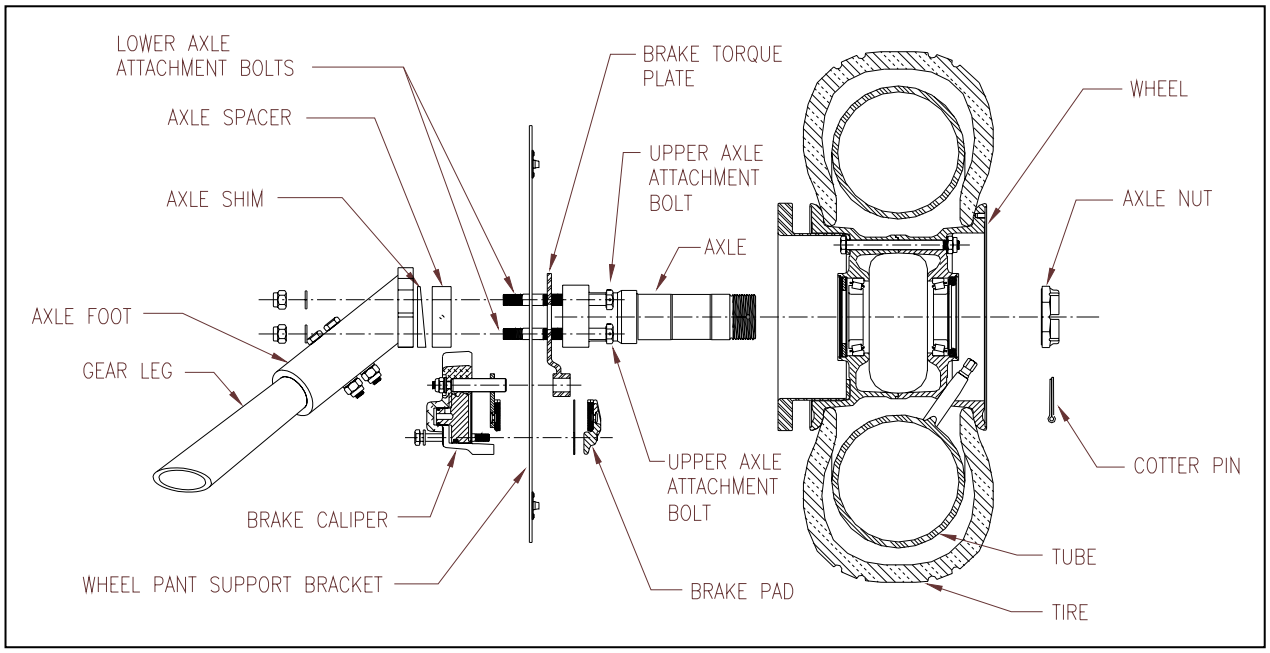


Figure 32 - 4 Main Gear Assembly, Exploded View

5. Remove the wheel from the axle by sliding it straight off.
6. Cut and remove the lockwire from the upper axle attachment bolts.
7. Remove the upper and lower axle attachment bolts to disassemble the remaining parts. Note order of assembly for reinstallation.
8. Remove the access panels from the aft cabin floor to gain access to the upper end of the gear legs.
9. Remove the gear leg attachment bolt located at the top end of the gearbox.
10. Remove the gear legs from the gearbox by pulling the leg straight out.

CAUTION

Take care when removing the gear legs so as not to cause nicks, scratches, or other damage to the surface of the leg.

11. Installation is in the reverse order. Re-safety wire the upper axle attachment bolts prior to installing the wheel.
12. Torque the gear leg attachment bolt 80 to 85 ft.-lbs. Torque the axle attachment bolts 240 to 280 in.-lbs., and torque the axle foot attach bolts 140 to 180 in.-lbs. Rotate the wheel while tightening the axle nut 150 to 200 in.-lbs. to seat the bearing, back off the axle nut to zero torque, tighten the axle nut to 30 to 40 in.-lbs. while rotating the wheel, rotate the axle nut to the nearest slot and cotter pin hole, and install the cotter pin.
13. Assure that the wheel rotates freely without perceptible play.

b. **Wheel Toe-in/Toe-out and Camber** (See Figure 32 - 5 and Figure 32 - 7) **–** During wheel installation it is important to determine the wheel toe-in/toe-out (θ) and camber (α). Wheel toe-in/toe-out and camber is established by inserting one or more shims between the axle block and spacer, if necessary, as shown in Figure 32 - 4. It is unlikely that the shims will need to be changed from what was initially manufactured, however, if the proper toe-in/toe-out and camber is not achieved, contact the manufacturer for the required shim(s).

1. **Toe-in/Toe-out –** Wheel toe-in/toe-out is the angle between the aircraft centerline and the wheel. Toe-out is set with all wheels on the ground, on a low-friction surface. Set toe-out, to the value θ indicated in the toe-out vs. weight chart shown in Figure 32 - 6. No more than 4 shims may be used with a maximum total thickness of 0.254 inches.
2. **Camber –** The camber is the angle from horizontal to the centerline of the axle. Camber is set with all wheels on the ground, on a low-friction surface. Set camber to the value α indicated in the camber vs. weight chart shown in Figure 32 - 8. No more than 4 shims may be used with a maximum total thickness of 0.254 inches.

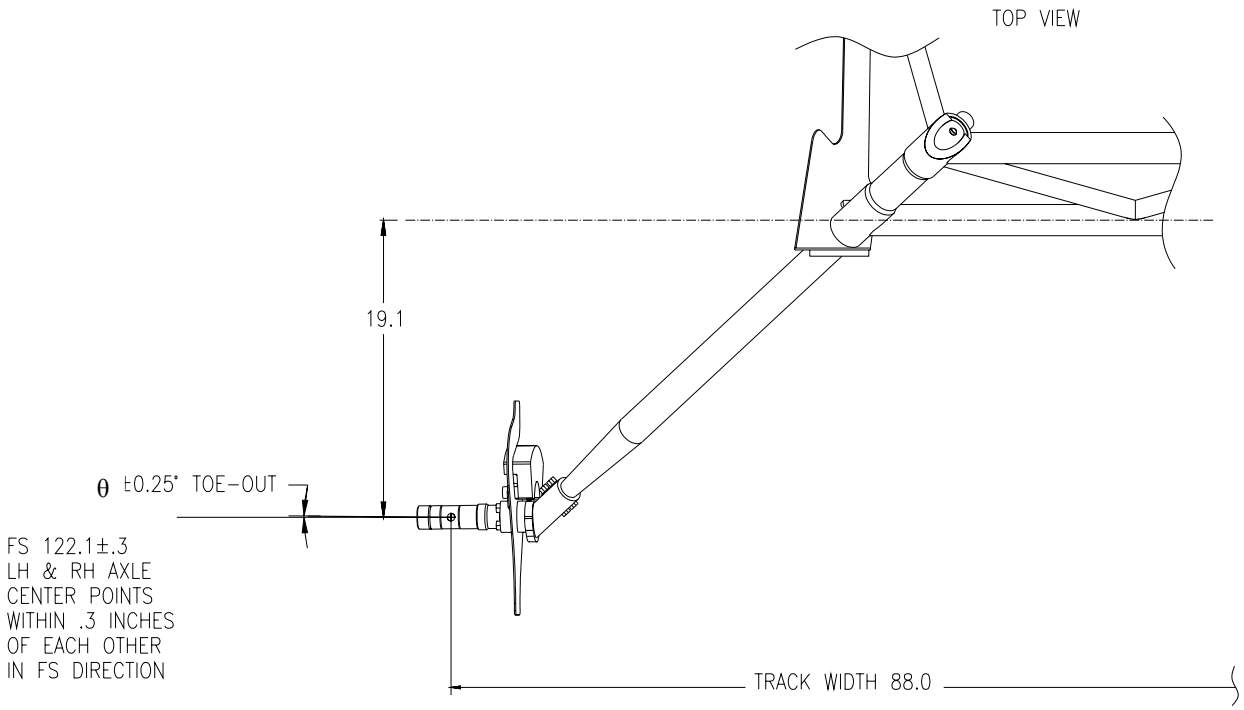


Figure 32 - 5 Toe in/Toe out

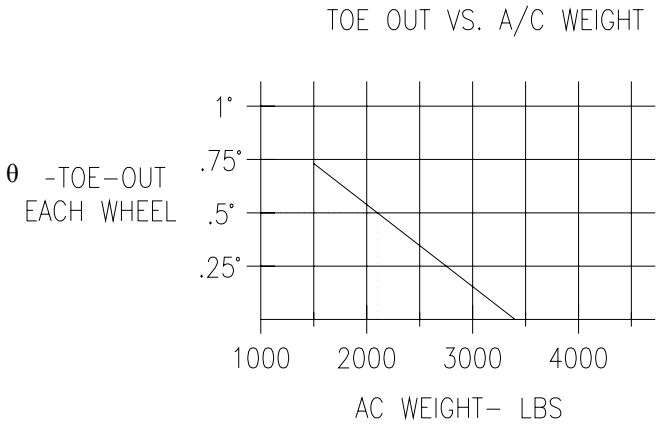


Figure 32 - 6 Toe-out vs. Aircraft (A/C) Weight

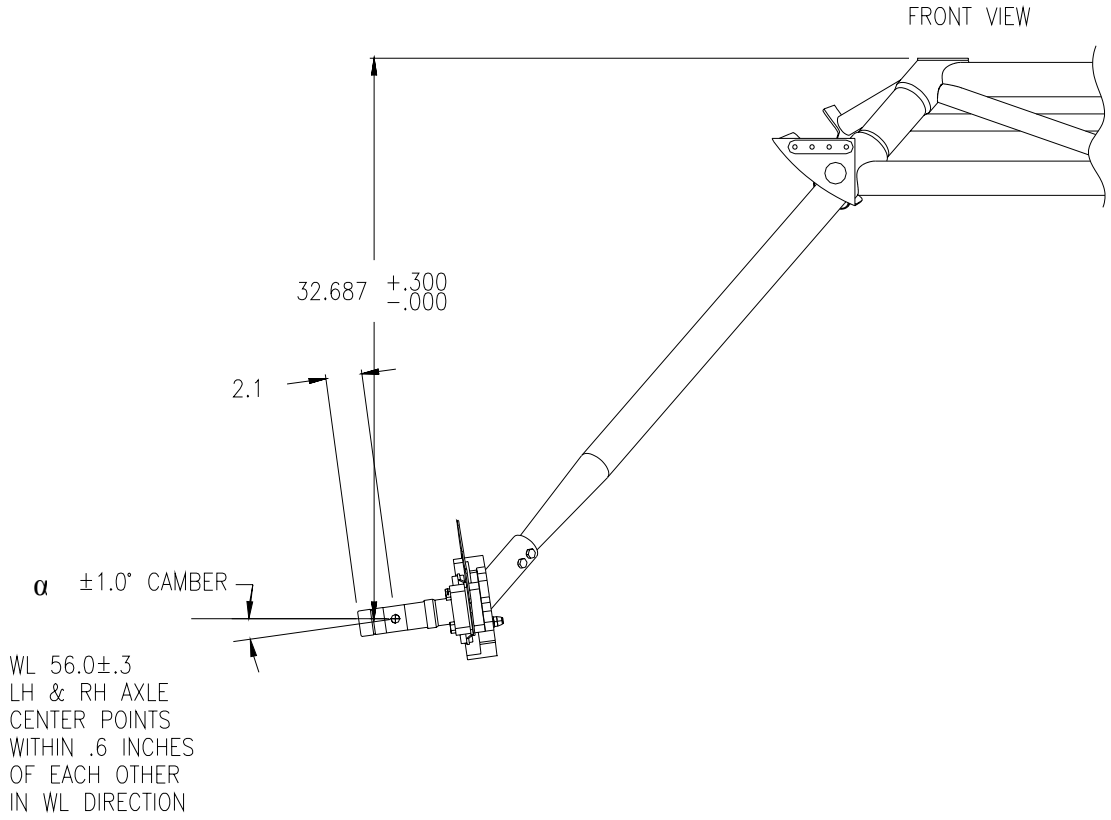


Figure 32 - 7 Camber

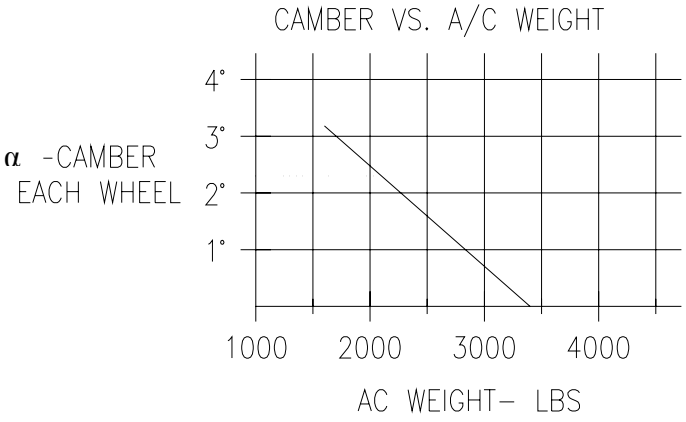


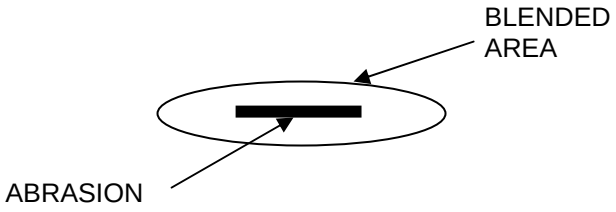
Figure 32 - 8 Camber vs. Aircraft (A/C) Weight

c. **Servicing and Maintenance –** The landing gear legs should be inspected every 100 hours or annually. Some of the items to inspect for are scratches, nicks, corrosion, and signs of overstress. Visually inspect the gear legs for straightness. The axles and other components should also be inspected for corrosion, cracks, signs of overstress, and wear. The main gear assembly should fit snugly together with very little, if any, free play.

32-5. MAIN LANDING GEAR LEG ABRASION REPAIR

The main landing gear legs may develop abrasions as a result of rubbing with the landing gear fairing upper rib. Typically, there are two abrasions opposite each other. The abrasions are located 16.0" from the upper tip of the gear leg; they are approximately 0.7" long and 0.020" deep.

a. Clean the surface area surrounding the abrasions.
b. Determine the depth and length of the abrasions. If the abrasion depth and length are greater than 0.020" and 1.0" respectively, contact Cessna for instructions.
c. Use Scotch-Brite™ or other equivalent material to blend out nicks, scratches, or cuts at the abrasion area. The repair area must be blended over a distance at least 10 times the depth of the abrasion.



d. After polishing or blending the abrasion, treat all bare metal with zinc chromate or zinc oxide.
e. Before installing the repaired main landing gear on the aircraft, contact Cessna for instructions on modifying the main landing gear fairing to prevent future rubbing.

32-6. GEARBOX

The main landing gear is attached to the fuselage structure via a welded steel gearbox. The gearbox is bolted directly to the fuselage structure.

a. **Removal and Installation** – The gearbox is not easily removable from the fuselage due to its location. Should the gearbox need to be removed or replaced, please contact the manufacturer for further instructions.
b. **Servicing and Maintenance** – The gearbox is a major structural component and should be inspected every 100 hours or annually for wear and corrosion. Remove the wheel pants and fairings per section 32-3a and remove the gear leg per section 32-4a. Inspect the gear leg sockets of the gearbox for wear and corrosion. If surface corrosion is present, lightly hone the interior diameter of the socket using a cylinder bore hone and light machining oil. A Scotch-Brite™ pad may also be used. After the corrosion has been removed, apply a thin film of machine oil or other surface protectant to help prevent further corrosion.

   The bushing at the lower end of the socket should also be inspected for excessive wear and replaced if required. When replacing the bushing, P/N GLY PGZ 3216 F, thoroughly clean the surface of the socket and install with Loctite 660 retaining compound. Allow the Loctite to fully cure prior to reinstalling the gear leg.

Every 3000 hours, the gearbox bolts must be checked for evidence of corrosion. Remove one attachment bolt from each side of the gearbox and check for corrosion. If corrosion is noted, replace all attachment bolts in the gearbox. If no corrosion is noted, replace bolts that were removed and check the torque on all bolts. Torque bolts 40 to 50 in.-lbs.

32-7. NOSE GEAR – GENERAL

a. The nose gear is a fixed hydraulic unit that casters freely during taxiing. Turning is accomplished via differential braking and optional rudder assist. Landing loads and shimmy dampening are controlled using hydraulic oil within the strut. Nitrogen gas pressure controls the ride height of the strut and supplies the necessary pressure to lock in the centering cam mechanism.

On the ground with the weight of the aircraft bearing down on the nose strut, an internal keyway within the strut slides out from a locking cam making rotational movement possible. The moment the aircraft takes flight and all weight is off the nose gear, gas pressure moves the keyway into the locking cam eliminating any ability to caster during flight.

During landing the weight of the aircraft overpowers the gas pressure and slides the keyway out from the locking cam, allowing full rotational movement of the nose wheel.

32-8. NOSE GEAR WHEEL PANT AND FAIRING

a. **Removal and Installation** – Use the following procedures for removal and installation of the nose gear wheel pant and fairing.

1. Remove the upper and lower engine cowls per instructions in Chapter 71.
2. Remove the aft screw attaching the nose strut fairing to the fuselage.
3. Raise the front of the airplane such that the nose wheel is completely clear of the ground.
4. Remove the four AN5-12A bolts and washers attaching the nose wheel assembly to the base of the nose strut (see Figure 32 - 9).

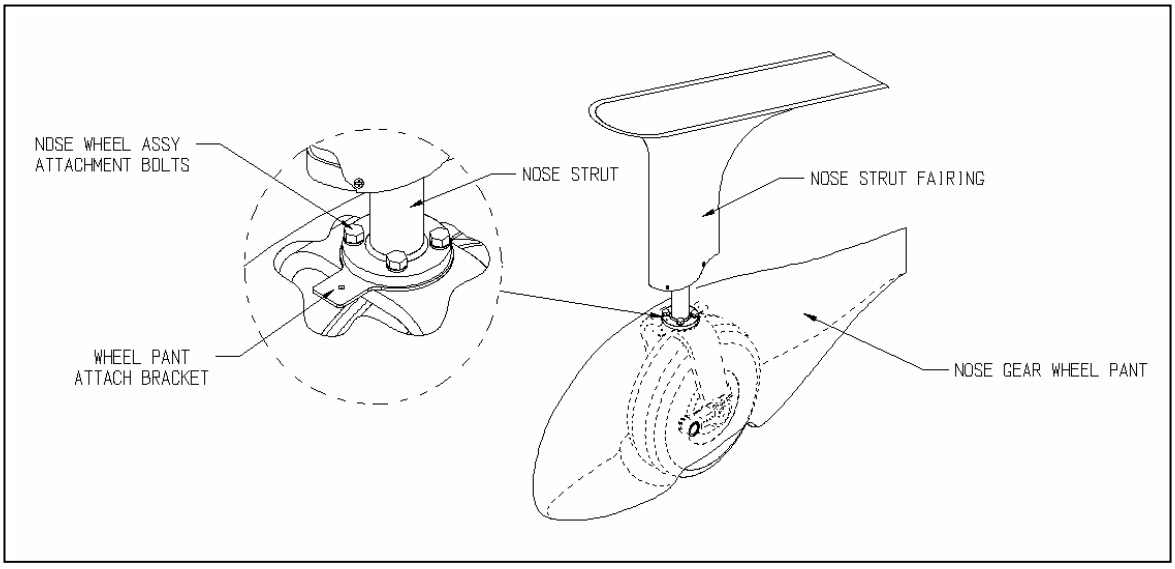


Figure 32 - 9 Nose Wheel Pant and Leg Fairing

5. For S/N 42001 to 42089 remove the four MS21207-6-8 self-tapping screws from the base of the strut fairing, and slide the internal rib straight down and off of the nose strut.
6. for S/N 42090 and on remove the two AN526C1032 or MS51958 truss head screws and NAS1149FN832P washers from the rear of the base of the strut fairing, and remove the heel plate.
7. Slide the nose strut fairing straight down and off the nose strut.
8. Remove the two AN526C1032 or MS51958 truss head screws from the top of the wheel pant attaching the wheel pant to the attach bracket (see Figure 32 - 10).
9. Remove the two AN526C1032 or MS51958 truss head screws from either side of the wheel pant (see Figure 32 - 10).

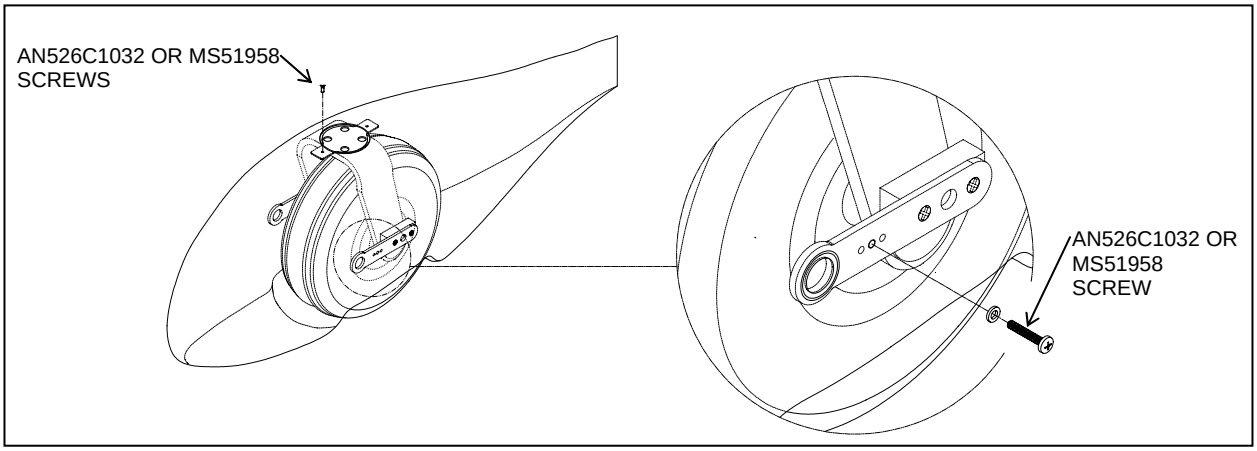


Figure 32 - 10 Wheel Pant Attachment Hardware

10. Remove the nose gear wheel pant from the wheel and fork assembly.
11. Installation is in the reverse order. Torque all AN526C1032 or MS51958 screws 30 to 36 in.-lbs. Hand tighten the MS21207 screws holding the internal rib, or the AN526C1032 or MS51958 screws holding the heel plate, to the nose strut fairing snug. Torque the AN5-12A bolts 140 to 180 in.-lbs.

b. **Servicing and Maintenance –** The nose gear wheel pant and fairing are a glass fiber composite construction and do not require any servicing or maintenance. The wheel pants and fairings should be inspected periodically for cracking, chips, etc. and repaired and replaced as necessary. Contact the manufacturer for replacement parts. The interior of the wheel pant should also be cleaned of all foreign debris.

A few owners or operators have reported hairline paint cracking in the area adjacent to the nose wheel tow bar hole. While the paint cracking does not present a structural problem, the visual presentation of the wheel pant may degrade somewhat over several years. The onset of the cracking depends, in part, on the method used to tow the airplane and the frequency of towing. Enlarging the wheel pant hole may mitigate the onset of paint cracking since the tow bar will be less likely to apply a lateral force directly to the wheel pant. To enlarge the wheel pant hole, complete the following instructions.

1. Remove the nose wheel pant per instructions in this section.
2. Using a suitable tool (i.e. die grinder or drill with sanding barrel), carefully enlarge the size of the lugholes to 1.25 to 1.30 in. diameter. If necessary, the hole may be enlarged eccentrically to the current hole in the wheel pant to maintain a consistent gap between the tow lug and wheel pant.

3. Re-install the nose wheel pant per instructions in this section.

32-9. NOSE GEAR WHEEL AND FORK ASSEMBLY

a. **Disassembly – Wheel and Fork Assembly**

1. Raise the front of the airplane such that the nose wheel is completely clear of the ground. Remove the four AN5-12A bolts and washers attaching the nose wheel assembly to the base of the nose strut (see Figure 32 - 9).
2. Remove the wheel pant per the instructions in section 32-8a.
3. Remove the four MS24693-C279 countersunk screws and remove the towbar plates and blocks from the fork (see Figure 32 - 11).
4. Secure the nose wheel assembly in a soft jaw vise or equivalent fixture. Remove the nose wheel axle nut, bolt, and two washers. Using a plastic mallet, gently tap inward on the fork area around the axle bushing areas until the bushings become dislodged from the fork. Remove the bushings, wheel assembly, and axle spacer from the fork (see Figure 32 - 11).

b. **Reassembly – Wheel and Fork Assembly**

1. Secure the nose wheel fork in a soft jaw vise or equivalent fixture with fork legs open and unobstructed.
2. Insert the axle spacer into the wheel. Install the tire and wheel assembly into the nose wheel fork. Insert the axle bushings into the fork with the axle spacer inserted into the bushings (see Figure 32 - 11).
3. Install the axle bolt, nut, and two washers in reverse order as disassembled, but do not tighten the axle nut (see Figure 32 - 11).
4. With the axle bushings partially installed, check the clearance between the outside diameter of the tire and the edge of the nutplate on the fork (see Figure 32 - 12). Clearance must not be less than 0.25 in. If clearance is less than 0.25 in., replace the 5.00-5, 10 ply tire with one that has an inflated outside diameter of 14.20 in. or less per TSO-C62C.
5. Rotate the axle bushings so that the flat edges of the bushings are parallel with the edge of the tow bar blocks (blocks prevent axle bushings from rotating within the fork). If necessary, temporarily install the tow bar plates to the fork to assist alignment of bushings during installation.

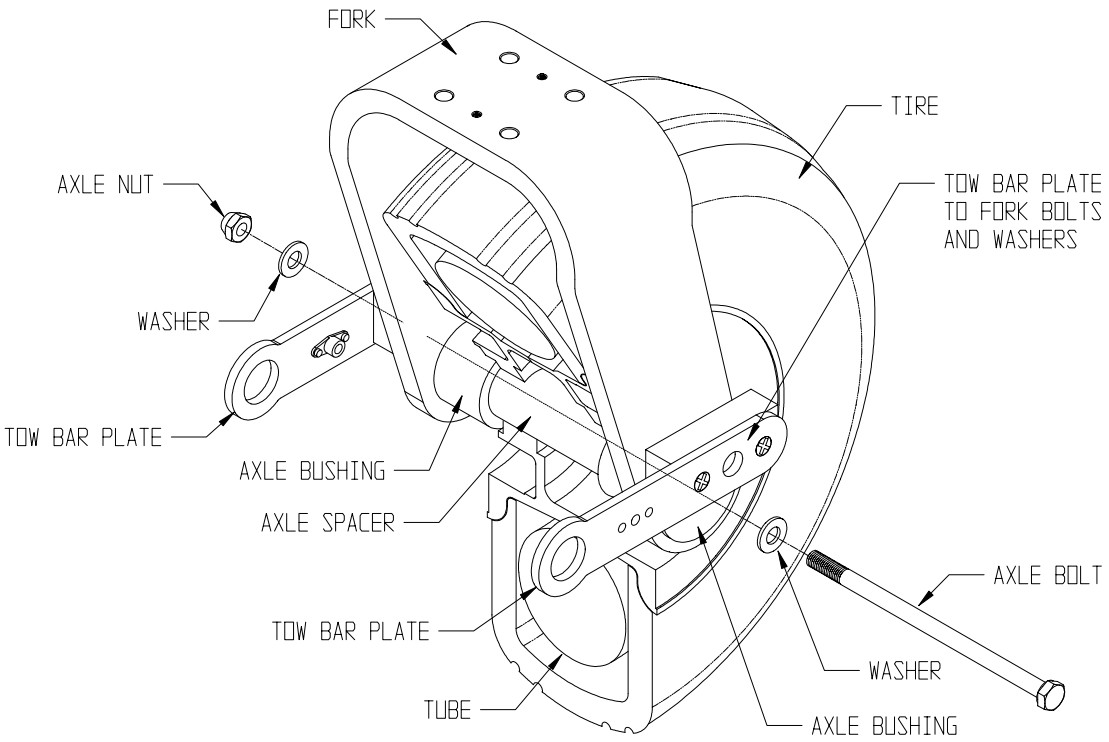


Figure 32 - 11 Nose Gear Wheel and Fork Assembly

6. Tighten the axle nut until the axle-bushing flanges are fully seated against the outside surface of the fork. If necessary, lightly tap the fork legs outward with a plastic or rubber mallet while tightening the axle bolt. Do not allow bushings to become cocked or jammed. If bushings become hard to install, stop and remove the bushings to check for any possible damage to the bushing or fork bushing holes. Replace all damaged parts. Do not file, grind, polish or sand bushings or fork bushing holes. Do not install a warm bushing into a cold fork. If necessary, place the axle bushings in a freezer for one hour and submerge fork in hot water for ten minutes prior to assembly. Anti-seize compound may be used to help install axle bushings into the fork.

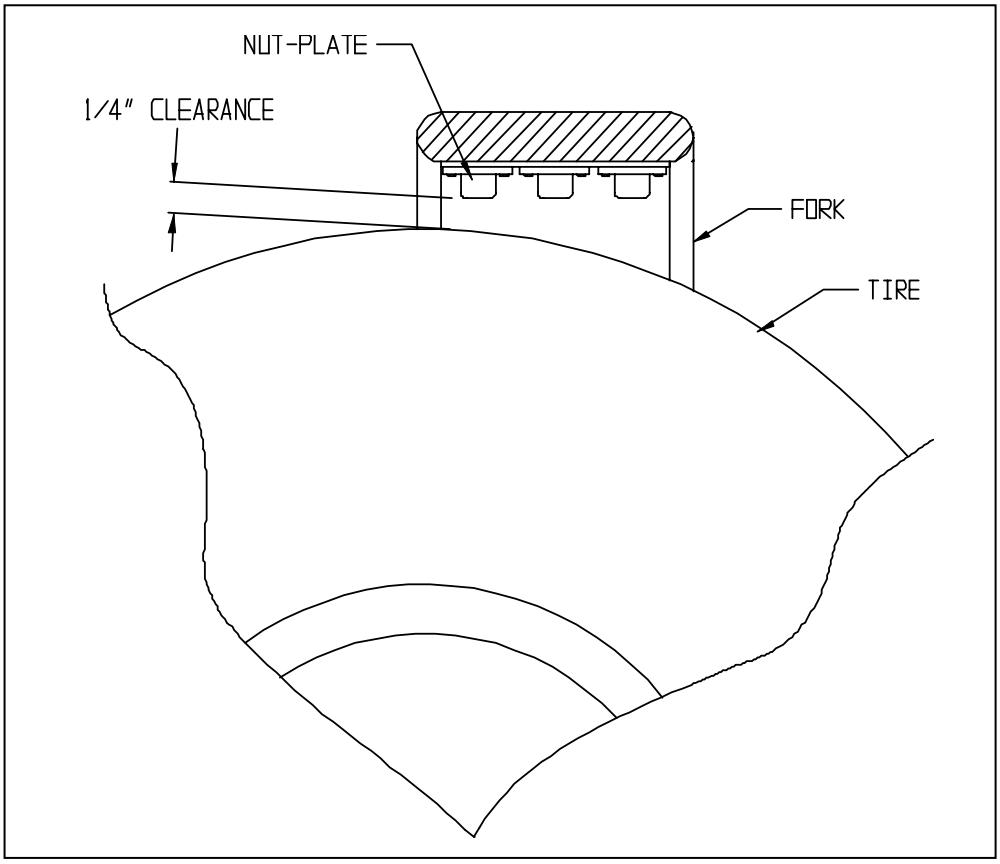


Figure 32 - 12 Minimum Tire Clearance

7. With axle bushings fully installed, tighten nut until all side play is eliminated from wheel bearings, then continue tightening an additional ¼ turn. The wheel should have a slight drag when the bearings are properly preloaded. Verify that the wheel bearings are fully seated by attempting to rock the wheel from left to right. No movement or "play" should be felt.
8. With towbar plates and blocks in place, torque countersunk screw 30 to 36 in.-lbs. See Figure 32 - 11.
9. Attach the nose wheel assembly to the base of the nose strut with the four AN5-12A bolts and washers (see Figure 32 - 9). Torque the bolts 140 to 180 in.-lbs.
10. Install the nose wheel pant over the fork and wheel assembly. Install four AN526C1032 wheel pant to fork attachment screws and washers in reverse order as disassembled. Torque screws 30 to 36 in.-lbs.

c. **Servicing and Maintenance**

The nose wheel assembly does not require any service or maintenance. The nose wheel assembly should be inspected for corrosion and other damage annually or every 100 hours. Damaged parts should be replaced as required. The nose wheel is to be serviced per the procedures outlined in Chapter 12.

32-10. NOSE STRUT

The nose gear may be equipped with either the Cessna nose strut or the ESCO nose strut. The Cessna strut nitrogen gas inlet schrader valve is located at the bottom rear of the nose strut. The ESCO strut has "ESCO" clearly labeled and the nitrogen gas inlet schrader valve is located at the upper front of the strut.

a. **Cessna Nose Strut**

1. **Removal of the Nose Strut**

a) Remove the nose strut fairing per the instructions in section 32-8a.

b) Relieve all load from the nose gear by pushing down on the tail of the aircraft.

c) Remove the valve cap from the strut schrader valve. Using a small flat bladed screwdriver covered by a rag, gently press in on the schrader valve core stem to slowly bleed off nitrogen gas pressure in the strut.

CAUTION

Always wear safety goggles when releasing gas pressure.

d) Using a felt tip marker, apply an aligning index mark between the engine mount and the strut housing.

e) Remove the bolts, nuts (either castellated or self-locking) and washers connecting the nose strut to the engine mount (see Figure 32 - 13).

f) Using a twisting motion, pull the nose strut straight down and out of the engine mount.

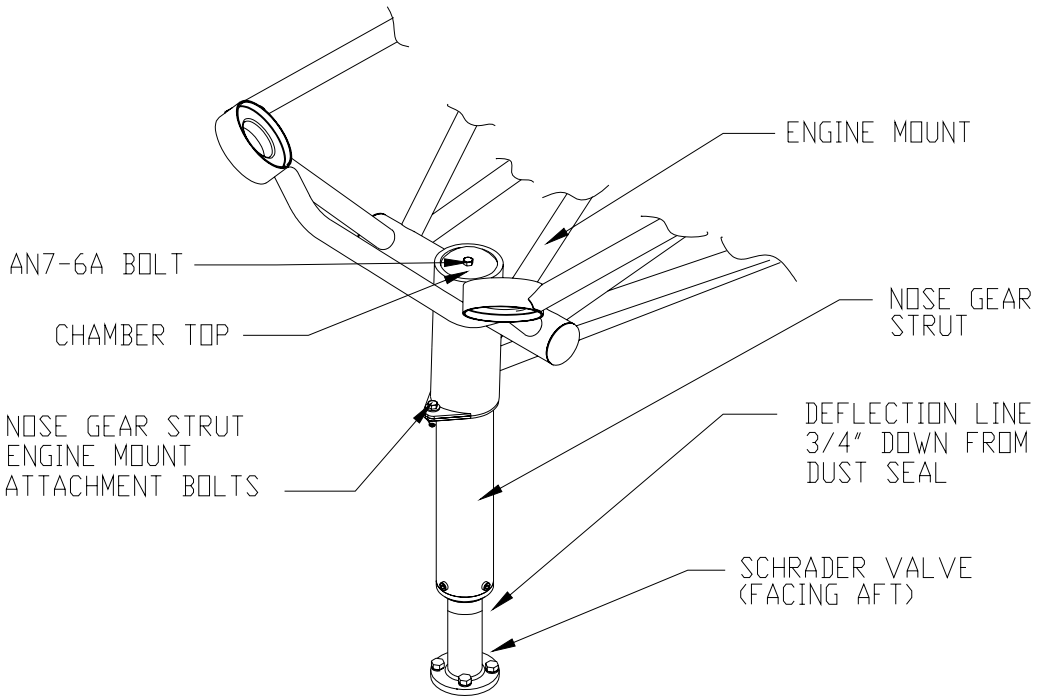


Figure 32 - 13 Cessna Nose Strut Assembly

2. **Installation of Nose Strut**
   a) Using a twisting motion, slide the nose strut housing into the engine mount until the housing collar is seated flush against the engine mount tube and index marks are aligned.
   b) Install the bolts, nuts, and washers connecting the nose strut to the engine mount in reverse order as disassembled (see Figure 32 - 13). Torque nuts 72 to 84 in.-lbs. Install new cotter pins in the castellated nuts.
   c) Service the nose strut per the instructions in Chapter 12.
   d) Reinstall the nose strut fairing per the instructions in section 32-8a.
   e) Reinstall the nose wheel assembly per the instructions in section 32-8.
3. **Inspection and Servicing**
   a) Inspection (every 100 hours or annually)
      1) The nose gear strut should be inspected for cracks, corrosion, scratches on the strut shaft, fluid leakage, and signs of overstress. The strut attachment to the engine mount should be checked for security. If oil leakage is present at any one of these locations, replacement of the strut is required.
      2) With the full weight of the aircraft on the nose and main gear inspect the ride height of the nose gear strut. There should be 3 to 4 in. of the chrome strut rod visible between the lower flange and the wiper seal. If ride height is low and no oil leakage is present, recharge the strut per Chapter 12. Recheck the filler valve and the filler valve gasket for signs of leakage.
      3) Remove weight off the nose gear to allow rotation of the nose wheel assembly left and right to its full range of rotation. A smooth resistance from the internal hydraulic shimmy damper device should be felt throughout its rotation. If binding is felt during rotation service the strut per Chapter 12. If the strut has been properly serviced and continues to bind, replacement of the strut is required.
      4) Rotate the nose gear wheel assembly to one side and remove just enough weight off the nose gear so the strut is fully extended and the wheel assembly is no longer in contact with the floor. If the wheel assembly does not center, in line with the direction of flight, service the strut per Chapter 12. If the strut has been properly serviced and does not center, replacement of the strut is required.
   b) Servicing (every 200 hours or annually) – The strut should be serviced per the instructions in Chapter 12.

b. **ESCO Nose Strut**

1. **Removal of Nose Strut**
   a) Remove the nose strut fairing per the instructions in section 32-8a.
   b) Relieve all load from the nose gear by pushing down on the tail of the aircraft.
   c) Remove the valve cap from the strut schrader valve. Using a small flat bladed screwdriver covered by a rag, gently press in on the schrader valve core stem to slowly bleed off nitrogen gas pressure in the strut.

CAUTION

Always wear safety goggles when releasing gas pressure. Apply rags around the schrader valve and a drip pan under the strut to catch any oil that may discharge with the nitrogen gas.

d) Remove the schrader valve from the strut housing.
e) Using a felt tip marker, apply an aligning index mark between the engine mount and the strut housing.
f) Remove the bolts, nuts (either castellated of self-locking), and washers connecting the nose strut to the engine mount (see Figure 32 - 14).
g) Using a twisting motion, pull the nose strut straight down and out of the engine mount.

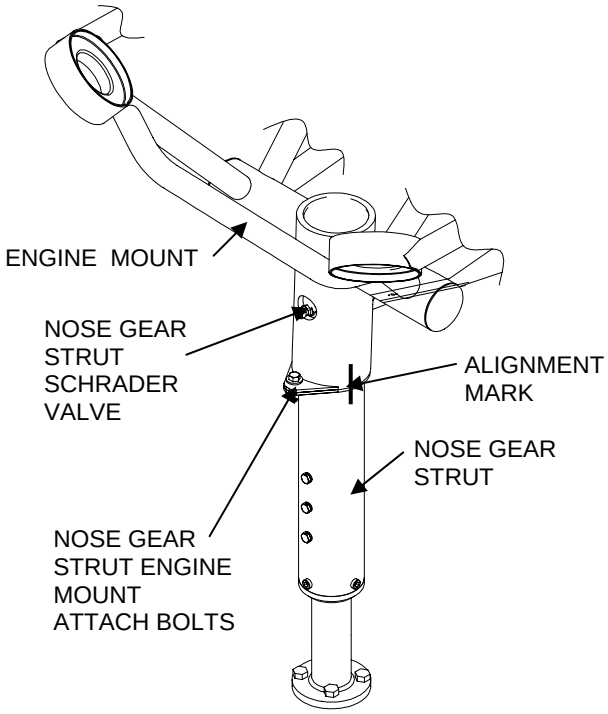


Figure 32 - 14 ESCO Nose Strut Assembly

2. **Installation of Nose Strut**
   a) Using a twisting motion, slide the nose strut housing into the engine mount until the housing collar is seated flush against the engine mount tube and index marks are aligned.
   b) Install bolts, nuts, and washers connecting the nose strut to the engine mount in reverse order as disassembled (see Figure 32 - 14). Torque nuts 72 to 84 in.-lbs. Install new cotter pins in the castellated nuts.
   c) Screw the schrader valve into the nose strut and torque the valve 150 to 200 in.-lbs.
   d) Service the nose strut per the instructions in Chapter 12.
   e) Reinstall the nose strut fairing per the instructions in section 32-8a.

f) Reinstall the nose wheel and wheel pant assembly per the instructions in section 32-8.

3. **Servicing and Maintenance**
   a) Maintenance (every 100 hours or annually)
      1) The nose gear strut should be inspected for cracks, corrosion, scratches on the strut shaft, fluid leakage, and signs of overstress. The strut attachment to the engine mount should be checked for security. If oil is present at any one of these locations, replacement of the strut is required.
      2) With the full weight of the aircraft on its nose and main gear inspect the ride height of the nose gear strut. There should be 3 to 4 in. of the chrome strut rod visible between the lower flange and the wiper seal. If ride height is low and no oil is present, recharge the strut per Chapter 12. Recheck the filler valve and the filler valve gasket for signs of leakage.
      3) Remove weight off the nose gear to allow rotation of the nose wheel assembly left and right to its full range of rotation. A smooth resistance from the internal hydraulic shimmy damper device should be felt throughout its rotation. If binding is felt during rotation service the strut per Chapter 12. If the strut has been properly serviced and continues to bind, replacement of the strut is required.
      4) Remove just enough weight off the nose gear so the strut is fully extended and the wheel assembly is no longer in contact with the floor. If the nose wheel assembly does not center, in line with the direction of flight, service the strut per Chapter 12. If the strut has been properly serviced and does not center, replacement of the strut is required.
   b) Servicing (every 200 hours or annually) – The strut should be serviced per the instructions in Chapter 12.

32-11. HARD LANDING INSPECTION

a. After a hard landing or a landing over maximum landing weight, the aircraft must be inspected for damage in accordance with this section. If any damage is found, it must be repaired prior to the next flight.

1. Nose Gear – The nose gear strut should be inspected for cracks, scratches on the strut shaft, fluid leakage, and signs of overstress. Check for proper nitrogen pressure in the strut. The strut attachment to the engine mount should be checked for security. If oil leakage is present at any one of these locations, replacement of the strut is required.

   Inspect the ride height of the nose gear strut. There should be 3 to 4 in. of the chrome strut rod visible between the lower flange and the wiper seal. If ride height is low and no oil leakage is present, recharge the strut per Chapter 12. Recheck the filler valve and the filler valve gasket for signs of leakage.
2. Main Gear – Inspect the wheel, axle, axle mounting bolts, wheel shims, gear leg, and gear leg attachment bolts to assure there are no broken, bent, or loose parts. The main gear assembly should fit snugly together with very little, if any, free play.
3. Wheel Toe-in and Camber – Assure that wheel toe-in and camber meets the requirements of section 32-4.
4. Wheel Pants and Fairings – Visually inspect the main and nose gear wheel pants and fairings for chips, cracks, or other damage.
5. Gearbox – Determine if the gearbox is bent or broken. With equal pressure and wear on the tires, the aircraft evenly loaded (including fuel), and the aircraft on a level surface, the wings should be level. If a bent or broken gearbox is suspected, contact the manufacturer, and do not fly the aircraft.
6. Tires – Visually inspect the tires for cuts, wear, or broken belts.
7. Engine Mount – Visually inspect the welds on the engine mount using a bright light and magnifying glass.
8. Propeller – Visually inspect the propeller for nicks or chips.

32-12. WHEELS, TIRES, AND BRAKES – GENERAL

The Cessna 350 (LC42-550FG) uses a 5.00-5 nose wheel and a 6.00-6 main wheel. The nose and main wheels are both tube type wheels. The nose wheel uses a 5.00-5 10-ply tire and the main wheel uses a 15x6.00-6 6-ply tire. Wheels and tires are certified under the requirements of TSO-C26c, and all replacement parts should be in accordance with the TSO.

32-13. WHEELS, TIRES, AND BRAKES

All service and maintenance practices for the main wheels are in accordance with the latest issue of Cleveland Wheels and Brakes Maintenance Manual AWBCMM0001. The manual includes the disassembly and reassembly of the wheel and brake disk, the installation of tires and tubes, and the service and repair of wheel bearings and bearing races. Service and maintenance of the brake calipers and brake master cylinders is also covered in the Cleveland maintenance manual.

32-14. BRAKE SYSTEM

The Cessna 350 has dual brakes located on the main gear. The brakes are hydraulically actuated and used for steering (differential braking) as well as stopping.

a. **Removal and Installation of the Brake Calipers.**

1. Gain access to the brake calipers by removing the main gear wheel pant access panel.
2. Remove the four bolts from the brake caliper that hold on the brake pads. Pads will drop away from behind the wheel disc.
3. Slide the brake caliper straight out from the torque plate.

CAUTION

Depressing the brake pedals at this point will push the brake caliper pistons out of the calipers and release brake fluid.

4. Disconnect the brake line from the brake caliper. Once the brake line is removed, use a plug or cap to seal the end of the brake line to minimize the amount of brake fluid that will drain from the line.
5. Reinstall the brake calipers in the reverse order. Torque the brake pad attachment bolts per the latest issue of Cleveland Maintenance Manual AWBCMM0001.
6. Service the brake system per the instruction in section 32-14c.

b. **Removal and Installation of the Brake Master Cylinders**

The brake master cylinders are located in the cabin. There are four master cylinders, one at each rudder pedal. The brakes are actuated by depressing the rudder pedals, which in turn depresses the brake master cylinders. Use the following procedure for removing and installing the master cylinders.

1. Remove the main gear wheel pant access panel to gain access to the brake caliper.
2. Place a small pan or bucket under the brake caliper, and remove the lower bleeder valve.
3. Depress the rudder pedals several times to facilitate draining of the brake system.
4. Once the brake system is drained of hydraulic fluid, disconnect the brake lines from the brake master cylinder.

NOTE

Place some rags or a small pan underneath the master cylinder and rudder pedal to catch any excess hydraulic fluid that may still be present in the line.

5. Remove the cap plug, cotter pins, nuts, washers, and bolts attaching the master cylinder to the rudder pedal and the rudder crossover tube (see Figure 32 - 15 and Figure 32 - 16).

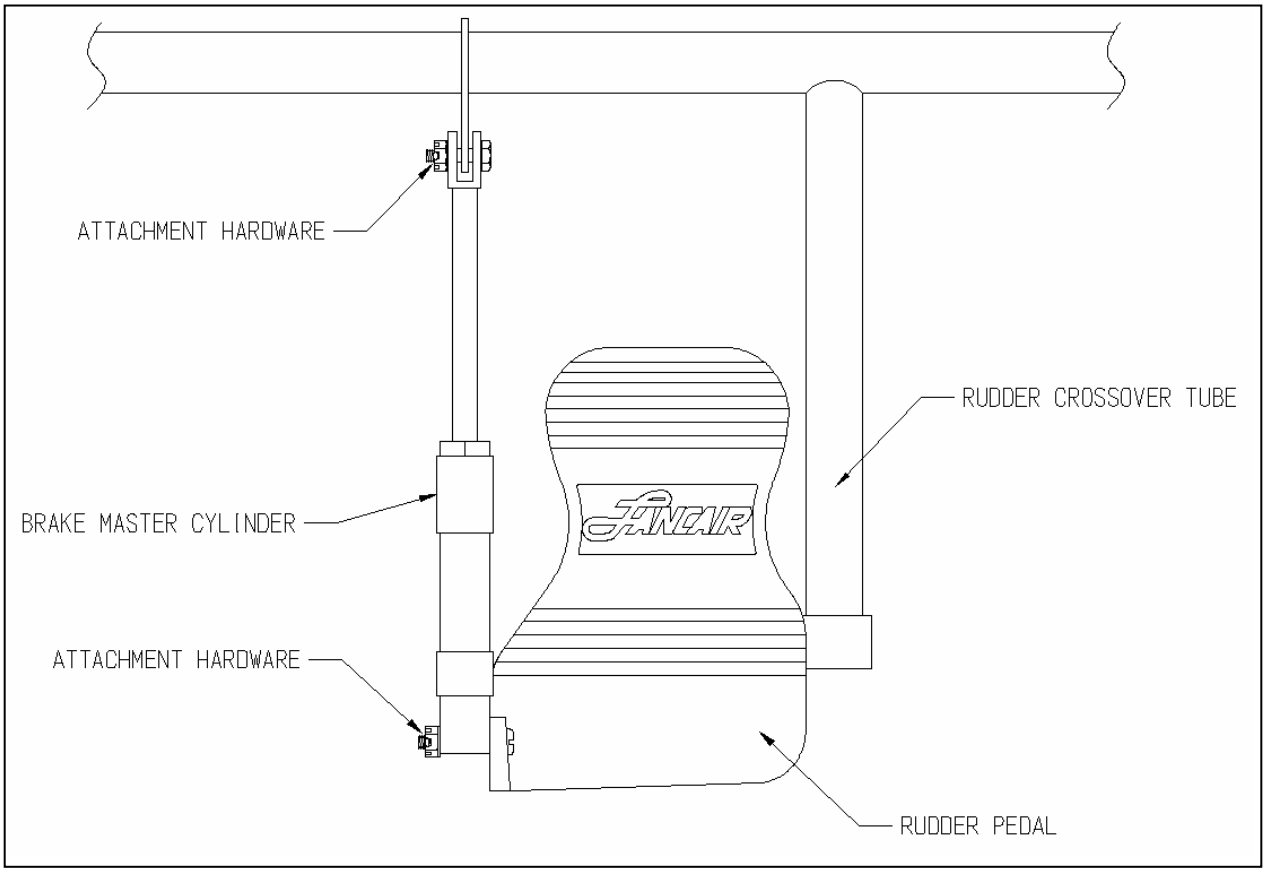


Figure 32 - 15 Rudder Pedal and Brake Master Cylinder Attachment

6. Install the brake master cylinder in the reverse order. Check all fittings and attachments for security. Fill a new MS-3 cap plug half full with Wacker SWS-951 sealant, and install over the nut with the cotter pin.
7. After the master cylinder is re-installed, re-fill the brake system with hydraulic fluid, MIL-H-5606, per the instructions in section 32-14c.

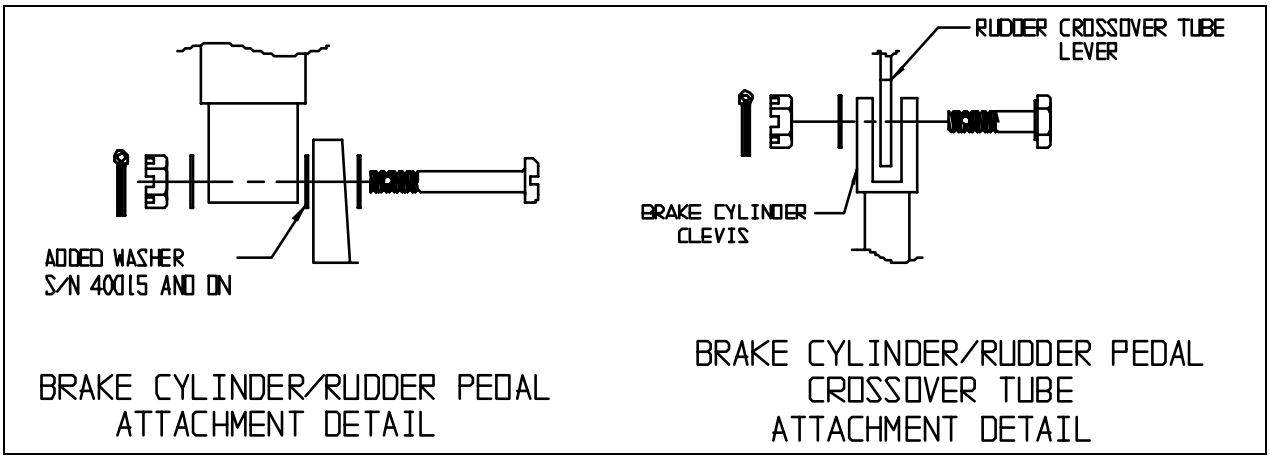


Figure 32 - 16 Rudder Pedal and Brake Master Cylinder Attachment Details

c. **Servicing and Maintenance of the Brake System** – The brake system is a self-contained system and under normal operating conditions should not need any service or maintenance. The system will need to be serviced if the brake calipers or other components are replaced. If the brakes begin to feel "soft", the brake system may have air in the lines and will need to be serviced. The brake system will also need to be serviced if any of the brake lines are disconnected. Servicing the brake system will require the use of a small "garden sprayer" with a modified nozzle or other similar device. The spray nozzle should be replaced with a small section of tubing that can be fitted to the brake caliper bleeder valve. Use the following procedure for bleeding and servicing the brake system.

WARNING

Only service the brake system when the brake calipers are attached to the wheel and disc assembly. If the brake system is pressurized with the caliper unattached to the wheel and disc, the caliper pistons will be pushed out of the caliper.

1. Place a small bucket or pan under the drain valve located at the rear of the lower engine cowl.
2. Gain access to the brake caliper by removing the inner wheel pant access panel.
3. Loosen the bleeder valve located on the bottom of the brake caliper.
4. Fill the "garden sprayer" with MIL-H-5606 hydraulic fluid, and connect the end to the bleeder valve. Pressurize the sprayer.
5. Depress the handle to release hydraulic fluid into the brake system.
6. Depress the brake pedal several times until all air is forced out of the system. When all the air is forced out of the system, the brakes will feel "firm".
7. Once the brake system has been serviced, tighten the bleeder valve prior to removing the "sprayer".
8. Clean any excess brake fluid that may have spilled on or near the brake caliper, wheel, and brake disc.
9. Reinstall the wheel pant access panel.

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32-15. THE PARKING BRAKE

The Cessna 350 uses a parking brake to lock the brakes and prevent the aircraft from rolling. The parking brake is located on the left side of the cabin just forward of the circuit breaker panel.

a. **Removal and Installation**

1. Remove the parking brake interior panel per the instructions in Chapter 25.
2. Drain the hydraulic system per the instructions in section 32-14b.
3. Disconnect the brake lines from the parking brake. Place a rag under the brake lines to prevent any excess brake fluid that may still be in the lines from leaking onto the carpet.
4. Remove the two AN526C1032R24 truss head screws and two NAS1149F0363P washers that attach the parking brake (see Figure 32 - 17), and slide the brake and spacer straight out.
5. Reinstall the parking brake in the reverse order. Torque the attachment screws 30 to 36 in.-lbs.
6. Service the brake system per the instructions in section 32-14c.

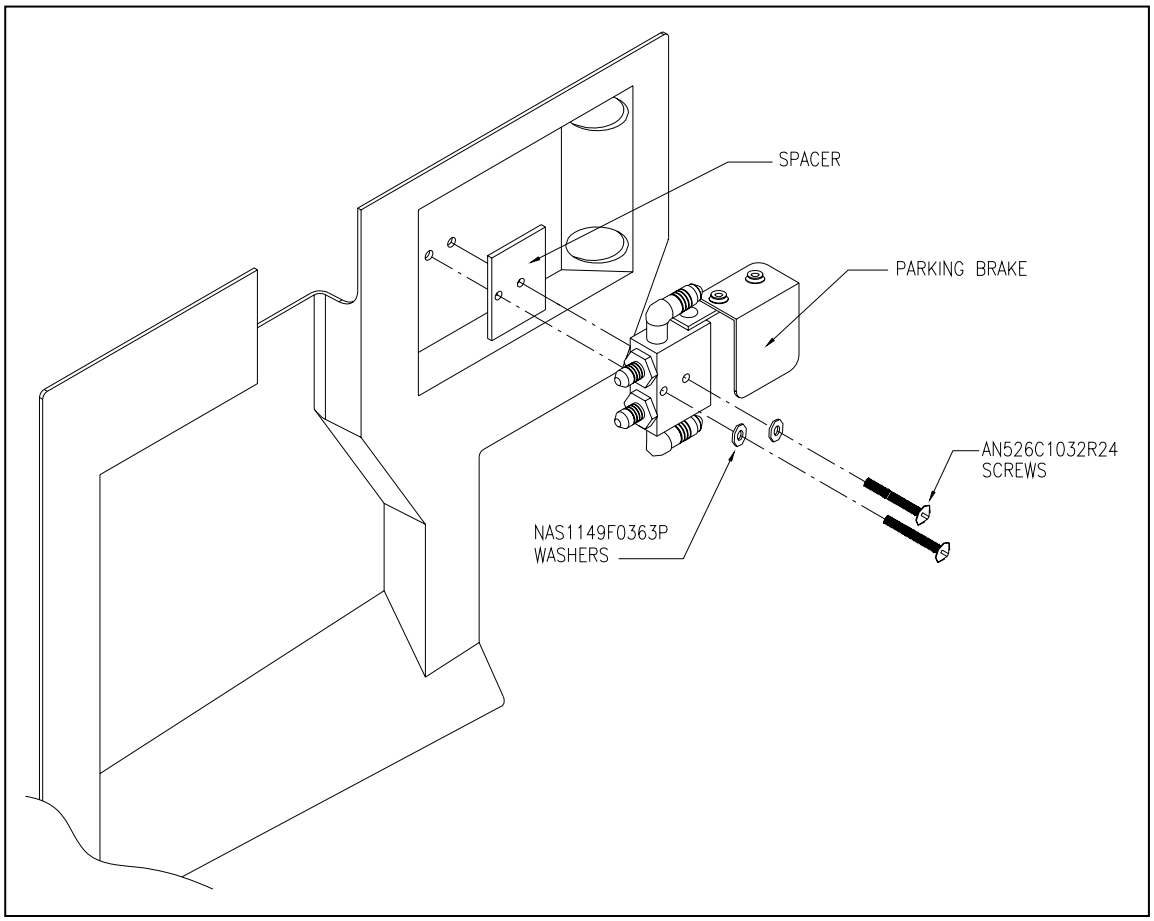

Figure 32 - 17 Parking Brake Installation

b. **Service and Maintenance** – The parking brake is a sealed system and should not need any service or maintenance. The parking brake should be inspected periodically to ensure that it is still functioning properly. Use the following procedure to check if the parking brake is functioning properly.
   1. Depress the rudder pedals to actuate the brakes.
   2. While pressing on the rudder pedals, pull the parking brake lever out and towards the pilot's seat to engage the parking brake and lock the brakes.
   3. Using the tow bar, attempt to pull the airplane forward. If the airplane will not roll or the tires skid, the parking brake is functioning correctly. If the airplane does roll forward with little or no resistance the parking brake should be inspected for proper attachment of the brake lines (see Figure 32 - 18) and possible service of the parking brake itself. Service of the parking brake is in accordance with the latest issue of Cleveland Maintenance Manual AWBCMM0001.

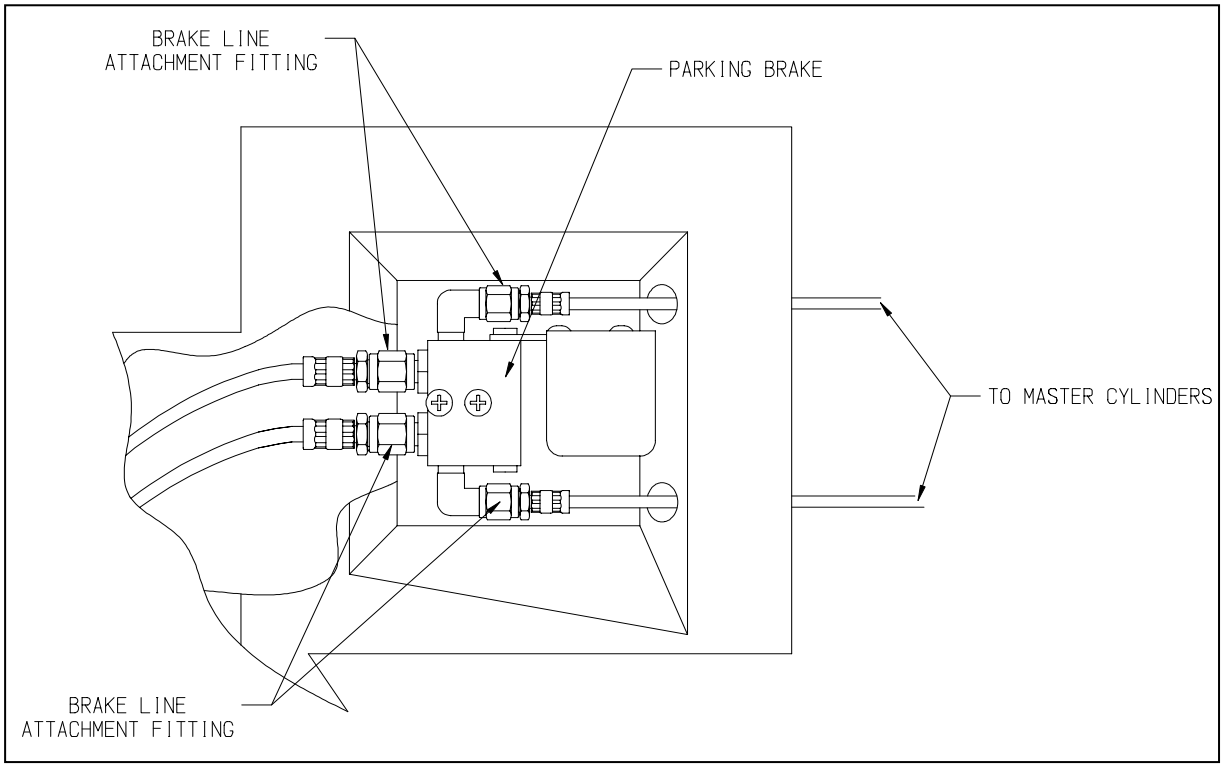

Figure 32 - 18 Parking Brake and Brake Line Attachment

CHAPTER 33

LIGHTS

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Chapter 33

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33-1. INTERIOR LIGHTING SYSTEM – GENERAL

a. The interior lighting system of the Cessna 350 is one of the more sophisticated systems available for small airplanes. There are six different lighting systems within the interior of the airplane. For a summary of these lighting systems see Figure 33 - 1 or Figure 33 - 2. For information about trim, annunciators, and flap position LEDs, see Chapter 25. For information about the fuel tank LED's, see Chapter 28.

b. **Flip & Access Lights** – The flip-lights are rectangular shaped fixtures located in the middle of the overhead panel and in the baggage compartment. The lights bypass the system master switch and operate without turning on power to the system. The two flip-lights are turned on by rotating or flipping the lens right or left. In the center position, they are used as part of the airplane's access lighting system.

   1. When either main entrance door is unlatched, a switch in the door latching mechanism activates the two flip-lights and the two lights that illuminate each entrance step. The access lights are on a ten-minute timer and turn off automatically unless reset by activating both main door latching mechanisms when all the doors are closed.
   2. This design has two advantageous features. First, opening either of the main cabin doors provides an immediate light source for preflight operations, passenger access, and baggage loading. Second, the flip-lights, when rotated either left or right, serve as emergency lighting in situations which necessitate turning off the master switch. The only disadvantage is that the flip-lights can inadvertently be left on, depleting battery power. To prevent this from happening, ensure the flip-lights are in the centered or flush position when securing the airplane at the end of a flight.

c. **Overhead Reading Lights** Each light is on a swivel that can be adjusted to an infinite number of positions. The intensity of these lights is adjusted by moving the left thumb-wheel (basic or Avidyne option) or slide-type (Garmin G1000 option) dimmer switch in the center of the overhead panel, near the windshield. The dimmer has an on-off switch at the extreme aft position. Rotating the thumb-wheel, or sliding the slide, forward increases the light intensity. The airplane's position lights must be on for the overhead reading lights to operate.

   1. For S/N 42001 to 42003 there are three overhead reading lights, two in the forward overhead panel between the front seats and one in the aft overhead panel between the two backseats. Each light has its own switch.
   2. For S/N 42004 and on there are four overhead reading lights, two in the forward overhead panel between the front seats and two in the aft overhead panel between the two backseats.

d. **Instrument Flood Bar** – There is a tube array of either color-corrected lights or LEDs inserted under the glare shield. This indirect light source complements the backlighting in each instrument and facilitates the use of adjustable instruments such as the true airspeed indicator, directional gyro, and navigational instruments. The intensity of the lights can be adjusted by moving the right wheel-type (basic or Avidyne option) or slide-type (Garmin G1000 option) dimmer switch in the center of the overhead panel, near the windshield. The dimmer has an on-off switch at the extreme aft position. Rotating the thumb-wheel, or sliding the slide, forward increases the light intensity. The airplane's position lights must be on for the instrument flood bar lights to operate.

e. **Upper Instrument and Engine Panels**

1. **Basic or Avidyne Option** – The instruments in the flight and engine instrument panels have backlighting, i.e., a small light within each instrument case that illuminates its dial. The right thumb-wheel switch between the pilot's legs on the knee bolster, controls the dimmer for these lights. The dimmer has an on-off switch at the extreme down position of its rotation. Rotating the thumb-wheel up increases the light intensity. The airplane's position lights must be on for the upper instrument and engine panel lights to operate.
2. **Garmin G1000 Option –** The brightness of backlighting for the PFD, MFD, audio panel, and keypad are controlled by photo cells located on the devices. The brightness of backlighting for the backup flight instruments is controlled by the left slide dimmer switch at the front of the center console. The dimmer has an on-off switch at the extreme up position, and moving the slide down increases the light intensity.

f. **Lower Instrument Panel, Circuit Breaker Panel, and Rocker Switches**

1. **Basic or Avidyne Option –** The lower instrument, circuit breaker, and rocker switch panels contain switches and controls that have backlighting. The lighting illuminates the placards on or next to the breaker, switch, or control and the internally lighted engraved rocker switches. The dimmer for these lights is controlled by the left thumb-wheel switch between the pilot's legs on the knee bolster. The dimmer has an on-off switch at the extreme down position of its rotation. Rotating the thumb-wheel up increases the light intensity.
2. **Garmin G1000 Option –** Backlighting of the 5 pack (pitot heat, door seals, and optional equipment) switches, flap panel, lighted position bar, slide dimmer labels, master switches and circuit breaker panel is controlled by the right slide switch at the front of the center console. Backlighting of the fuel pump "armed" light is controlled by the position lights switch. The backlighting illuminates the placards on or next to the breaker, switch or control, and the internally lighted switches. The dimmer has an on-off switch at the extreme up position, and moving the slide down increases the light intensity. Backlighting of the pitot heat, door seals, and optional items switches will dim down to a preset value while all other lighting controlled by this switch will dim to zero.

LIGHT	LOCATION OF LIGHTS	LOCATION OF SWITCH	REMARKS
Courtesy Lights	• Front and rear flip lights in overhead console • Exterior lights near the right and left entrance steps	♦ If all doors are latched, flip light is activated by flipping the lens from the neutral position. ♦ If a door is unlatched, a switch activates flip lights when the lens is in the neutral position. ♦ Step lights are turned on with door switch.	▪ Door switch activates timer that turns off access lights after 10 minutes. ▪ Operates with master switch on or off
Overhead Swivel Lights	• Two overhead swivel lights in the front seat area • S/N 42001 to 42003 - One centered swivel light in the rear seat area • S/N 42004 and on - Two centered swivel lights in the rear seat area	♦ The left thumb wheel dimmer switch is in the overhead panel. ♦ S/N 42001 to 42003 - Individual switch at each light. ♦ S/N 42004 and on – No individual switch at each light.	▪ Master switch and navigation lights must be on for the system to operate.
Glare Shield Flood Bar	• Color corrected flood bar under the glare shield that lights the flight instruments and front panel areas	♦ The right thumb wheel dimmer switch is in the overhead panel.	▪ Master switch and navigation lights must be on for the system to operate. ▪ Part of the standby battery system
Upper Instrument Panel	• Provides backlighting for engine and flight instruments	♦ The right thumb wheel dimmer switch is in the knee bolster on the pilot's side.	▪ Master switch and navigation lights must be on for the system to operate.
Lower Inst. & Circuit Breaker Panels	• Provides backlighting for radios, switches, or placards next to switches, circuit breakers, and controls	♦ Left thumb wheel dimmer switch is in the knee bolster on the pilot's side	▪ Master switch and navigation lights must be on for the system to operate.
Trim, Flaps, Fuel Tank, & Annunciator Panel	• The trim position LED's are in the trim panel. • The flap position LED's are in the flap panel. • The fuel tank LED's are on the fuel quantity gauge. • Annunciator LED's are in the annunciator panel.	♦ The day/night and test features for these items are located in the trim panel, just above the engine control.	▪ Master switch and navigation lights must be on for the system to operate.

Figure 33 - 1 Lighting Systems (Basic or Avidyne Option)

LIGHT	LOCATION OF LIGHTS	LOCATION OF SWITCH	REMARKS
Courtesy Lights	• Front and rear flip lights in overhead console • Exterior lights near the right and left entrance steps	♦ If all doors are latched, flip-light is activated by flipping the lens from the neutral position. ♦ If a door is unlatched, a switch activates flip-lights when the lens is in the neutral position.	▪ Door switch activates timer that turns off access lights after 10 minutes. ▪ Operates with master switch on or off
Overhead Swivel Lights	• Two overhead swivel lights in the front seat area • Two overhead swivel lights in the rear seat area	♦ The left slide-type dimmer switch is in the overhead panel.	▪ Master switch must be on for the system to operate.
Glare Shield Flood Bar	• Flood bar under the glare shield which lights the flight instruments and front panel areas	♦ The right slide-type dimmer switch is in the overhead panel.	▪ Master switch must be on for the system to operate.
Upper Instrument Panel	• Provides backlighting for the backup flight instruments.	♦ The left slide-type dimmer switch is in the front of the center console.	▪ Master switch must be on for the system to operate.
Lower Inst. & Circuit Breaker Panels	• Provides backlighting for switches, or placards next to switches, circuit breakers, and controls	♦ The right slide-type dimmer switch is in the front of the center console.	▪ Master switch must be on for the system to operate.

Figure 33 - 2 Lighting Systems (Garmin G1000 Option)

33-2. EXTERIOR LIGHTING SYSTEM – GENERAL

a. **Description of System** – The exterior lighting system includes the position lights, the strobe or anti-collision lights, the landing light, and the taxi light. These lights are activated through use of switches in the rocker switch panel. The light system is protected by circuit breakers in the circuit breaker panel.

b. **Position and Anti-collision Lights** – The left and right position lights (red and green) are mounted on each wing tip. Each wing position light contains the required aft or rearward projecting white lights. The anti-collision lights are on each wing tip and contained within the same light fixture as the position lights.

c. **Taxi and Landing Lights** – The taxi and landing lights are contained in the leading edge of the left wing. The outboard bulb in the light housing is the taxi light that provides a diffused light in the immediate area of the airplane. The inboard bulb is the landing light, which has a spot presentation with a slight downward focus. The taxi and landing lights are sized for continuous duty and can be left on for operations in high-density traffic areas.

CAUTION

Before any maintenance, removal, or installation of equipment or light bulbs is performed, make sure both the master and avionics switches are in the off position. Before removing the light bulbs, be sure that they have not been in use for several minutes and have cooled sufficiently.

33-3. TROUBLESHOOTING LIGHT PROBLEMS

a. **Light Bulbs –** As a general rule, light bulbs that have a short life have good accessibility and can be exchanged easily. For example, most engine and flight instrument lights can be changed in 10 to 15 minutes depending on the location of the instrument. Light bulbs that have a long life require more time for access and replacement.

b. **Check for the easy solution –** If none of the lights in the flood bar work, it is unlikely that all 20 bulbs burned out at the same time. In this situation, there is clearly a problem with the supply to the flood bar, and the problem should be diagnosed from this perspective. Conversely, if the altimeter light is inoperative, but all other flight instrument lights are normal, the problem is probably a light bulb.

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33-4. FLIGHT COMPARTMENT LIGHTING

a. **Courtesy Lights (General)** – There are four courtesy lights. Two courtesy lights are located in the airplane's interior on the overhead console, and two are on the airplane's exterior, and are commonly referred to as step lights. The entrance step lights are located above and forward of the steps on both sides of the aircraft, on the underside of the wing-to-fuselage fairing. Each are easily accessible in the event of bulb burnout.

b. **Exterior Courtesy/Step Lights Removal**
   1. Remove the light assembly by gently prying it with a flat-head screwdriver. It should slide out with very little effort. See Figure 33 - 3.
   2. Gently pull the step light assembly out of the hole, and unplug it at the connector.

c. **Exterior Courtesy/Step Lights Installation or Replacement**
   1. Reconnect the connector and feed excess wire into the fuselage.
   2. Insert the light fixture into the hole, and push it in until it is flush with the surface.
   3. The exterior step light bulb is not replaceable. If a light fails, replacement of the whole assembly is required.

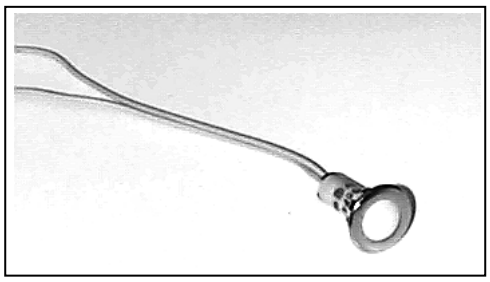

Figure 33 - 3 Step Light Assembly

CAUTION

Before doing any work on a system, ensure that all power and light switches are in the off position and the light that is to be replaced has not been in use for a least five minutes.

d. **Removing the Interior Courtesy Lights**
   1. The interior light housing can be removed by prying it off the panel with a screwdriver. See Figure 33 - 4 and Figure 33 - 5.
   2. Disconnect the three wires to the light housing.
   3. To remove the light bulb, push in the two tabs located to the left and right of the bulb.
   4. Push the bulb out of its slot to remove it.

e. **Installing the Interior Courtesy Lights**
   1. To install the interior light, place a new light bulb in the slot, and push it into place.
   2. Push the housing and the light cover together so that the two tabs click into place.
   3. Reconnect the three wires, and push the housing back into its slot in the panel. See Figure 33 - 4 and Figure 33 - 5.

f. **Troubleshooting the Interior Courtesy Lights**
   1. If the interior courtesy lights are not working, remove the housing from the panel, and make sure that all three wires are connected properly.

2. Check each light bulb for a failure.
3. Check the circuit protection for a blown fuse. The courtesy light system is connected directly to the battery (bypasses the master switch), and the fuse block is located next to the battery in the engine compartment. All fuses are labeled.

33-5. OVERHEAD SWIVEL LIGHTS

a. For S/N 42001 to 42003 there are three overhead swivel lights. Two are located in the front section of the cabin in the center of the overhead console. The third is located in the rear section of the cabin in the middle of the overhead console. For S/N 42004 and on there are four overhead swivel lights. Two are located in the front section of the cabin in the center of the overhead console and two are located in the rear section of the cabin in the middle of the overhead console These lights are also referred to as spot lights, map lights, or reading lights. Refer to Figure 33 - 4 and Figure 33 - 5 for the removal and installation of the overhead swivel lights.

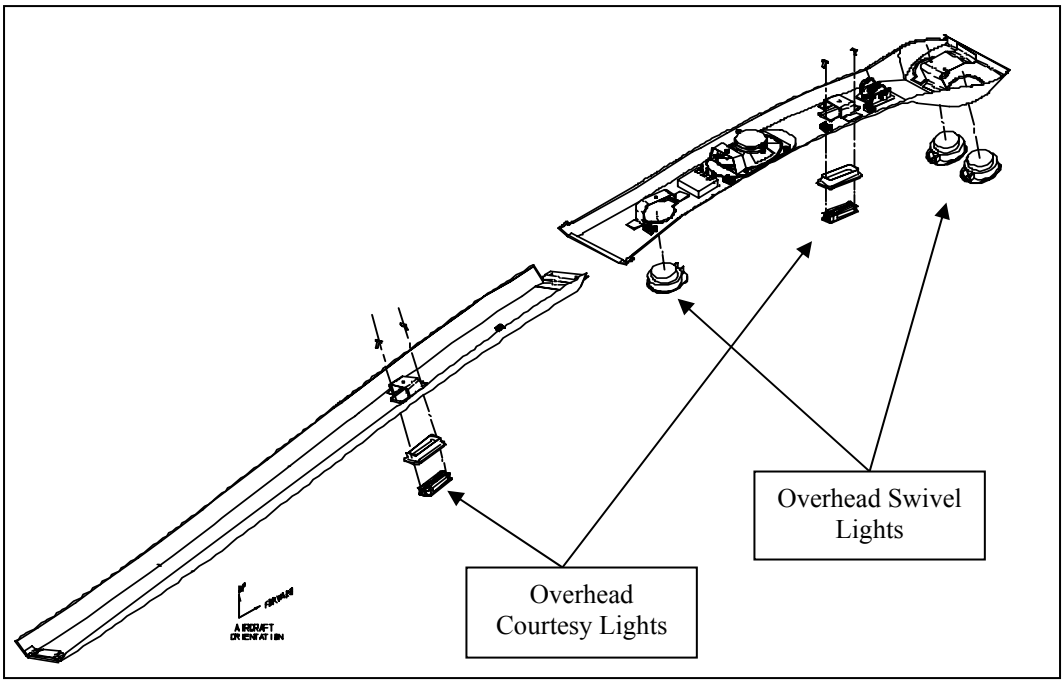

Figure 33 - 4 Overhead Lights (S/N 42001 to 42003)

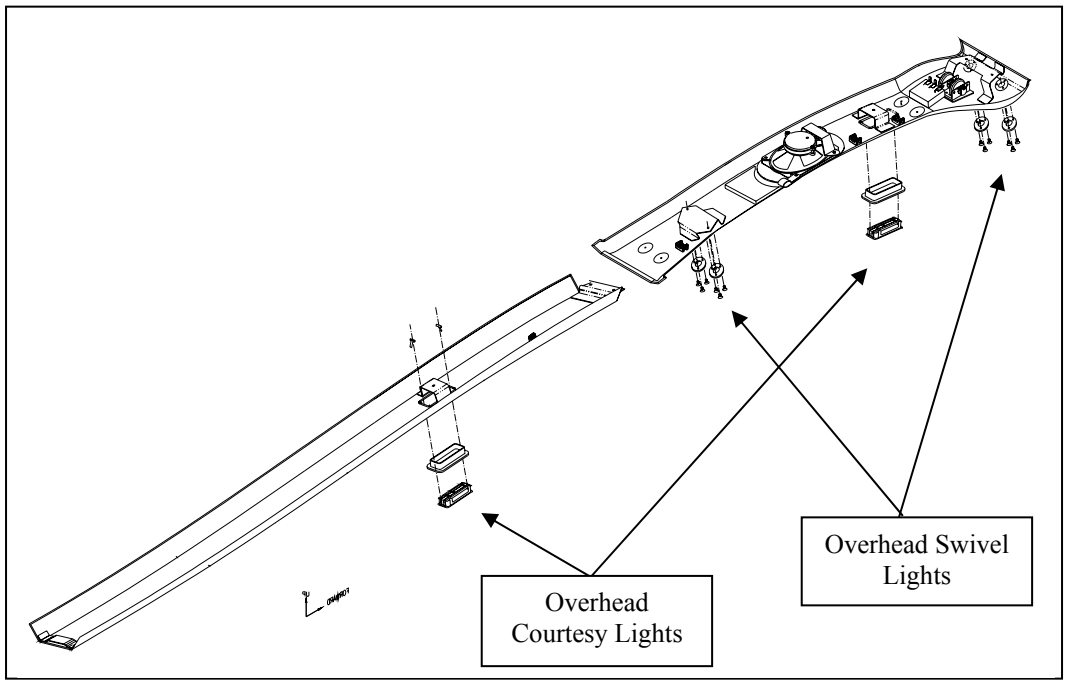

Figure 33 - 5Overhead Lights (S/N 42004 and on)

CAUTION

Before doing any work on a system, ensure that all power and light switches are in the off position and the light that is to be replaced has not been in use for a least five minutes.

b. **Removing the Overhead Swivel Lights**

1. S/N 42001 to 42003
    a) Remove the eyebrow cover from the swivel light by pulling it out of the light assembly. The eyebrow is pressure fitted into the assembly to limit side glare.
    b) The swivel lights can be removed from the panel by gently prying them with a screwdriver.
    c) Disconnect the two wires to the light housing. Push in the three plastic tabs that hold the light bulb housing together.
    d) To access the light bulb, push down on the housing, and turn it to remove the light bulb cover.
    e) Gently pull the light bulb out of the metal tabs to remove it.
2. S/N 42004 and on
    a) Unscrew three screws holding the swivel light in place.
    b) Remove the swivel light from the panel.
    c) Unplug the connector.

c. **Installing the Overhead Swivel Lights**
   1. S/N 42001 to 42003
      a) To install the swivel light, place a new light bulb in the slot, and gently push it in until the tabs hold the light bulb in place.
      b) Put the cover over the light bulb, push it down, and twist it until the cover and the housing latch together.
      c) Push the housing down into the frame until the three tabs click into place.
      d) Reconnect the two wires to the housing, and push the light back into its mounting hole in the panel.
      e) Reinstall the eyebrow cover.
   2. S/N 42004 and on
      a) Plug in the connector.
      b) Push the light into it's mounting hole in the panel.
      c) Screw in three screws to hold the swivel light in place.
      d) The overhead swivel light bulb is not replaceable. If a light fails, replacement of the whole assembly is required.

d. **Troubleshooting the Overhead Swivel Lights**
   1. S/N 42001 to 42003
      a) If the overhead swivel lights are not functioning properly, make sure that both of the wires are hooked up properly.
      b) Check the light bulb to be sure it is not burned out.
      c) Check the light dimmer assembly.
   2. S/N 42004 and on
      a) If the overhead swivel lights are not functioning properly, make sure the connector is plugged in correctly.
      b) Check the light to be sure it is not burned out
      c) Check the light dimmer assembly.

33-6. GLARE SHIELD FLOOD BAR

a. The glare shield flood bar is located under the glare shield in the cabin. The flood bar is composed of light emitting diodes (LED) spaced at 1/2 in. intervals. If more than eight LED burn out, a new flood bar must be installed if the airplane is to be used for IFR operations. Contact the factory for ordering a replacement flood bar assembly.

b. **Flood Bar Inspection/Airplane Restrictions** – With the flood bar turned on, use a mirror to count all the burned out LED. If 8 or more LED are burned out, then the flood bar must be replaced, or the airplane must be restricted to VFR operations. The flood bar is an integral component in the airplane's IFR certification. See the minimum equipment list in Appendix A, Section 6, of the POH/AFM.

c. **Removing the Glare Shield Flood Bar**
   1. Remove the glare shield per instructions in Chapter 25, and place it inverted on a soft work bench.
   2. Pull the flood bar to remove it from attachment clips and to break the adhesive from the glare shield.
   3. Clean excess adhesive off the glare shield.

d. **Replacing the Glare Shield Flood Bar**
   1. Glare shield should be inverted on a work bench.

2. Attach the flood bar to the underside of the glare shield at the front lip by snapping onto the attachment clips.
3. Use several strips of RTV sealant along the wiring to secure it in place.
4. Note that the wiring tail must hang from the left (pilot) side of the assembly.
5. Reinstall the glare shield per instructions in Chapter 25.

33-7. UPPER INSTRUMENT PANEL

a. **Field Repair Instrument Lights** – The field repair instrument lights are located in the engine and flight instruments. The upper instrument lights are broken into two groups, (1) flight instruments and (2) engine instruments. Flight instruments include: the air speed indicator, altimeter, vertical speed indicator, turn coordinator, directional gyro, and artificial horizon. Engine instruments include: fuel quantity, manifold pressure/fuel flow, vacuum/ammeter, oil pressure/temperature, and cylinder head/exhaust gas temperature gauges. The magnetic compass is also included with the upper panel discussion.

b. **Factory Repair Instrument Lights** – There are a few flight, engine, and navigational instruments that have sealed lights, i.e., lights that cannot be accessed in the field. These instruments include the flight director, horizontal situation indicator (HSI), navigational indicator (OBS), and the tachometer. The lights in these instruments are designed to last several thousand hours and, under normal conditions, should never require replacement. If a light does become inoperable, and it is confirmed that there is power to the instrument, then the instrument must be removed and sent to an instrument shop for bulb replacement.

c. **Removal and Replacement of Flight Instrument Lights** – These instruments use a light tray for lighting. To change a bulb use the following steps:
   1. Remove the glare shield, and partially remove the instrument panel to gain access to the desired flight instrument. See Chapter 25 for details.
   2. If necessary, remove the flight instrument panel by removing the screws in the instrument panel.
   3. Unscrew the two screws holding the light tray to the top of the instrument. The whole light tray must be replaced since the bulb is soldered to it. See Figure 33 - 6.
   4. Once the light tray is replaced, reverse the above steps for installation of the instrument. The screws holding the instrument to the instrument panel do not have torque values and should be snug tight.

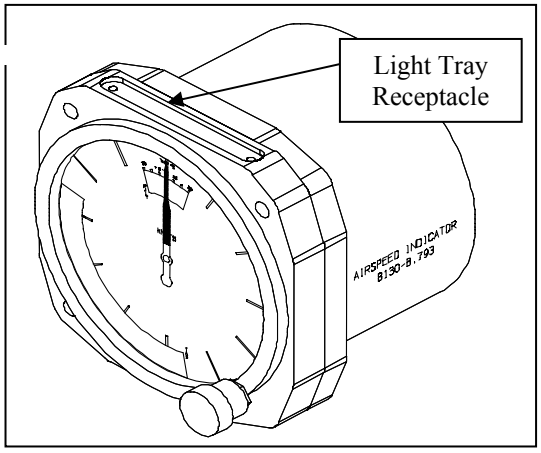

Figure 33 - 6 Light Tray-Type of Instrument

d. **Removal and Replacement of Engine Instrument Lights (Basic or Avidyne Option Only)**
   1. Remove the engine instrument panel per instructions in Chapter 25.
   2. If necessary, remove the engine instrument by removing the screws in the instrument panel.
   3. To change the light bulb, unscrew the lights on top of the instrument ¼ turn, and screw in a new light. See Figure 33 - 7.
   4. Once the engine instrument light is replaced, reverse the above steps for reinstallation of the instrument. The screws holding the instrument to the instrument panel do not have torque values and should be snug tight.

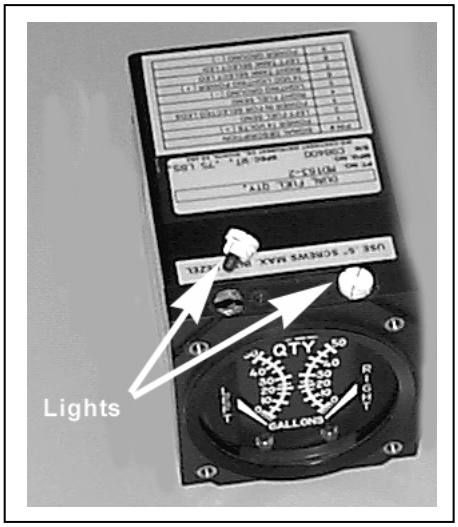

Figure 33 - 7 Engine Instrument

e. **Magnetic Compass** – The compass light is an integral part of the compass and cannot be replaced. If the light is inoperative, replacement of the entire compass assembly is required.

33-8. LOWER INSTRUMENT PANEL

a. **Lower Instrument Lights** – The lower instrument panel lights include switches in the ignition panel, rocker panel, ECS panel, and circuit breaker panel.

b. **Ignition, Rocker, and ECS Panels** – There are two rocker switches in the ignition panel, eight switches in the rocker panel, and one switch in the ECS panel. The light bulb in these switches is soldered into the unit, which means the switch must be replaced if the bulb in the switch burns out.

c. **Rocker Switch Removal and Installation** – Use the following steps to remove and install a switch. See Figure 33 - 8.
   1. Remove the panel per instructions in Chapter 25.
   2. Identify the rocker switch that is to be removed, and label each wire in relationship to its spade connector.
   3. After each wire is labeled, remove it from the switch.
   4. To remove the switch, push it out from the back of the panel. To do this, simultaneously apply pressure to the top and bottom (usually the thumb and index finger) across the width of the switch to compress the four retaining clips.
   5. Install the switch by compressing the retaining clips and inserting it into the panel.
   6. Connect the wires to the spade connectors, and with the master switch on, test the switch for proper operation.

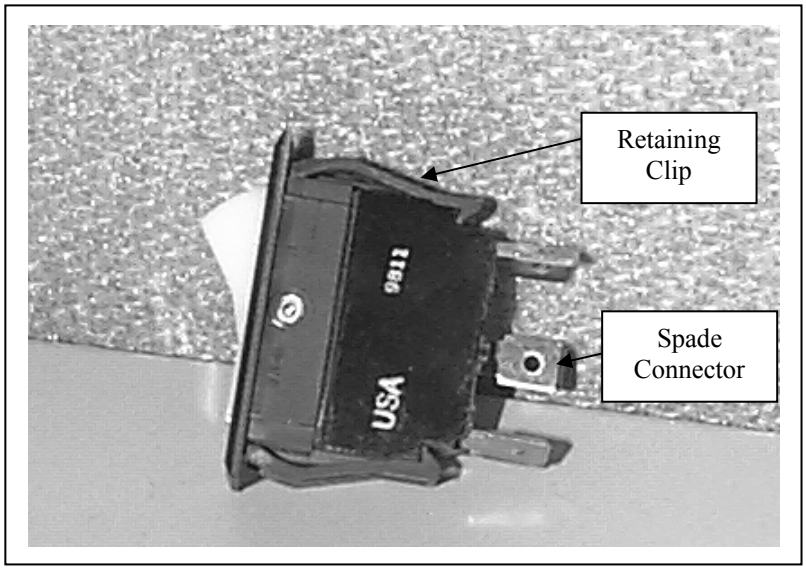

Figure 33 - 8 Rocker Switch

d. **Circuit Breaker Panel Lights (Overview)** – The circuit breaker panel is located on the left side of the airplane, just forward of the pilot's seat. The panel contains five tiered rows. Above each row is a small overhang that spans the width of the row. See Figure 33 - 9.

Within each overhang there are light elements, which are epoxied into holes in the overhead row. The number of lamps in each row depends on the length of the row. Rows 1-3 have three lamps, row 4 has four lamps, and row 5 has five. The bulbs are soldered into place in series, in pairs of two. This means that if one bulb in a pair burns out, the other bulb in series will not function.

e. **Removal and Replacement of Circuit Breaker Panel Lighting**

1. Remove the circuit breaker panel per instructions in Chapter 25.
2. Remove the two large bus cables, disconnect the three black connectors, and remove the circuit breaker panel. Be sure to label each cable and connector location before disconnecting it.
3. To gain access to the bulbs in rows 2-5, remove the required circuit in the row one number lower than the row with the malfunctioning light. For example, if the lights in row three are inoperative, remove the required circuit breakers (for access to the lights) in row two.
4. Break the burned out light bulb and the companion series bulb out of their epoxied holes, and remove excess epoxy and fragments.
5. Solder replacement bulbs.
6. Place each bulb in its hole and, while holding it in, place with a screwdriver, add a BB-sized drop of epoxy on to the bulb and panel, and hold in place for about 30 seconds.
7. Reconnect all plugs to the circuit breaker panel and, before reinstallation of the panels, turn on the power to the system to check the bulbs for proper operation.

Figure 33 - 9 Circuit Breaker Panel

33-9. TRIM, FLAPS, FUEL, AND ANNUNCIATOR LED's

a. **14 V Basic or Avidyne Avionics**

1. The trim, flap, fuel, and annunciator panel LED's are located in the trim and flap panels, the fuel quantity gauge, and the annunciator panel. These LED's are not field repairable. Contact a factory representative for referral to a qualified technician to replace the LED's.
2. **Electroluminescent Lamps (EL) Lighted Panels** – There are four EL panel overlays in the airplane. These include: the ignition panel, trim panel, flap panel, and ECS panel. The EL portion of each panel is AC powered, and through a so-called cool chemical reaction process, illuminates the letters on the panel. The inverter (AC power source) for the flap panel is located on the back of the flap assembly. The inverter for the ignition, trim, and ECS panel is located on the back of the trim assembly. Therefore, if the trim panel inverter fails, all three panels are affected.
3. **Panel Replacement**– The panel overlays are durable, and the light source has a very high mean failure cycle and should be serviceable through four airframe service cycles, or about 12,000 hours. Should a panel fail to function, contact a factory representative for referral to a qualified technician that can replace the overlay or the inverter, depending on the problem. In some instances, particularly for airplanes with a high number of hours, it might be more cost effective to replace the entire panel assembly.

b. **28 V Basic or Avidyne Avionics and Garmin G1000 system –** All panels are LED surface lit. These LED's are not field repairable. Contact a factory representative for referral to a qualified technician to replace the LED's.

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33-10. POSITION LIGHTS

a. There are a total of four position lights located on the airplane. Each wing tip contains a housing that holds a colored lens forward and a clear lens aft. The position light housing also contains the strobe lights. See Figure 33 - 10. All four bulbs are clear. The forward lens on the left side is red and the forward lens on the right side is green.

CAUTION

If lights have been in use, use protective gloves when removing hot bulbs.

b. **Removal of Position Light Lamp**
   1. Remove the screws securing the lens shield to the wing tip.
   2. Remove the lens from the position light assembly.
   3. Grasp the lamp, depress slightly, and turn counterclockwise to release and remove the lamp from the bayonet mount.

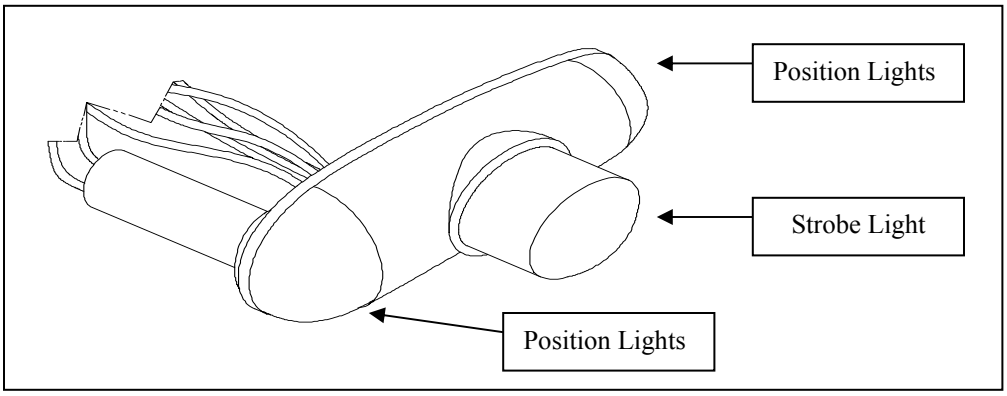


Figure 33 - 10 Position and Strobe Lights

c. **Installation of Position Light Lamp**
   1. Place the lamp in the bayonet socket, depress, and gently turn clockwise until the lamp seats in the socket.
   2. Position the lens and gasket in place.
   3. Secure the lens by installing the lens shield to the wing tip with screws.

d. **Maintenance of Position Light** – General maintenance is limited to lamp replacement and cleaning of ozone buildup inside the position light lenses. The lenses may be cleaned with dish soap and water using a soft cotton cloth.

33-11. STROBE LIGHTS

a. The strobe lights are contained in the left and right wing tip in the same housing as the position lights. See Figure 33 - 10. The strobe lights and lenses are both clear. Switching the strobe rocker switch to the on position activates the strobe lights. This position supplies power to the to the strobe power supply which alternates the power between the left and right strobe light.

b. **Strobe Tube Assembly Removal (Left or Right)**
   1. Remove the screws securing the lens shield to the wing tip.
   2. Remove the lens from the front of the strobe tube assembly.

CAUTION

If the same strobe tube assembly is to be reinstalled after removal, use protective gloves or a cotton wrap to ensure fingertip oil does not come into contact with the strobe tube assembly.

3. Carefully pull the strobe light assembly wiring to retrieve the strobe connector (P17/P18) from inside the wing tip. Disconnect the strobe connector.
4. Remove the strobe tube assembly from the base assembly, and remove from the wing tip.

CAUTION

Use protective gloves or a cotton wrap to ensure fingertip oil does not come into contact with the strobe tube assembly.

c. **Installation of Strobe Tube Assembly**
   1. Position the strobe tube assembly, and connect the strobe connector.
   2. Feed the strobe connector and the strobe light assembly wiring into the wing tip.
   3. Position the spacer, and install the strobe tube assembly to the base assembly.
   4. Position the lens at the strobe tube assembly, and place the ground wire inside of the lens.
   5. Place the lens on the strobe tube assembly.
   6. Secure the lens by installing the lens shield with screws.

d. **Maintenance of Strobe Tube Assembly –** General maintenance is limited to lamp/tube replacement, removal and installation of the strobe light power supply, and cleaning of ozone buildup inside the strobe lenses. The lenses may be cleaned with dish soap and water using a soft cotton cloth.

CAUTION

Use protective gloves or a cotton wrap to ensure fingertip oil does not come into contact with the strobe tube assembly.

e. **Removal of Strobe Power Supply**
   1. Ensure the system master and strobe rocker switches are in the off position.
   2. To access the strobe power supply, remove the center console access panels (basic or Avidyne Options) or remove the tower (Garmin G1000 option). See Chapter 25.
   3. Disconnect the strobe connector and strobe power supply connector. Be sure to label the location of each wire.
   4. Remove the screws securing the strobe power supply to the cabin floor, or to the tower, and disconnect the mode jumper.
   5. Remove the strobe power supply from the airplane.

f. **Installation of Strobe Power Supply**
   1. Position the strobe power supply to the nutplates in the cabin floor, or in the tower.
   2. Secure all required screws.
   3. Connect the ground wire with the screw used to secure the strobe power supply.
   4. Connect the strobe connectors, strobe power supply connector, and mode jumper.

5. Install the left and right center console access panels, or the tower, as applicable.
6. Set the system master and strobe rocker switches to the on position, and check strobes for proper operation.

33-12. LANDING AND TAXI LIGHTS

a. The taxi and landing lights are contained in the leading edge of the left wing. The outboard bulb in the light housing is the taxi light which provides a diffused light in the immediate area of the airplane. The inboard bulb is the landing light, which has a spot presentation with a slight downward focus. See Figure 33 - 11. Xenon taxi lights are only available on 28V aircraft.

CAUTION

The landing and taxi lights become extremely hot in a short period of time. Do not touch the lens or any part of the lights under any circumstances as they can cause serious burns.

CAUTION

Use protective gloves or a cotton wrap to ensure fingertip oil does not come into contact with the landing or taxi light lens.

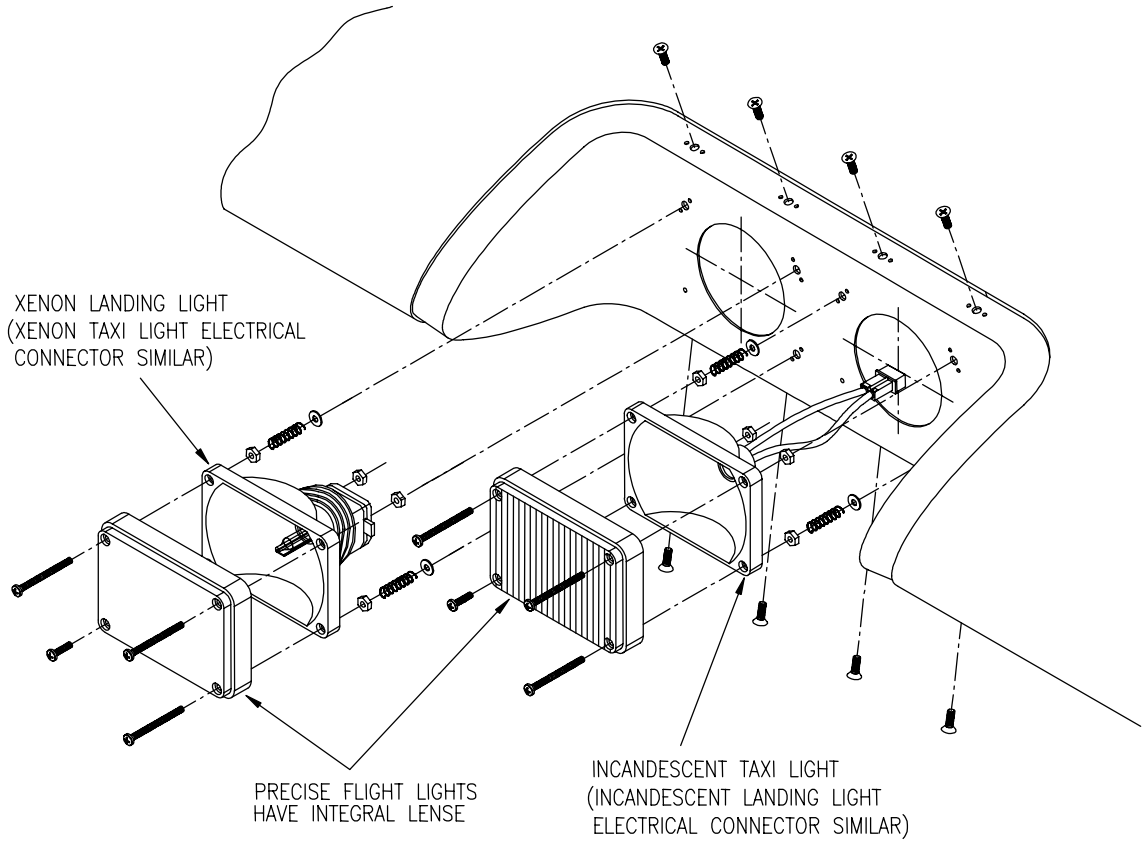
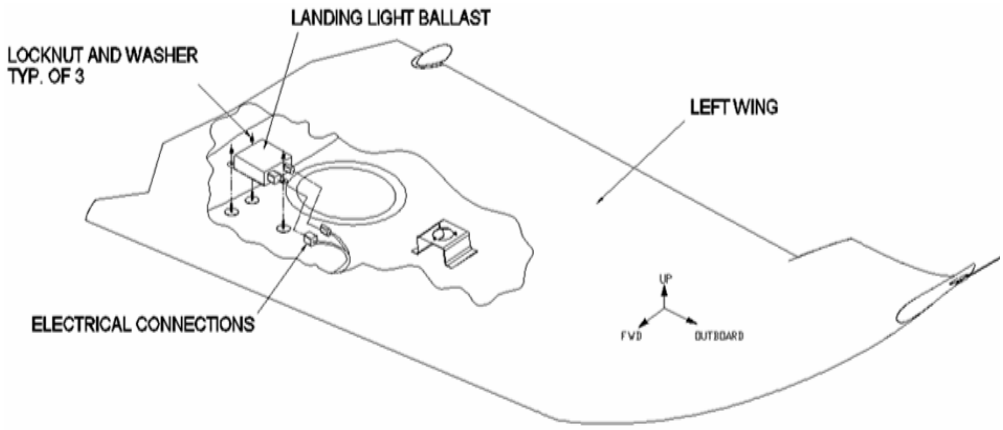


Figure 33 - 11 Landing and Taxi Light Assembly

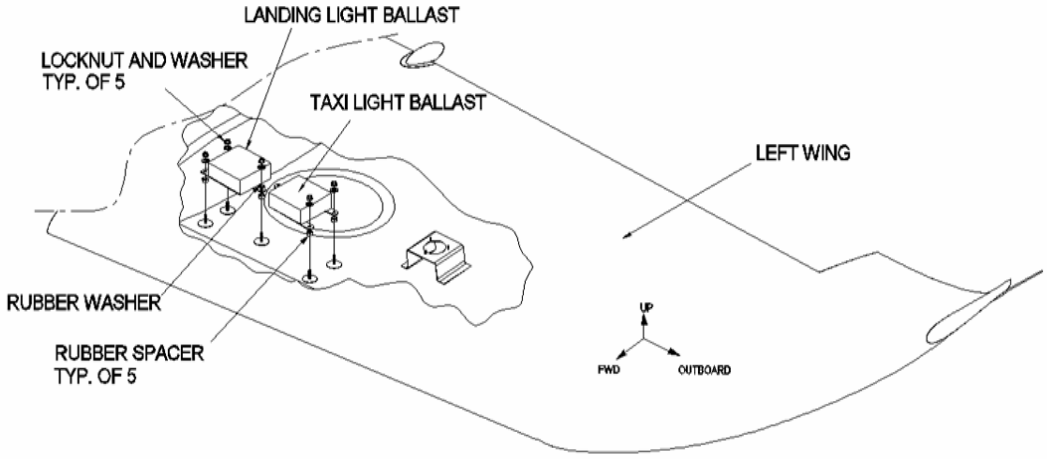
b. **Landing and Taxi Light Removal**
   1. Remove the four screws on top and four screws on bottom of the landing and taxi light shielding.
   2. Remove the mounting screws on each corner of the light.
   3. Pull the light out and disconnect.

c. **Landing and Taxi Light Installation**
   1. Install the light connector.
   2. Place the light in the slot and install the mounting screws.
   3. Reinstall the landing and taxi light shielding with four screws on top and four screws on bottom. Fill the gap between the edge of the joggle and the edge of the shielding with RTV 162 white silicone caulk.

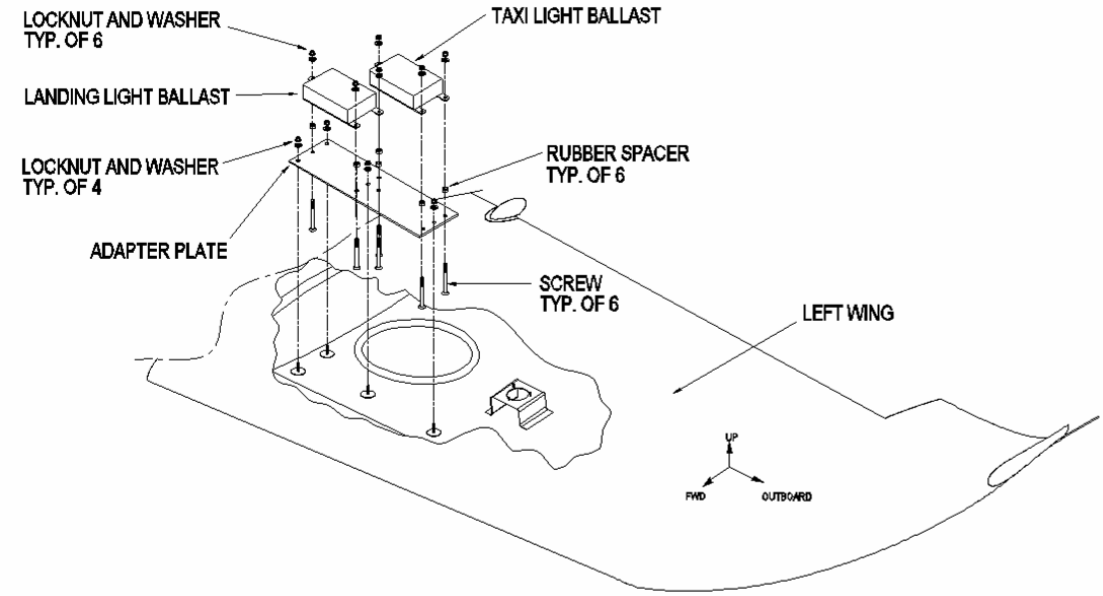
d. **Xenon Landing and Taxi Light Ballast Removal and Installation** (See Figure 33 - 12)
   1. Remove the nine 6-32 flat head screws from the outboard access panel on the left wing and remove the panel.
   2. Disconnect the electrical connections from the ballast.
   3. Remove the locknuts, washers, and rubber spacers (as applicable) holding the ballast to the wing and remove the ballast.
   4. Installation is the reverse of removal. Tighten the nuts hand tight plus 1/4 turn. Apply Permatex #2 or Loctite 222 threadlocker to each screw and install the access panel. Torque the access panel screws 12 to 15 in-lbs.



TYPE A – Xenon Landing Light



TYPE B – Precise Flight Landing and Taxi Light



TYPE C – Precise Flight Landing and Taxi Light Retrofit

Figure 33 - 12 Xenon Landing and Taxi Light Ballast

33-13. TAXI AND LANDING LIGHT ADJUSTMENT

CAUTION

The landing and taxi lights become extremely hot in a short period of time. Do not touch the lens or any part of the lights under any circumstances as they can cause serious burns.

a. The taxi and landing lights are aimed in a manner that maximizes coverage for each respective flight operation. Over time, hard landings and rough field operations, to mention a few reasons, may cause the lights to lose their optimum alignment. Since both lights have a 100% duty cycle (can be used 100% of the time), their luminosity is somewhat reduced. For this reason it is important that the lights be kept in proper alignment. The following paragraphs describe the method for checking the alignment of both the taxi and landing lights and, if necessary, procedures for adjusting each lens.

b. **Reference Card** – To determine how the taxi and landing lights are aimed, a simple reference card must be constructed. The card should be approximately four feet high and six feet wide; corrugated cardboard works quite well and is easy to handle. Draw two horizontal, parallel lines and one vertical line on the cardboard. The horizontal lines are drawn on the 6 ft. length up from the bottom of the card: 25 in. for incandescent lights or 29 in. for xenon lights, and 35 in. The vertical line bisects the 6 ft. length. The horizontal line 35 in. from the bottom is the taxi light reference, and the horizontal line 25 in. or 29 in from the bottom is the landing light reference. Please see Figure 33 - 13 Light Adjustment Reference Card below for a drawing of the reference card.

c. **Landing/Taxi Light Cover Removal** – It is best to remove the outer light cover before performing the test to facilitate a more accurate measurement and allow easy access to the adjustment screws for re-aiming both of the lights. The acrylic plastic cover for the light assembly is secured with eight screws, four on the top and four on the bottom. Remove all eight screws, and pull forward on the cover to remove it. If the light lenses are dirty, they should be cleaned at this time.

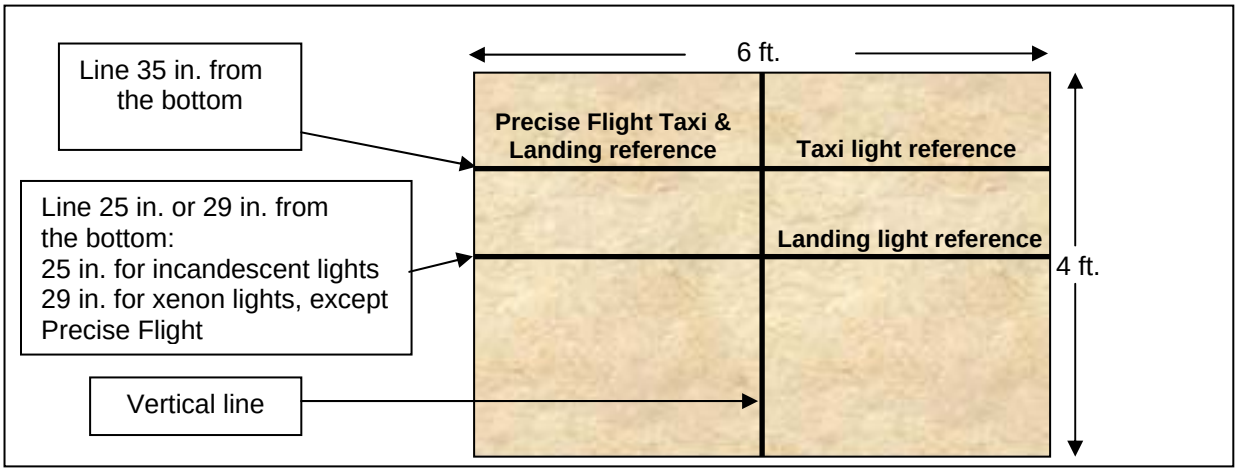


Figure 33 - 13 Light Adjustment Reference Card

d. **Ambient Light** – The test must be performed in an area of reduced ambient light such as a hanger with the doors closed and part of the overhead lighting off. Adequate lighting conditions are also available outside about one hour after sunset.

e. **Placement of Reference Card** – Place the reference card in front of the airplane so that the card is 9 ft. forward of the landing/taxi lights, and the vertical line on the card is 10 ft., 3 in. to the left (when facing forward) of the propeller spinner tip. See Figure 33 - 14.

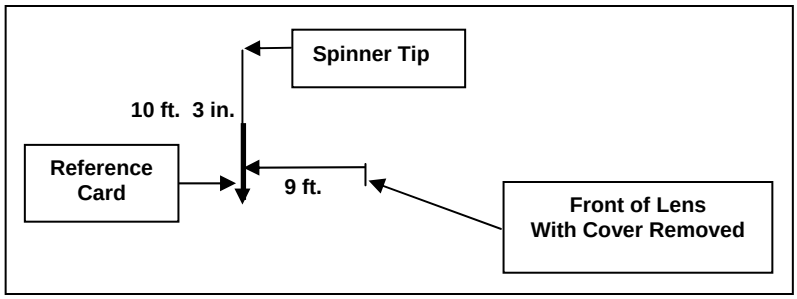


Figure 33 - 14 Placement of Light Adjustment Reference Card

f. **Checking the Alignment** – Each light is operated and checked separately and should produce the displays discussed below. An approximate representation of the light dispersion patterns are shown in Figure 33 - 15.

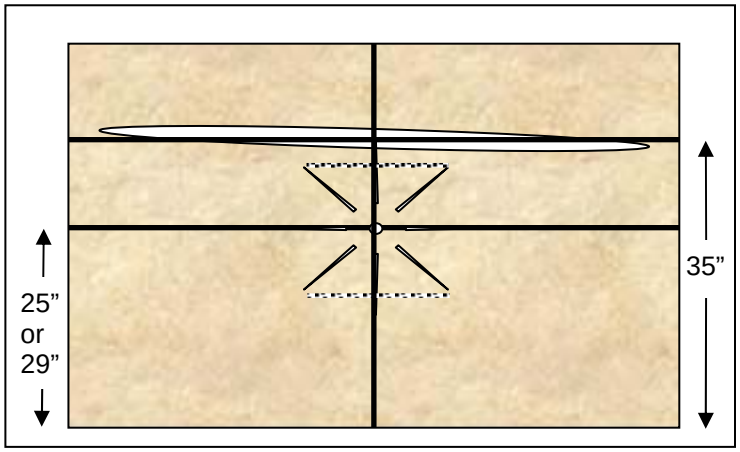


Figure 33 - 15 Light Adjustment

1. Taxi Light, and Precise Flight Landing and Taxi Lights – The brightest concentration of the taxi light should focus laterally on the line 35 in. from the bottom and the pattern should be equally dispersed on the left and right side of the vertical line. Towards the edges of the reference card, the concentration will be somewhat above the line on the left side and below the line on the right side.
2. Landing Light (except Precise Flight) – The center of this light should be focused on the intersection of the vertical line and the line 25 in. for incandescent lights or 29 in. for xenon lights from the bottom.

g. **Aiming or Adjusting the Lights** – If the lights are not properly aimed, there are adjustment screws on the face of each lens that can be turned so that each lens is properly aimed. Turning the screw on the face of the lens clockwise will cause the lens to move. The amount of adjustment to each screw is limited by physical stops, so it works best to adjust both sides of the lens in small increments. For example, a full clockwise turn on a right screw should be followed by a full counterclockwise turn on the left screw.

h. **Lens assembly** – Once the lenses are properly aimed, ensure both lights and the master switches are off. Clean the acrylic plastic housing cover, and check the condition of the rubber gasket on the inside edge of the cover. Replace or re-glue the gasket as required. After the lights have cooled sufficiently, install the cover, and tighten the eight screws. Torque screws 10 to 12 in.-lbs.

CHAPTER 34

NAVIGATION

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34-1. GENERAL

a. **Scope –** This chapter describes the navigation systems that provide navigational information. Flight instruments are operated by ram air, barometric air pressure differences, electric current variations, aircraft electrical power, or references to the earth's magnetic field.

b. **Descriptions**

1. **Description of Pitot-Static System** – The pitot-static system has two components: ram air from the pitot tube and static air from the static air vent. Tubing in the pitot system is blue in color and tubing in the static system is white in color. See Figure 34 - 1, Figure 34 - 2 or Figure 34 - 3.
    a) The pitot tube is located under the right wing, and the static port is located on the right side of the fuselage, halfway between the copilot's cabin door and the horizontal stabilizer.
    b) The pitot system provides ram air pressure. The amount of ram compression depends on air density and the rate of travel through the air. The ram air, in conjunction with static air operates the airspeed indicator. The static system also provides static (uncompressed) air for the altimeter, vertical speed indicator, autopilot altitude sensor, and a blind encoder that is integrated with the airplane's transponder.
    c) The pitot-static system is extremely sensitive to changes in pressure, therefore, it must be kept free of moisture and obstruction. If moisture, leaks, or obstructions exist within the system, the three diaphragm gauges will give inaccurate readings.
    d) The pitot-static system includes drains located under the belly of the fuselage (center of the wing stub) and another located inside the fuselage near the static air vent.

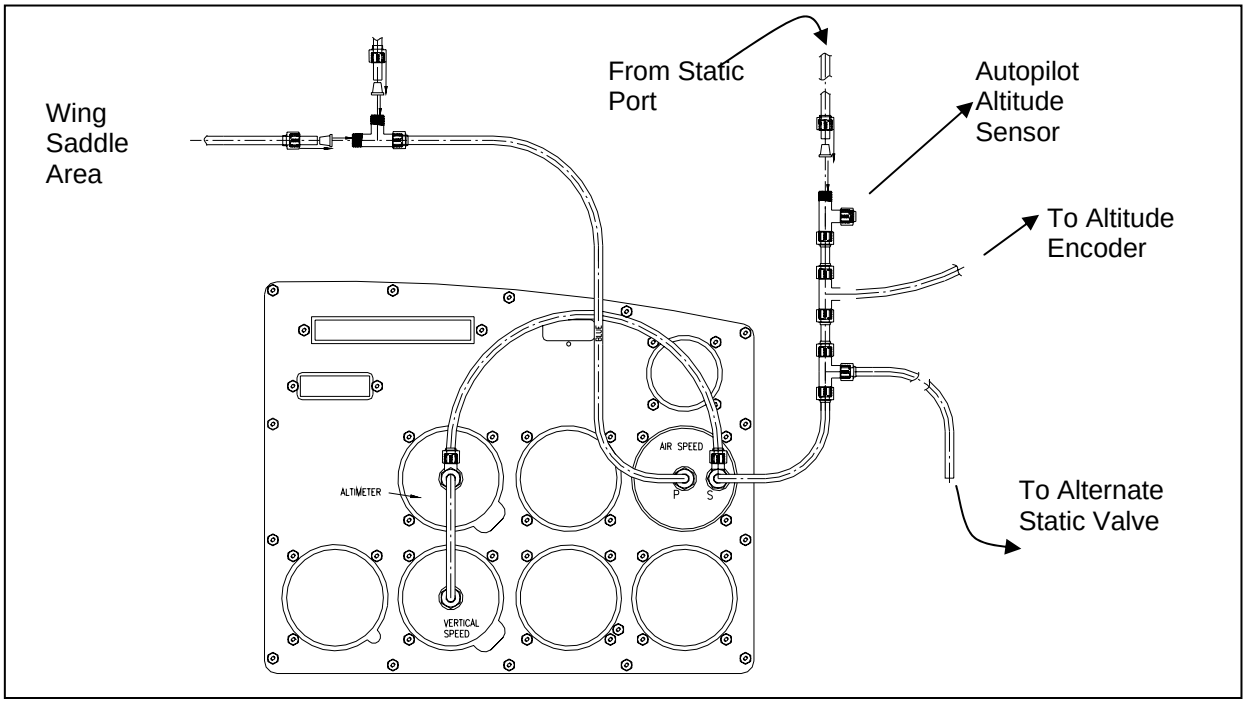

Figure 34 - 1 Pitot-Static System Diagram

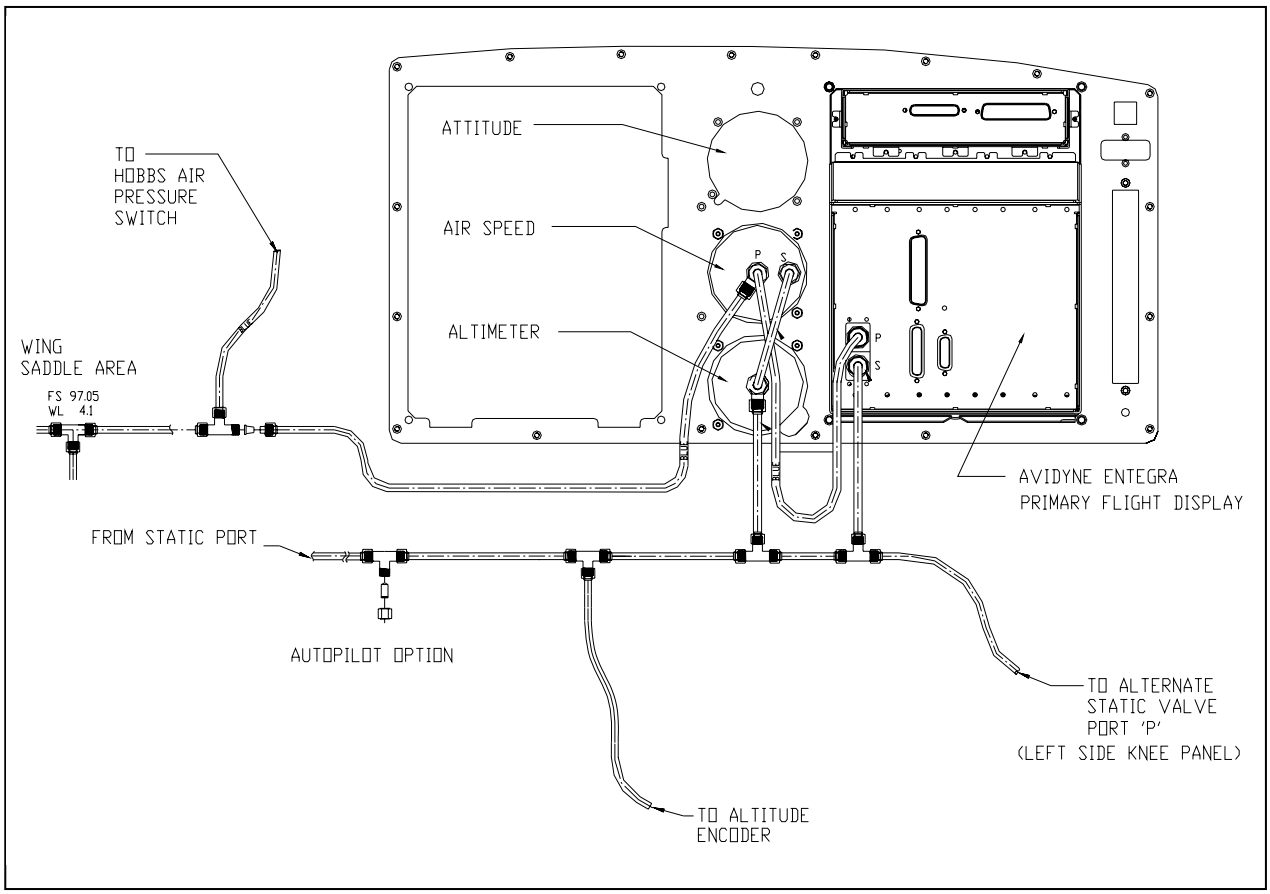

Figure 34 - 2 Pitot-Static System Diagram with Avidyne Entegra Option

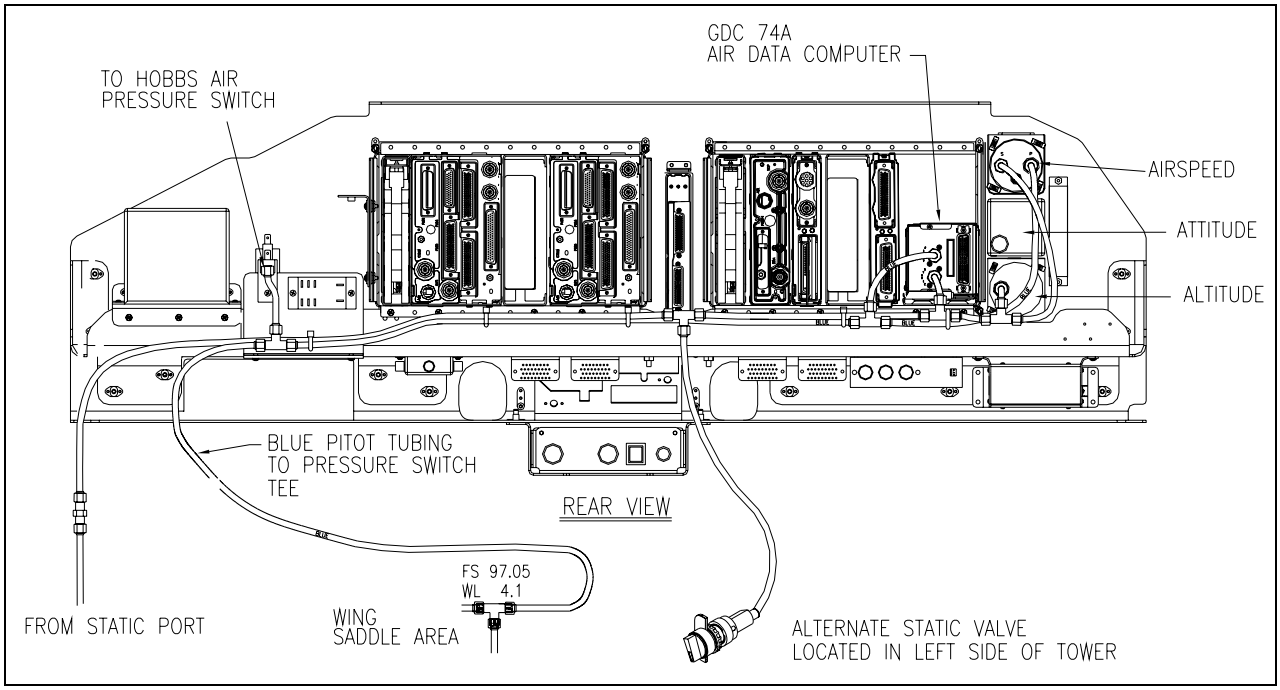

Figure 34 - 3 Pitot-Static System Diagram with Garmin G1000 Option

2. **Description of Stall Warning System** – The stall warning system consists of an audible buzzer mounted to the speaker in the forward section of the overhead console The system is activated by a vane-type switch located on the leading edge of the left wing. Under normal flight conditions, the airflow over the wing keeps the vane in a neutral position. The vane is connected to an electrical switch that is open under normal flight operations. When the airplane approaches its critical angle of attack, the airflow moves the vane up and closes the switch. The switch is set to activate approximately 5 to 10 knots above the aircraft stall speed in all normal flight configurations.

WARNING

The stall warning system is a critical component of the Cessna 350 rudder limiter system and must be accurately set and calibrated by flight test.

3. **Description of systems using gyros** – The Cessna 350 uses three gyro systems: attitude gyros, directional gyro, and turn coordinator (roll rate gyro). The directional gyro, attitude gyros, turn coordinator gyro, and the Allied Signal KI 525 horizontal situation indicator (HSI) are electrically driven.

4. **Description of Marker Beacon/Instrument Landing System –**
   a) **Marker Beacon Receiver** – The marker beacon receiver is located on the far left, top side of the audio panel. There are three lights: blue (outer marker), amber (middle marker), and white (inner/airway marker), which give a visual and aural indication when the respective marker is crossed in flight. The MKR button in the audio panel must be depressed for the audio portion to function. There is also a marker beacon repeater indicator located in the flight instrument panel above the artificial horizon. The repeater indicator must be functional for IFR operations using the marker beacon receiver.
   b) **High and Low Sense Switches** – The high and low sense switches control the sensitivity of the marker beacon receiver. In the High position, the outer marker is received about a mile from the receiver. In the Low position, the airplane must be directly above the marker beacon receiver to receive the aural and visual indication. Many pilots set the marker switch to the higher sensitivity for an advance indication of approaching the outer marker and then set the sensitivity to the lower level when the marker signal is received. Doing this will silence the tone and visual indication until the airplane is closer to the marker, giving the pilot a more precise indication.
   c) Holding the three-position switch in the Test position applies voltage to all three marker lamps to indicate they are functioning. The Test position is spring-loaded so that when the toggle switch is released, it returns to the Low position. The photocell in the faceplate automatically measures ambient light conditions and dims the marker lights, the label backlighting, and the lights in the audio selector buttons.
   d) The aural portion of the marker is turned on and off by selecting and deselecting the MKR audio switch on the audio selector panel. The markers

do not have an automatic audio reset function, and if the audio for the markers is deselected, it stays off until reselected.

e) **Remote Marker Beacon Repeater Indicator** – A remote marker beacon (MB) repeater indicator (Outer, Middle, and Inner Markers) is located above the artificial horizon. The three remote lights are connected to the SL15 or GMA 340 audio panel, which contains the receiver and controls for the remote marker beacons. MB annunciations are also displayed on the SL15 or GMA 340 audio panel and provide a backup source for station passage. The remote marker beacon repeater indicator must be functional for IFR operations.

5. **Description of GPS, GPS/Comm, Nav Systems –**

a) **Garmin AT Apollo ACU** – The Apollo ACU is a self-contained unit that will provide all necessary functions required for switching HSI/CDI data inputs between a conventional Nav (VOR) receiver and an approach certified GPS receiver. The Apollo ACU will also denote specific modes selected by the front panel switches and various outputs from the GPS receiver. The GARMIN AT ACU is only installed when the GARMIN AT Apollo avionics suite and GPS is installed.

b) **Garmin AT Apollo GX50 GPS** – The Apollo GX50 is an approach certified GPS receiver with a high definition moving map display on a sunlight viewable screen. The aircraft's position relative to airports, runways, VOR's, NDB's, intersections, and SUA's can be viewed on the large moving map display. Operation is through the use of Smart Keys. Airspaces and navaids can be added or deleted from the map with a single press of a button. The GX50 is TSO-C129a Class A1 approved for IFR non-precision approach operation.

c) **Garmin GNS 430 GPS System** – The GNS 430 system is a fully integrated, panel-mounted instrument, which contains a VHF communications transceiver a VOR/ILS receiver, and a GPS navigation computer. The aircraft's position relative to airports, runways, VOR's, NDB's, intersections, and SUA's can be viewed on the unit's display or the EX5000 moving map, if installed.

d) **Mid-Continent MD41-146A Annunciator Control Unit** – The Mid-Continent ACU is a self-contained unit that will provide all necessary functions required for switching HSI/CDI data inputs between a conventional Nav (VOR) receiver and an approach certified GPS receiver. The ACU will also denote specific modes selected by the front panel switches and various outputs from the GPS receiver. The Mid-Continent ACU is only installed when the Garmin avionics suite and GPS is installed.

e) **Garmin AT Apollo SL30 Nav/Comm –** The SL30 provides navigation and communications with a 760 channel VHF communications transceiver with 200 channel VOR, Glideslope, and Localizer receivers. Besides traditional Nav/Comm features, the SL30 also incorporates functions such as automatic decoding of the Morse code station identifier for VOR/LOC/ILS, most-used frequency storage in unit memory, and a built-in course deviation indicator. The SL30 has the ability to monitor the standby comm frequency. The SL30 VHF receiver operates from 108 MHz to 117.95 MHz, decoding both the

VHF Omni Range and Localizer navigation signals. The built-in glideslope receiver will automatically tune the corresponding glideslope paired frequencies (328 MHz to 335 MHz) when the localizer is tuned.

f) **Garmin AT Apollo SL15 Audio Panel –** The Apollo SL15 Audio Selector Panel/Intercom System (ASPIS) is frequently referred to as the audio panel. The primary purpose of the panel is to control communication and navigation selections, intercom functions, and the marker beacons. In addition, the unit has provisions for two stereo entertainment inputs. Refer to Chapter 23.

g) **Garmin GMA 340 Audio Panel –** The Garmin GMA 340 audio panel is the primary control of communications and navigation selections, intercom functions, marker beacons, and entertainment units. Refer to Chapter 23 for more details.

h) **Garmin AT Apollo SL70 Transponder –** The Apollo SL70 Mode A/C transponder is designed for both VFR and IFR operations. Features include a squawk code display and selection; transponder mode selection for standby, on, and altitude; built-in altitude hold; and ident button and single-button VFR squawk.

i) **Garmin GTX 327 Transponder –** The Garmin GTX 327 Mode A/C transponder is designed for both VFR and IFR operations. Features include a squawk code display and selection; transponder mode selection for standby, on, and altitude; flight time, countdown, and count up timers; pressure altitude display; and ident button and single-button VFR squawk.

j) **Garmin GTX 330 Transponder –** The Garmin GTX 330 Mode A/C and S transponder is designed for both VFR and IFR operations. Features include a squawk code display and selection; transponder mode selection for standby, on, and altitude; flight time, countdown, and count up timers; pressure altitude display, altitude monitor; OAT/DALT display, ident button and single-button VFR squawk, verbal alerts, and datalink traffic capability.

k) **Bendix-King KCS 55A Compass System –** The KCS 55A system consists of four basic components: the panel mounted KI 525A Horizontal Situation Indicator (HSI), a magnetic slaving transmitter, a directional gyro, and the panel mounted KA 51B Slaving Control and Compensator Unit.

l) **Bendix-King KI 525A Horizontal Situation Indicator (HSI) –** The Bendix/King KI 525A pictorial navigation indicator displays constantly slaved gyro magnetic heading information along with VOR/LOC course deviation and glideslope deviation indications. It is part of the King KCS 55A compass system, which also includes a KA 51B slaving control and compensator unit, a KMT 112 magnetic slaving transmitter, and a KG 102 directional gyro.

m) **Attitude Indicator –** The attitude indicator is an electrically-driven basic attitude display.

6. **Description of Avidyne Entegra Primary Flight Display (PFD) –** The Avidyne Entegra is a panel mounted 10.4-inch diagonal Primary Flight Display (see Figure 34 - 4) that provides the following display capabilities:

Attitude	Airspeed
Altitude	FMS-supplied Flight Plan
Horizontal Situation Indication	Glideslope Indication
Outside Air Temperature	Navigation Source Slection
True Airspeed	Slip Indication
VOR/LOC Indication	Vertical Speed
Wind Direction/Velocity	

The PFD is capable of receiving data from and transmitting data to compatible GPS Nav/Comm systems and autopilots.

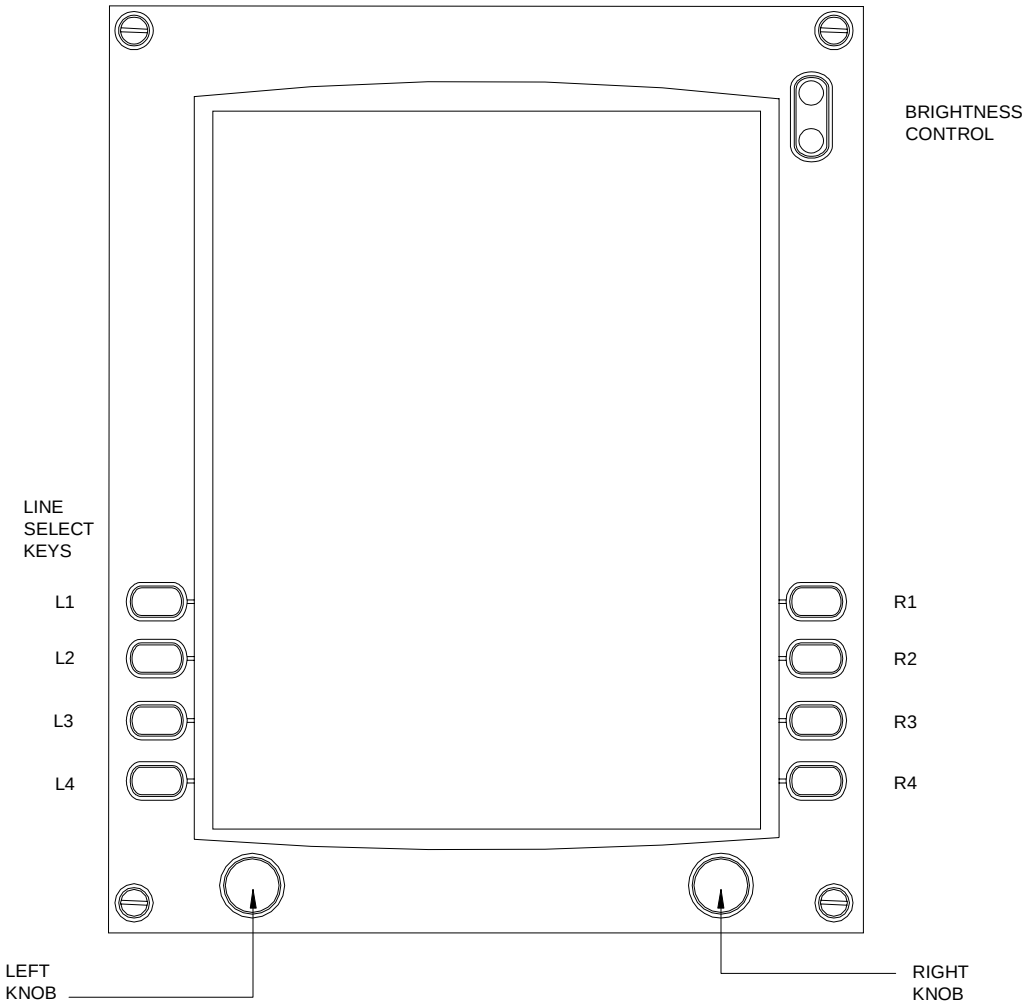

Figure 34 - 4 Avidyne Entegra Primary Flight Display (PFD)

The PFD operation is controlled through 8 bezel keys, 2 rotary knobs, and a rocker switch. Both the right and left knob's function are specific to the particular page that is currently selected, and are appropriately labeled on the LCD above each knob. The rocker switch located in the top right corner of the bezel controls the display brightness, which is adjusted manually by pushing the top portion of the switch to increase intensity

and pushing the bottom portion of the switch to decrease intensity. The functionality of the 8-bezel keys is specific to the currently displayed page, and each key is labeled accordingly on the LCD adjacent to the key.

7. **Description of Avidyne Entegra Multi-Function Display (MFD) –** The Avidyne Entegra MFD is a panel mounted 10.4-inch diagonal situational awareness display as shown in Figure 34 - 5.

The MFD provides the following display capabilities:

1. Moving Map;
2. Traffic (optional);
3. Lightning;
4. FMS-supplied Flight Plan;
5. Obstacles;
6. Datalink Weather (optional);
7. Engine Instruments (optional);
8. Outside Air Temperature

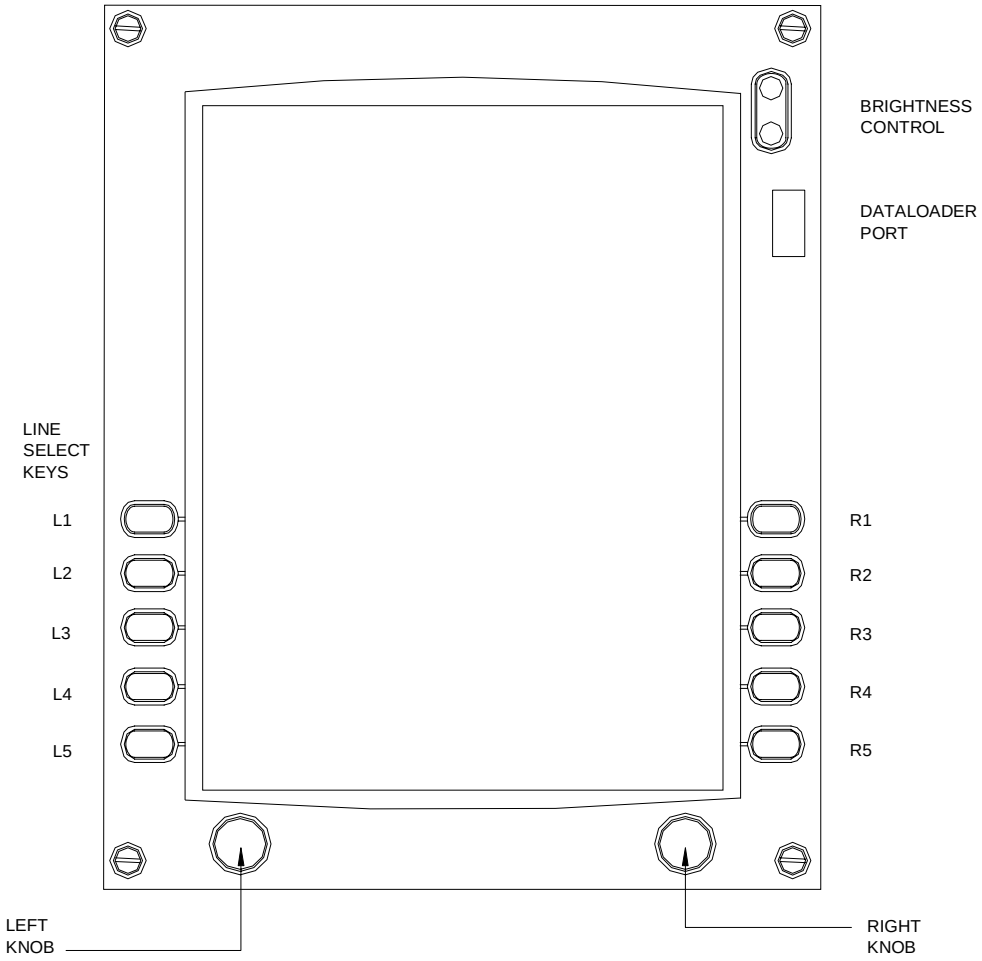

Figure 34 - 5 Avidyne Entegra Multi-Function Display (MFD)

The MFD operation is controlled through 10 bezel keys, 2 rotary knobs, and a rocker switch. The left knob selects the desired page, which is identified on the LCD above the knob. The particular page that is currently selected defines the right knob's function. The knob is appropriately labeled on the LCD above the knob. The rocker switch located in the top right corner of the bezel controls the display brightness, which is adjusted manually by pushing the top portion of the switch to increase intensity and pushing the bottom portion of the switch to decrease intensity. A pop-up window that presents the percentage of full intensity, along with a bar graph representation, indicates the intensity setting of the display. The functionality of the 10 bezel keys is defined by the page that is currently displayed, and are labeled accordingly on the LCD adjacent to the key.

8. **Description of Avidyne Magnetometer/OAT Sensor Assembly –** The Magnetometer/OAT Sensor Assembly provides magnetic heading and outside air temperature information to the PFD for display and computations. See Figure 34 - 6.

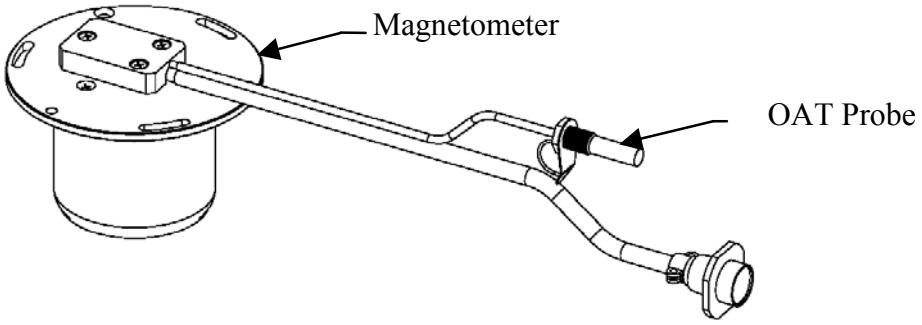

Figure 34 - 6 Avidyne Magnetometer/OAT Sensor Assembly

9. **Description of Avidyne Engine Instrument Sensor Interface Unit (SIU)**
   a) **The SIU interface** – The SIU interfaces with the following engine-mounted sensors: exhaust gas temperature (EGT), cylinder head temperature (CHT), fuel flow, manifold pressure, tachometer, and oil pressure. The SIU interfaces with the following airframe-mounted sensors: battery current and OAT.
   b) **Wiring Harness –** A wiring harness collects signals from six cylinder head temperature (CHT) sensors and six exhaust gas temperature (EGT) sensors and routes them through a firewall feed-through. The feed-through is sealed with a heat resistant sealant, such as RTV736. A wire harness aft of the firewall routes the signals from the firewall bulkhead and from the existing engine indicators, mounted in the instruments panel, to the SIU.
   c) **The SIU data stream –** The SIU data stream transmits on an RS-232 data bus, and includes diagnostic information. Connect the SIU to an Avidyne MFD. The RS-232 serial data are transmitted at 9600-baud, no parity bit, 8 data bits, 1 stop bit, at a rate of 5 Hz. One header message is transmitted after each group of 15 data records.

CAUTION

Before any maintenance, removal, or installation of equipment is performed, make sure both the master and avionics switches are in the off position. Failure to do so could result in damage to equipment.

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34-2. FLIGHT ENVIRONMENT DATA/PITOT-STATIC SYSTEM

The pitot heat may be protected by a thermal device between P16 and the pitot heater limiting the temperature to approximately 175 ºF. When the pitot heat is turned on, this temperature is achieved in 3 to 4 minutes. Once the pitot has reached this temperature and the pitot heat is interrupted, the pitot can take up to 10 minutes to cool before the thermal device resets.

a. **Pitot System – Removing Pitot Assembly**

CAUTION

To avoid injury, turn off the master switches, and pull the circuit breaker for this unit prior to touching it.

1. The pitot assembly is located under the right wing.
2. Remove the access panel next to the pitot assembly.
3. Remove the screws securing the pitot assembly to the wing. See Figure 34 - 7.
4. Disconnect the ram air tube connector at the B-nut.
5. Disconnect the electrical plug.
6. Remove the pitot assembly from the airplane.

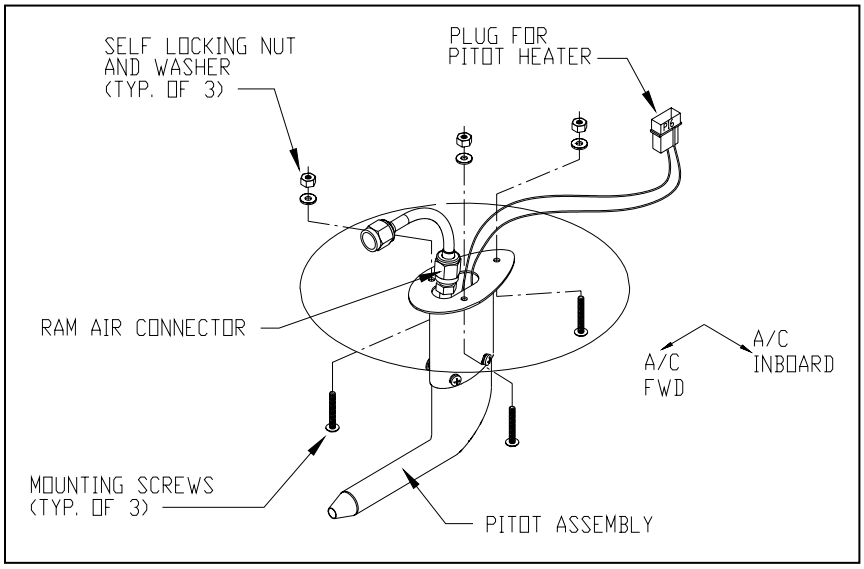

Figure 34 - 7 Pitot Installation

b. **Pitot System – Installing Pitot Assembly**

1. Secure the pitot assembly to the wing structure with screws. Dry torque 10 to 12 in.-lbs. See Figure 34 - 7.
2. Connect the electrical wire plug for the pitot heater.
3. Connect the ram air tube connector (‘B’ nut) to the female connector. Tighten the nut finger-tight, and turn a quarter-turn with a wrench.
4. Install the access panel next to the pitot assembly.

c. **Pitot System – Pitot Heat Testing**

1. Turn the master switch (battery side) on.
2. Turn the pitot heat rocker switch on.
3. Touch the pitot assembly to determine if the temperature is increasing.

CAUTION

To avoid injury, do not grip the pitot assembly if the heat has been turned on for more than one minute.

4. If the assembly does not become warm to the touch within 60 seconds, the heating component is malfunctioning and should be replaced.

d. **Pitot System – Clean Out and Drain**

1. The pitot system moisture collection tube is located at the low point of the pitot system in the wing (underside of the fuselage). See Figure 34 - 8.
2. Remove the fuel module access panel to gain access to the moisture collection tube.

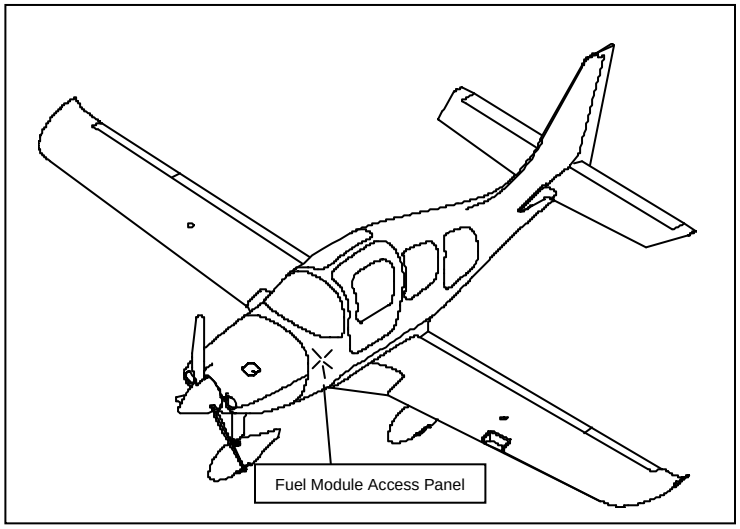

Figure 34 - 8 Fuel Module Access Panel

3. Remove the plastic nut from the union fitting. See Figure 34 - 9.

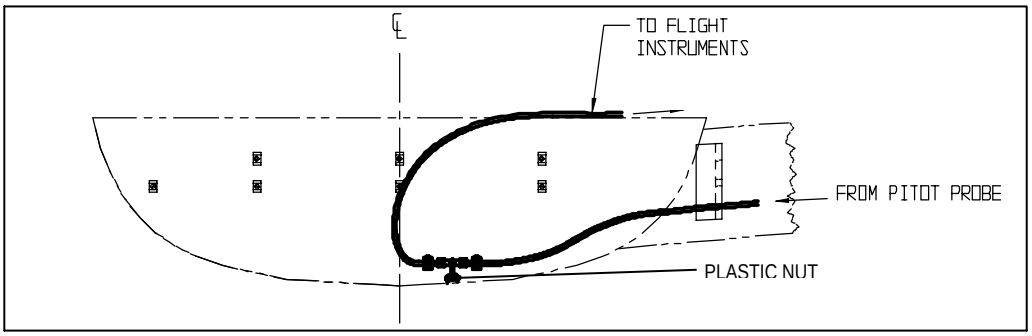

Figure 34 - 9 Pitot Moisture Collection Tube

4. If moisture is present, remove the entire moisture collection tube for cleaning or replacement.
5. Reassemble the moisture collection tube (tighten the nut finger-tight and turn a quarter-turn with a wrench).
6. Reinstall the fuel module access panel.

e. **Static Pressure System Inspection and Leakage Test** – The following procedure outlines inspection and testing of the static pressure system, assuming the altimeter has been tested and inspected in accordance with current Federal Aviation Regulations.

1. Ensure the static system is free from moisture and restrictions.
2. Ensure no alterations or deformations of the airframe surface have been made which would affect the relationship between air pressure in the static pressure system and true ambient air pressure for any flight configuration.
3. Ensure the static pressure alternate source valve, located on the pilot-side knee bolster, is turned to "NORM." See Figure 34 - 10 or Figure 34 - 11.
4. Attach a source of suction to the static system moisture collection tube using sealant tape at the connection points to ensure a tight seal, and tape over the normal static pressure port (located on the right side of the fuselage, halfway from the door to the tail).
5. Slowly apply suction until the altimeter indicates a 1000 ft. increase in altitude.

WARNING

When applying or releasing suction, do not exceed the range of the vertical speed indicator or airspeed indicator.

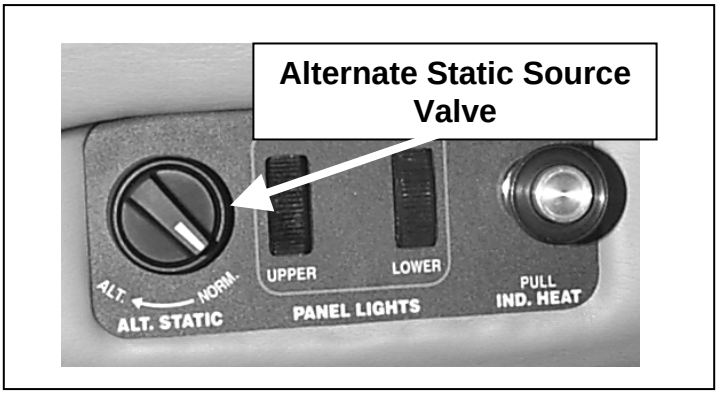

Figure 34 - 10 Alternate Static Source Valve

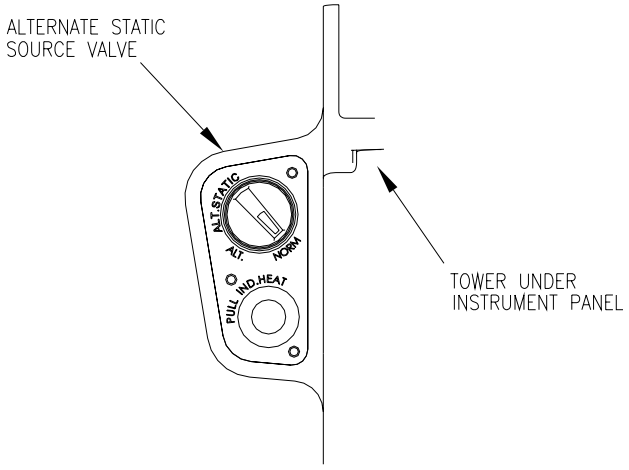

Figure 34 - 11 Alternate Static Source Valve (Garmin G1000 Option)

6. Cut off the suction source to maintain a closed system for one minute. Leakage shall not exceed 100 ft. of altitude loss as indicated on the altimeter.
7. If the leakage rate is within tolerance, slowly release the suction source, and remove the tape from the static port. If the leakage rate exceeds the maximum allowable, tighten all connections, and repeat the leakage test. If the leakage rate still exceeds the maximum allowable, proceed to the next step, otherwise, the test is complete.
8. Disconnect the static pressure lines from the airspeed indicator, vertical speed indicator, and altitude encoder. Use suitable fittings to connect the lines together so the altimeter is the only instrument still connected into the static pressure system.
9. Repeat the leakage test to check whether the static pressure system or the bypassed instruments are the cause of leakage. If instruments are at fault, they must be repaired by an appropriately rated repair station or replaced. If the static pressure system is at fault, use the following procedure to locate leakage.
10. Attach a source of positive pressure to the normal static port.

WARNING

Do not apply positive pressure with the airspeed indicator connected to the static pressure system.

CAUTION

To avoid damage to equipment, always apply pressure or suction slowly when testing.

11. SLOWLY apply positive pressure until the altimeter indicates a 500 ft. decrease in altitude, and maintain this altimeter indication while checking for leaks. Coat the line connections and static source flange with LEAK-TEC or a solution of mild soap and water. Watch for bubbles to locate leaks.
12. Tighten leaking connections, and repair or replace parts found to be defective.
13. Reconnect the airspeed and vertical speed indicators into the static pressure system, and repeat the leakage test steps 3 through 7.

f. **Static Pressure System – Clean Out and Drain**

1. The static system sump/water trap is located behind the interior panel on the right side of the aircraft's baggage area. See Figure 34 - 12.

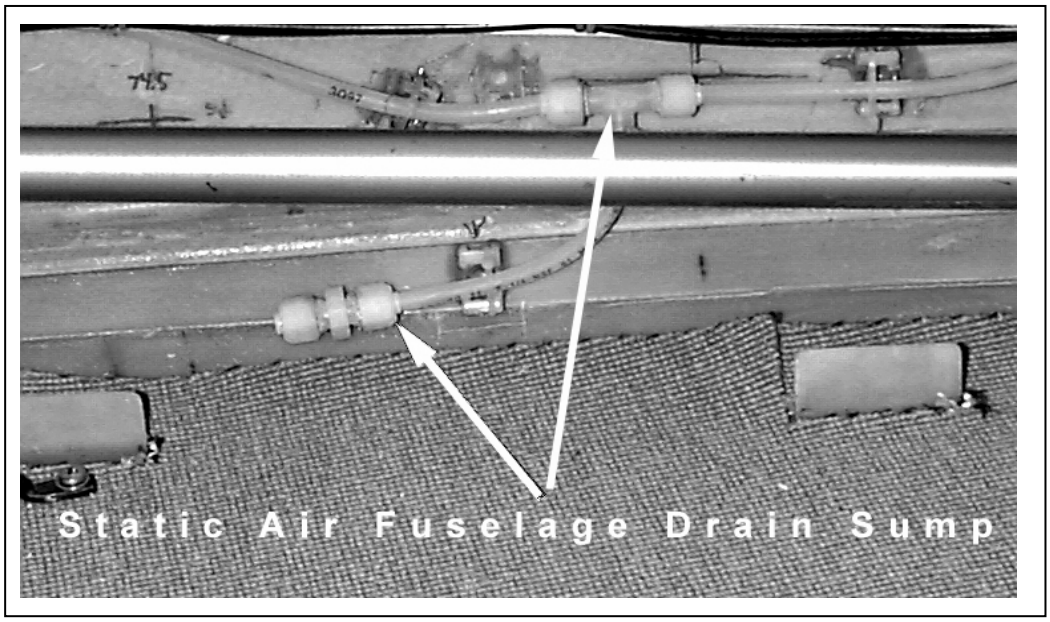

Figure 34 - 12 Baggage Area Static Air Sump Drain

2. Remove the right-side interior curved side panel to gain access to the static sump.
3. Remove the plastic nut from the union fitting.
4. If moisture is present, remove the entire moisture collection tube by unscrewing the B-nut at the T-connection. When the sump assembly is no longer connected to the pitot system, slide it out of its bracket.
5. Clean or replace the drain.

g. **Static Pressure System – Install Moisture Collection Tube**
   1. Before reconnecting the moisture collection tube assembly to the T-connector, be sure that both are clean and free of debris.
   2. Reassemble the moisture collection tube assembly to the static pressure line (tighten the nut finger-tight and turn it a quarter-turn with a wrench).
   3. Assure the collection tube is fastened to the tie-wrap block.
   4. Reinstall the interior panel.
   5. Retest the static system as per paragraph e above.

h. **Static Pressure System – Removal and Install Normal Static Port**
   1. The static port is bonded into the fuselage skin. It must be removed and replaced by qualified composite technicians.
   2. Refer to Chapter 51 for composite repairs.

i. **Alternate Static Source Valve Removal**
   1. Disconnect the left knee bolster from the instrument panel. See Chapter 25 for instructions on the removal of the knee bolster.
   2. Disconnect the static line from the alternate static source valve. See Figure 34 - 13 or Figure 34 - 14.
   3. Insert a small, flat bladed tool into the slot on the backside of the valve to spread the valve from the operator control knob, and remove the valve.

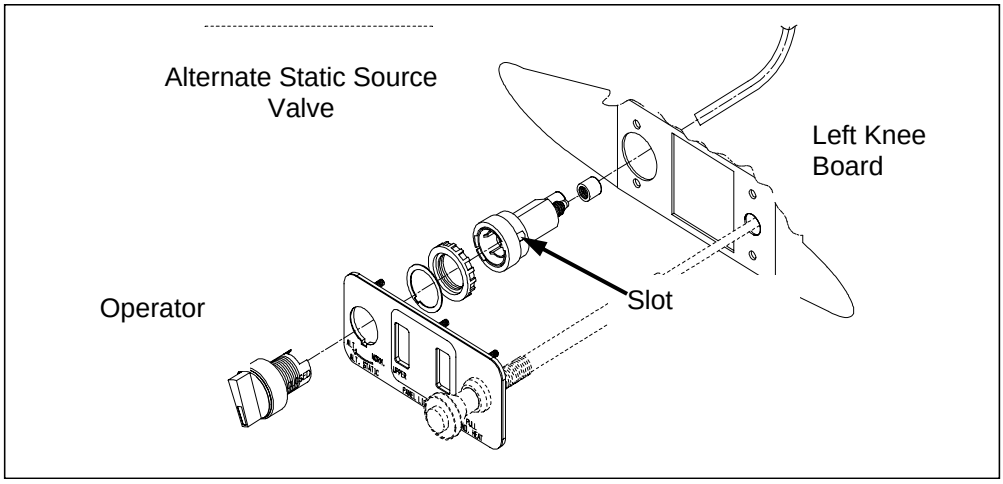

Figure 34 - 13 Alternate Static Source Valve

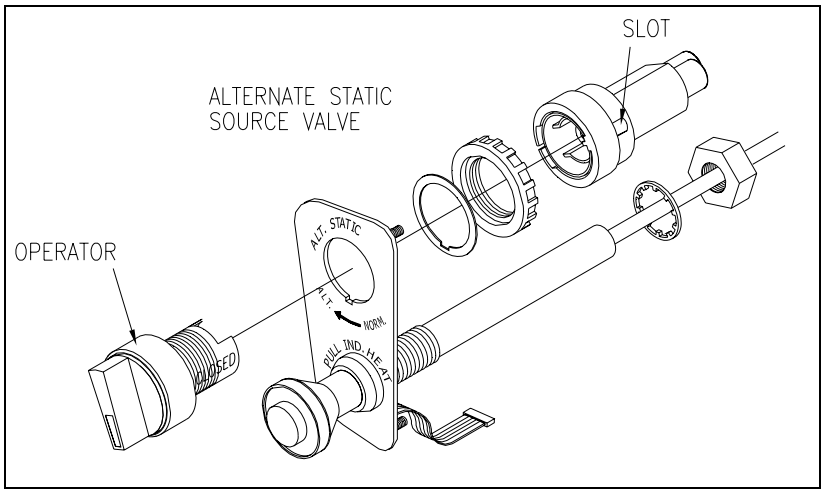

Figure 34 - 14 Alternate Static Source Valve (Garmin G1000 Option)

j. **Alternate Static Source Valve Installation**

1. Snap the static source valve to the operator. See Figure 34 - 13 or Figure 34 - 14.
2. Operate the valve to check for freedom of operation.
3. The back of the static source valve has two connectors labeled “P” and “V”, respectively. Reconnect the static line to the “P” connector of the valve and finger-tighten the nut plus a quarter-turn.
4. Reinstall the left knee bolster.

k. **Alternate Static Source Testing**

1. Note the altimeter elevation setting.
2. Turn the alternate static source valve operator knob to “NORM.”
3. Attach a source of suction to the normal static port.
4. SLOWLY apply suction until the altimeter indicates a 100 ft. increase in altitude.

CAUTION

To avoid damage to equipment, always apply pressure or suction slowly when testing.

5. Turn the alternate static source valve operator knob to "ALT." The altimeter should drop back to its original setting.
6. If the altimeter does not drop back to its original setting, replace the alternate static source valve assembly.

34-3. STATIC AIR INSTRUMENTS – The altimeter, vertical speed indicator, blind altitude encoder (not present if Garmin G1000 installed), and airspeed indicator all utilize static air as a major component of their functionality. Altitude, vertical speed, and airspeed information are displayed on the PFD if the Avidyne or Garmin G1000 options are installed. Removal and installation of redundant altimeter and airspeed indicator instruments is as follows:

a. **With Avidyne Entegra Primary Flight Display (PFD) –**
   1. Altimeter Vertical Speed Indicator, and Airspeed Indicator Removal
      a) Remove the Avidyne Multi-Function Display. See Section 34-23.
      b) Unfasten the instrument from the instrument panel.
      c) Disconnect the wires and the static lines going into the back of the instrument.
   2. Altimeter, Vertical Speed Indicator, and Airspeed Indicator Installation
      a) Slide the instrument into its slot in the instrument panel.
      b) Connect the proper wires and static lines into the back of the instrument.
      c) Fasten the instrument to the instrument panel.
      d) Reinstall the Avidyne Multi-Function Display. See Section 34-23.

b. **Without Avidyne Entegra Primary Flight Display (PFD)**
   1. Altimeter, Vertical Speed Indicator, and Airspeed Indicator Removal
      a) Unfasten the main plate of the instrument panel. See Figure 34 - 15.
      b) Carefully slide the plate out of the instrument panel.
      c) Unfasten the instrument from the plate.
      d) Disconnect the wires and the static lines going into the back of the instrument.
   2. Altimeter, Vertical Speed Indicator, and Airspeed Indicator Installation
      a) Slide the instrument into its slot in the main plate.
      b) Fasten the instrument to the main plate.
      c) Connect the proper wires and static lines into the back of the instrument.
      d) Gently slide the main plate into the instrument panel.
      e) Refasten the main plate to the instrument panel. See Figure 34 - 15.

c. **With Garmin G1000 Option**
   1. Altimeter, Vertical Speed Indicator, and Airspeed Indicator Removal
      a) Reach under the instrument panel and unfasten the instrument.
      b) Disconnect the wires and the static lines going into the back of the instrument.
   2. Altimeter, Vertical Speed Indicator, and Airspeed Indicator Installation
      a) Slide the instrument into its slot in instrument panel.
      b) Fasten the instrument to the panel.
      c) Connect the proper wires and static lines into the back of the instrument.

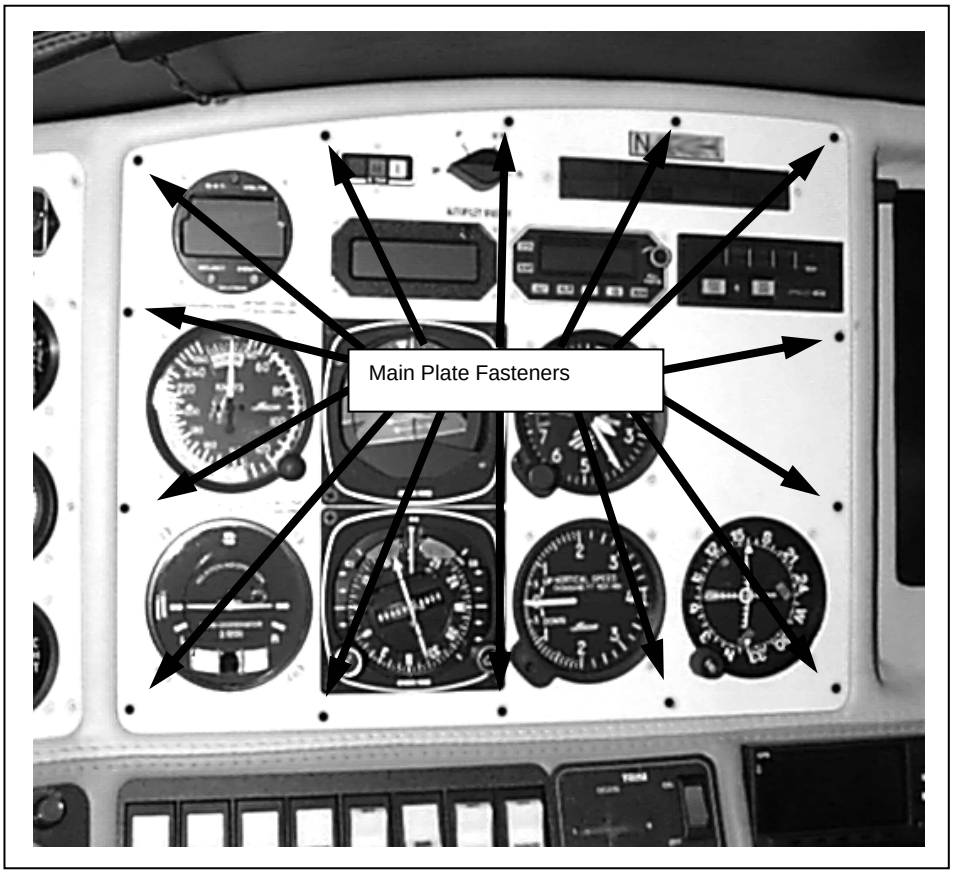

Figure 34 - 15 Instrument Panel Main Plate

d. **Blind Altitude Encoder – Removal**
   1. The blind altitude encoder is attached to a metal plate connected to the firewall on the passenger's side. To access the blind encoder, remove the instrument panel glare shield. The blind encoder is on the bottom left corner of the metal plate. See Figure 34 - 16.
   2. Unhook the wires and static lines connected to the blind encoder.
   3. Remove the knurled bolt above the blind encoder, and remove it from its mounting.

e. **Blind Altitude Encoder Installation**
   1. Place the blind encoder in its mounting.
   2. Finger-tighten the knurled bolt above the blind encoder.
   3. Reconnect the wires and static lines to the blind encoder.

f. **Blind Altitude Encoder Testing**
   1. Set the system master and avionics switch on.
   2. Wait for the blind encoder to warm up (must be warmed internally to within a few degrees of 140°F (60°C)). The instrument is encased within a heated thermostatically controlled manifold that keeps the temperature of the unit within a few degrees of 140°F (60°C). The encoder will not provide altitude information to the transponder until the manifold assembly reaches its operating temperature. The time required for the heat assembly to reach operating temperature is a function of cockpit ambient air temperature. Figure 34 - 17 shows the approximate warm-up times for a range of different ambient temperatures.

3. Set the barometric scale on the altimeter to 29.92 in. of Hg.
4. View the blind encoder altitude readout on the Garmin AT Apollo GPS or the Garmin AT Apollo Transponder. The difference between the blind encoder readout and the altimeter cannot be more than 125 ft.
5. If the altitude encoder readout is not within 125 ft., have a qualified avionics shop check out the system.

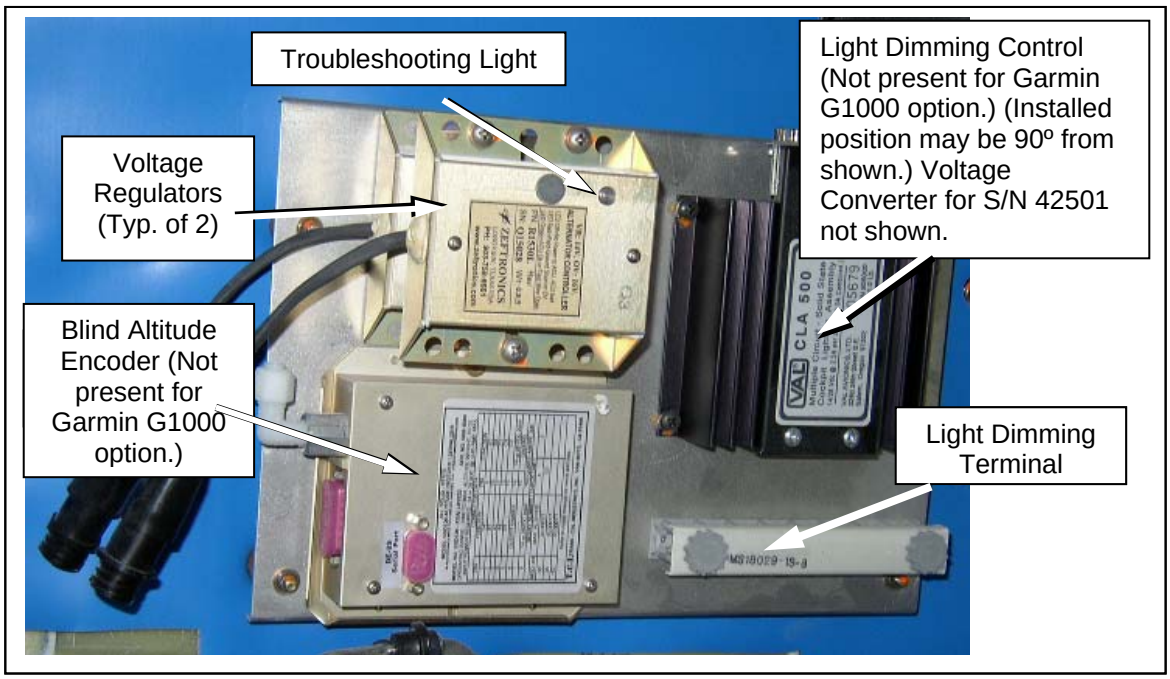

Figure 34 - 16 Accessories Unit Mounting Plate

Cockpit Ambient Temperature	Required Operating Temperature	Warm-Up Time (minutes)
70°F (21°C)	140°F (60°C)	4 to 6
0°F (-18°C)	140°F (60°C)	Up to 10
-55°F (-48°C)	140°F (60°C)	30 to 45

Figure 34 - 17 Blind Altitude Encoder Heating Requirements

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34-4. STALL WARNING SYSTEM

a. **Stall Warning Detector Assembly Location** – The stall warning detector assembly is located on the left wing on the leading edge about halfway between the fuselage and the wing tip. See Figure 34 - 18.

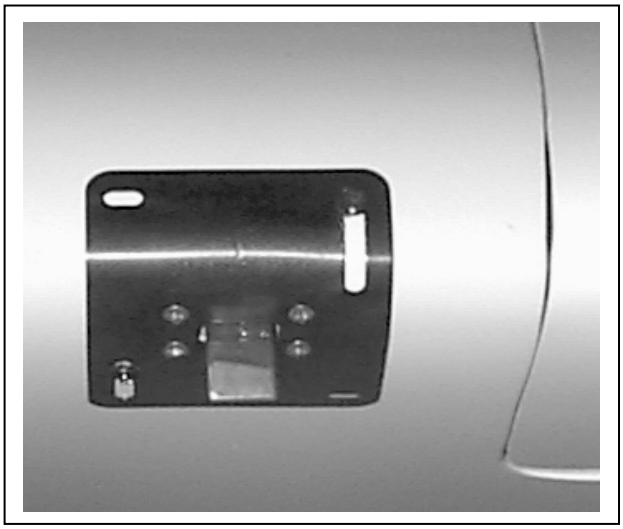

Figure 34 - 18 Stall Warning Detector Assembly

b. **Stall warning Detector Assembly Removal**
   1. With an erasable marker, mark the outline (position) of the faceplate onto the wing.
   2. Remove the two screws connecting the faceplate to the wing.
   3. Disconnect the two wires after labeling the wires with their corresponding terminal positions.

c. **Stall Warning Detector Assembly Installation**
   1. Connect the two wires to the proper terminals.
   2. To be sure the wires are connected correctly, turn the master switch on and push the stall warning vane up. If the stall warning buzzer sounds, the unit is installed correctly.
   3. Align the faceplate with markings.
   4. Fasten screws in the top right and bottom left corner.
   5. Proceed with the test of the stall warning detector assembly.

d. **Stall Warning Detector Assembly Testing**
   1. Test fly the airplane to determine if the stall warning detector actuates at the correct airspeed. Commence the test at 20 KIAS above the stall speed, and bleed off speed at a rate of 1 knot per second until 5-10 KIAS above stall speed. The stall warning buzzer should sound between 5-10 KIAS above the stall speed.

WARNING

This test should only be performed by qualified pilots at a sufficient altitude to safely recover from a stall and in an area where it is safe to perform this maneuver.

2. If the stall warning buzzer sounds at speeds in excess of 10 KIAS above the stall speed, loosen the mounting screws on the stall warning detector assembly, and move the assembly slightly down.

WARNING

Do not bend the metal tab (vane) to adjust the stall warning detector.

3. If the stall warning buzzer does not sound before reaching a speed of 5 KIAS above stall speed, loosen the mounting screws on the stall warning detector assembly, and move the assembly slightly up.

e. **Stall Warning Buzzer Removal**

1. Remove the overhead console. See Chapter 25 for overhead console removal. Also see Figure 34 - 19 for the location of the buzzer in the overhead console.
2. Disconnect the connector attached to the buzzer. See Figure 34 - 21.
3. Remove the attachment screw.

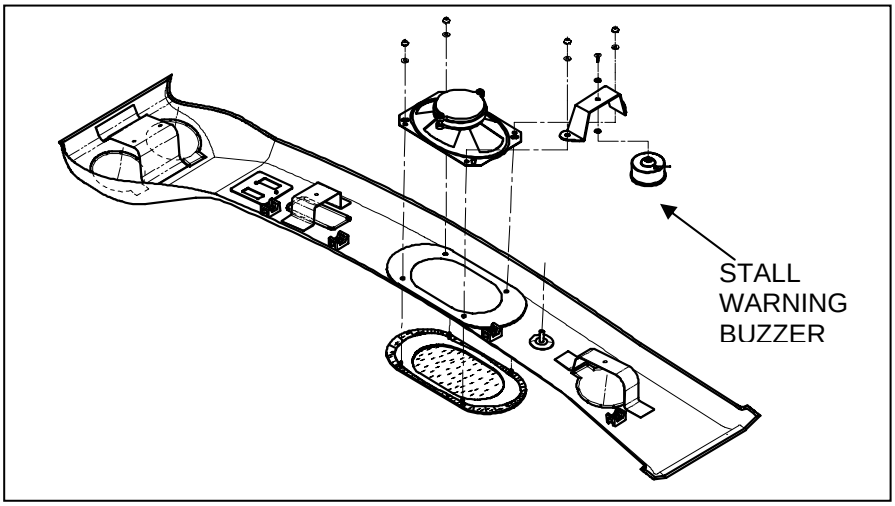

Figure 34 - 19 Stall Warning Buzzer Installation (S/N 42001 to 42003)

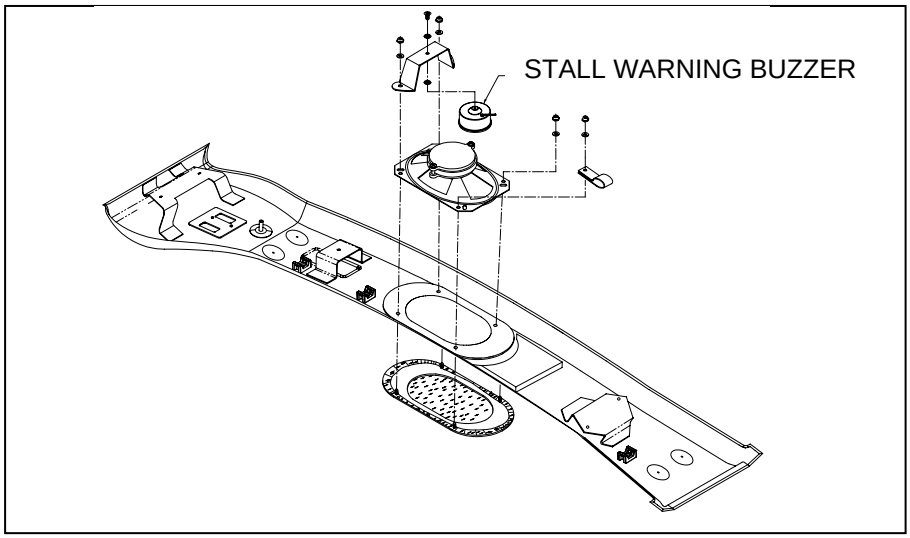

Figure 34 - 20 Stall Warning Buzzer Installation (S/N 42004 and on)

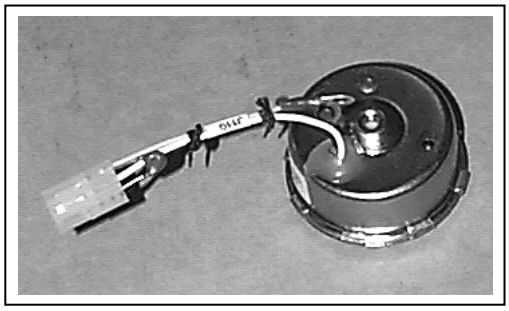

Figure 34 - 21 Stall Warning Horn

f. **Stall Warning Buzzer Installation**

1. Using the screw, fasten the stall warning buzzer to the overhead console. See Figure 34 - 19 or Figure 34 - 20.
2. Connect the wires to the buzzer using the connector shown in Figure 34 - 21.
3. Test the operation of the horn by turning the master switch on, and push the stall warning vane up. If the stall warning buzzer can be heard, it is installed correctly.
4. Reinstall the overhead console as shown in Chapter 25.

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34-5. ATTITUDE AND DIRECTIONAL GYRO INDICATORS

a. **With Avidyne Entegra Primary Flight Display or Garmin G1000 Option –** Attitude and Directional Gyro information are displayed on the PFD. Removal and installation of a backup attitude instrument located to the right of the PFD (basic or Avidyne option) or to the left of the PFD (Garmin G1000 option) is as follows.

1. **Attitude Indicator Removal**
   a) Turn the master power switches off.
   b) For the Avidyne option, remove the Avidyne Multi-Function Display (see Section 34-23). For the Garmin G1000 option, access the indicator from under the instrument panel..
   c) Unfasten the instrument from the instrument panel.
   d) Disconnect the wires going into the back of the instrument.
2. **Attitude Indicator Installation**
   a) Slide the instrument into its slot in the instrument panel.
   b) Connect the proper wires into the back of the instrument.
   c) Fasten the instrument to the instrument panel.
   d) Reinstall the Avidyne Multi-Function Display (See Section 34-23), as applicable.

b. **Without Avidyne Entegra Primary Flight Display (Basic option)**

1. Attitude and Directional Gyro Indicators Removal
   a) Unfasten the main plate of the instrument panel. See Figure 34 - 15.
   b) Carefully slide the plate out of the instrument panel.
   c) Unfasten the instrument from the plate.
   d) Disconnect the wires going into the back of the instrument.
2. Attitude and Directional Gyro Indicators Installation
   a) Slide the instrument into its slot in the main plate.
   b) Fasten the instrument to the main plate.
   c) Connect the proper wires into the back of the instrument.
   d) Gently slide the main plate into the instrument panel.
   e) Refasten the main plate to the instrument panel. See Figure 34 - 15.

c. **Attitude and Directional Gyros Testing**

1. Turn the master switches on. A red flag on the upper right side of the attitude indicator should disappear, indicating power to the unit.
2. Listen to determine if the gyro is operating properly.
3. If the gyro is not heard, remove the indicator, and take it to a qualified technician for replacement or repair.

34-6. TURN COORDINATOR (Basic or Avidyne Option Only)

a. **With Avidyne Entegra Primary Flight Display –** The turn coordinator is located in the avionics shelf under the baggage compartment floor see Figure 34 - 55). No turn coordinator is installed with the Garmin G1000 option.

1. Turn Coordinator Removal
   a) Remove the carpet and access panel from the baggage compartment floor.
   b) Loosen the mounting screws that hold the turn coordinator into its mounting bracket attached to the avionics shelf.
   c) Disconnect all connectors to the unit.
   d) Remove the unit.
2. Turn Coordinator Installation
   a) Perform steps (a)-(d) above, in reverse order.
   b) Level the aircraft (see Chapter 8).
   c) If the ball is not in the center, loosen the mounting screws and adjust the turn coordinator until the ball is centered.

b. **Without Avidyne Entegra Primary Flight Display**

1. Turn Coordinator Removal
   a) Unfasten the main plate of the instrument panel. See Figure 34 - 15.
   b) Carefully slide the plate out of the instrument panel.
   c) Unfasten the turn coordinator from the plate.
   d) Remove the connectors attached to the indicator.
   e) Remove the turn coordinator by pulling it out of the instrument panel.
2. Attitude and Directional Gyro Indicators Installation
   a) Perform steps (a)-(e) above, in reverse order.
   b) Level the aircraft (see Chapter 8).
   c) If the ball is not in the center, loosen the mounting screws and adjust the turn coordinator until the ball is centered.

c. **Turn Coordinator Testing**

1. Turn the master switches on. A red flag should disappear, indicating the power is on. See Figure 34 - 22 .
2. Listen to the indicator to determine if the gyro is properly turning.
3. If the gyro is not heard, remove the indicator, and take it to a qualified technician for replacement or repair.

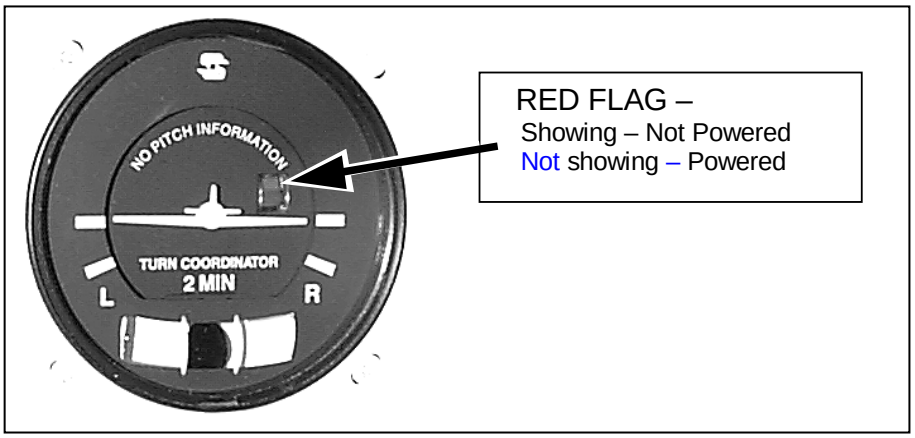

Figure 34 - 22 Turn Coordinator Flag

34-7. MAGNETIC COMPASS

a. **Magnetic Compass Removal and Installation –** The compass is mounted on the windshield, as shown in Figure 34 - 23. The base cannot be removed as it is adhered to the windshield. If the compass needs to be replaced, a new compass unit should be acquired, the pin attaching the compass to the base should be removed, the new compass installed, the pin replaced, and the connector reconnected.

b. **Magnetic Compass Base Re-installation –** Use the following procedure should the compass base become detached from the windshield.
   1. Ensure the windshield is warmer than 60º F.
   2. Disassemble the base from the compass. Remove all adhesive from the compass base and windshield using isopropyl alcohol.
   3. Apply 3M pressure sensitive adhesive P/N VHB 4947F to the base.
   4. Center the base on the windshield as shown in Figure 34 - 23 and adhere. Secure in place with 20 ± 3 pounds force for at least 30 minutes.
   5. Reinstall the compass onto the base.

NOTE

The magnetic compass should be checked for calibration each time avionics or instruments are installed in the instrument panel. A qualified technician must calibrate the magnetic compass under these conditions.

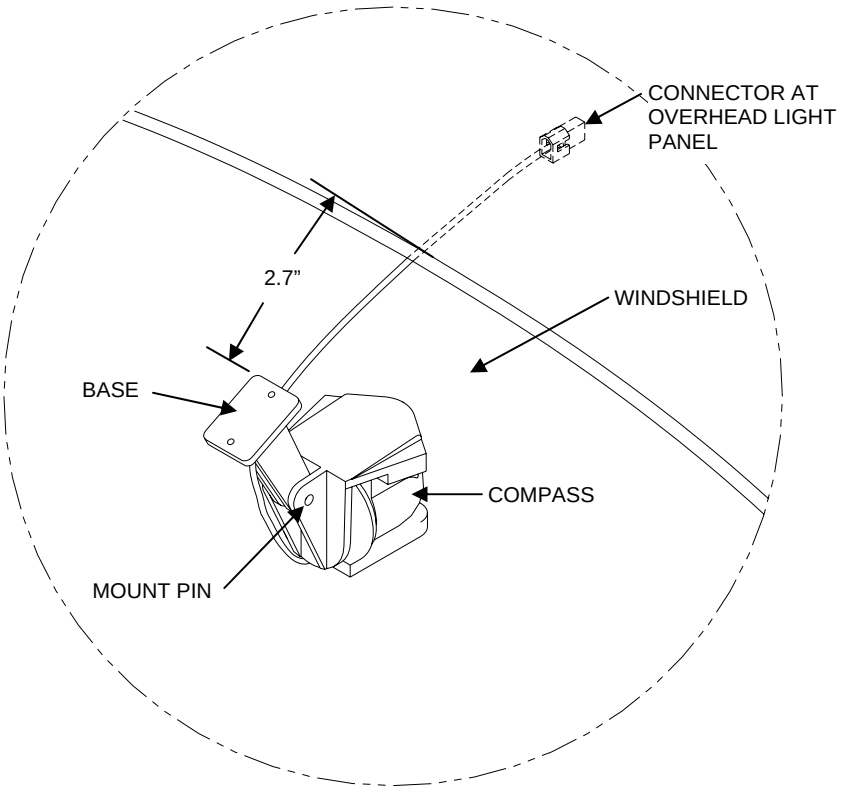

Figure 34 - 23 Magnetic Compass Windshield Attachment

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34-8. MARKER BEACON RECEIVER

a. **Marker Beacon Indicators**

1. There are two marker beacon indicators:
    a) As part of the GARMIN AT Apollo SL15 Audio Amplifier or Garmin GMA 340 Audio Panel, described in Chapter 23.
    b) An S-Tec remote marker beacon light, located in the upper part of the main plate on the instrument panel. See Figure 34 - 24 .

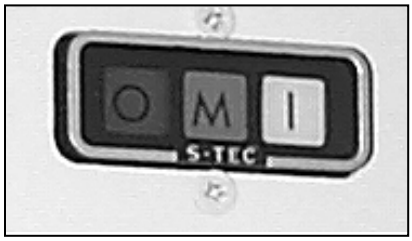

Figure 34 - 24 S-Tec Remote Marker Beacon Indicator

b. **Removal of Remote Marker Beacon**

1. Remove the glare shield and the glare shield electrical connectors.
2. Unfasten the remote marker beacon indicator from the front of the panel. See Figure 34 - 25 or Figure 34 - 53.
3. Disconnect the connector on the back side of the indicator.
4. Remove the marker beacon indicator.

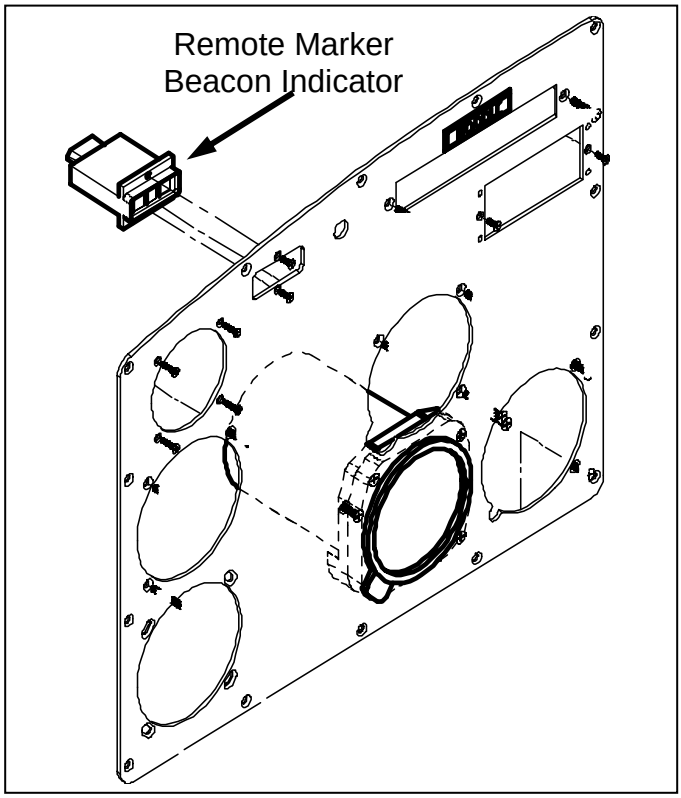

Figure 34 - 25 S-Tec Remote Marker Beacon Indicator

c. **Replacement of Remote Marker Beacon –** Perform steps 1-4 above in reverse order.

d. **Testing of Marker Beacon Indicator Lights**

1. Both the S-Tec remote indicator and the GARMIN AT Apollo SL15 audio amplifier or the Garmin GMA 340 audio panel marker beacon lights are tested by operating the test switch on the SL15 or the GMA 340, respectively.
2. If any light bulb fails to illuminate, take the unit to a qualified radio technician to have the bulbs replaced.
3. If all lights fail to illuminate, check for loose connections. If the lights still fail to illuminate, take the unit to a qualified radio technician for analysis.

e. **Marker Beacon – Removal of Antenna**

1. The Marker Beacon Receiver antenna is located beneath the passenger footwell. See Figure 34 - 26.

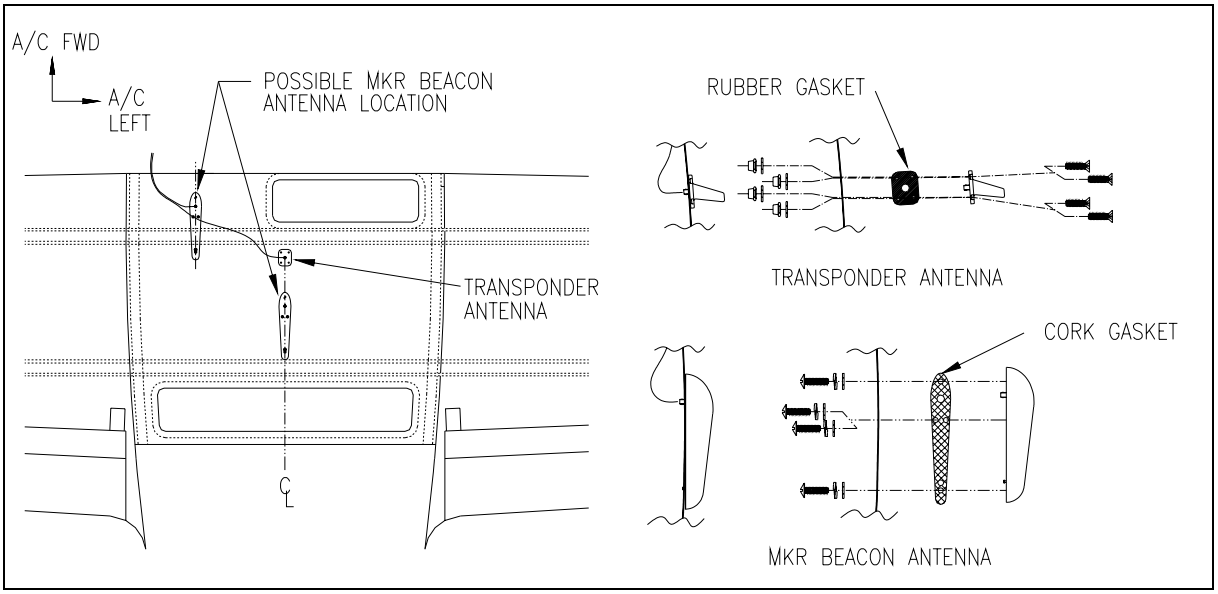

Figure 34 - 26 Marker Beacon Antenna Location

2. Removing the passenger footwell
   a) Remove all four seats.
   b) Lift the carpet in the footwell, and remove the screws connecting the footwell to the flanges.
   c) Remove the footwell from the plane.
3. Unfasten the screws and washers connecting the antenna to the fuselage.
4. Disconnect the antenna cable at the connector.
5. Remove RTV white silicone sealant, if present, and remove the antenna.

f. **Installation of Marker Beacon Antenna**

1. Connect the antenna cable to the antenna.
2. Fasten the antenna to the fuselage with the screws and washers. If there is a gap between the base of the antenna and the wing skin after installation, fill the void around the antenna base with RTV white silicone sealant.
3. Place the footwell into the flange, and fasten it.
4. Install the carpet.
5. Install the four seats.

34-9. MARKER BEACON ANTENNA – TUNED CONDITIONS

a. The marker beacon antenna is tuned at the factory when the airplane is manufactured. In most cases, further antenna tuning is not necessary. When the marker beacon is properly tuned, the conditions shown in Figure 34 - 27 will be observed in flight. At an airplane speed of 90 knots, an altitude of 1000 ft. above the marker beacon transmitter, and the receiver set at the low-sense setting, a station passage time of 13 to 20 seconds should be observed. Station passage is indicated by both a visual and an aural indication. The times listed assume that the airplane is on the centerline of the localizer course or tracking on the extended centerline of the ILS runway. If station passage times outside the parameters shown in Figure 34 - 27 are observed in flight, the antenna on the airplane can be tuned using the steps described in the next paragraph.

Times for Marker Beacon Station Passage		
Ground Speed	**Time for Station Passage**	**Altitude AGL/Sense Setting**
90 Knots	13 to 20 Seconds	1000 feet/Low
110 Knots	11 to 16 Seconds	1000 feet/Low

Figure 34 - 27 Marker Beacon Acceptable Conditions

b. **Marker Beacon Antenna Tuning Adjustments** – Construct a Test Antenna Platform (TAP). The TAP is a 2 ft., square piece of sheet aluminum (any thickness) with a CI-102 Comat marker beacon antenna attached in the exact center of the aluminum sheet. The gasket that comes with the antenna provides a template for drilling the attachment screw holes, the hole for the antenna jack, and the hole for the tuning slug. Be sure to install the gasket when mounting the antenna. See Figure 34 - 28 and Figure 34 - 29 for a picture of the top and bottom of a typical TAP.

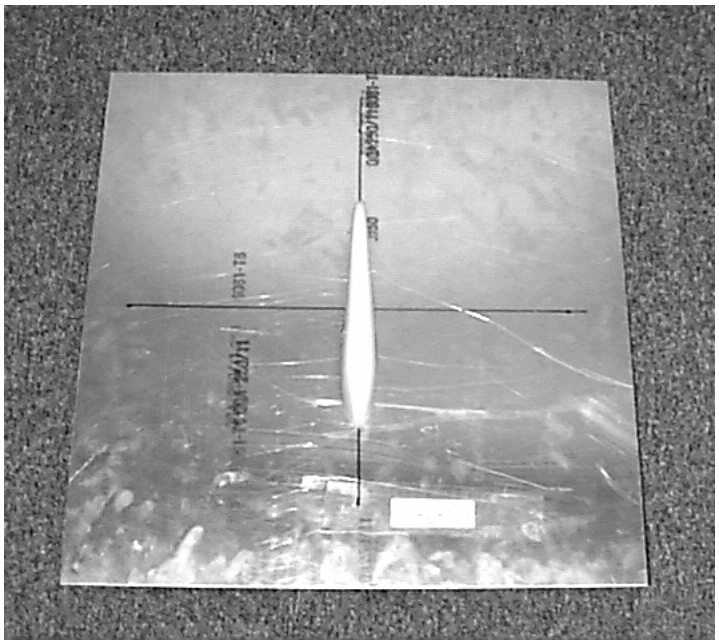

Figure 34 - 28 TAP Top

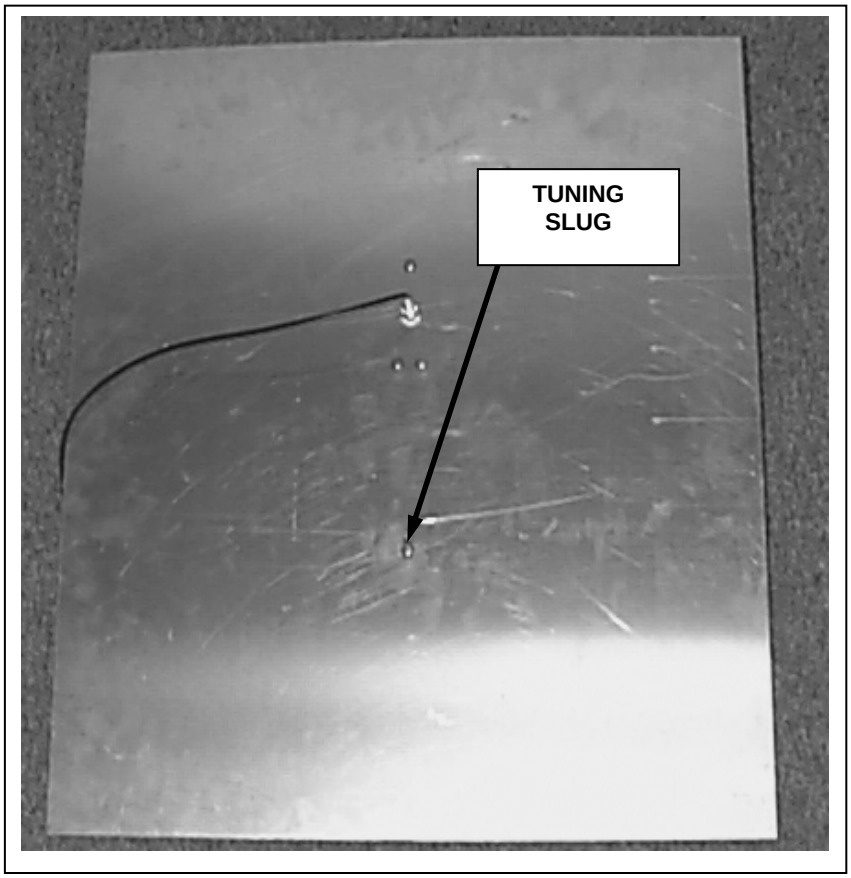

Figure 34 - 29 TAP Bottom

1. When the TAP is assembled, attach a 6 to 10 ft. antenna cable to the antenna jack on the bottom side.
2. Remove the access cover under the wing center section, just aft of the airplane's marker beacon antenna. The cover spans the width of the airplane's underside and provides access to the aileron bellcranks and crossover assembly, flap motor, and the marker beacon tuning slug.
3. In a level area, such as an airplane hangar, place the TAP directly below the airplane's marker beacon antenna. The TAP must be placed on a box or similar type structure so that the distance between the hangar floor and the top of the aluminum sheet is exactly 6 in. See Figure 34 - 30 .
4. When the TAP is directly under the airplane's marker beacon antenna (mirror image) and in a level position, 6 in. above the hangar floor, connect an IFR Model Nav 401 (or equivalent) signal generator to the TAP antenna cable.
5. Turn on the master switch, avionics master switch, and audio panel, and set the marker beacon receiver to the high sense position.
6. Turn on the signal generator, and set the selector to the marker beacon mode.
7. Turn the signal output level of the Nav signal generator down (higher dBm) until the orange MM light on the audio panel just goes out. With an observer in the cockpit, adjust the tuning slug on the airplane's marker beacon receiver antenna with a ¼ in. socket wrench. Rotate the tuning slug until there is a visual and aural indication in the

cockpit. Since the setting of the airplane's antenna is unknown, it will be necessary to experiment with the rotation direction of the tuning slug, i.e., clockwise or counterclockwise. When performing the test, ensure that both the lights on the audio panel and repeater indicator are illuminated.

CAUTION

Do not rotate the tuning slug on the antenna more than 1-1/2 turns in either direction from where the procedure began.

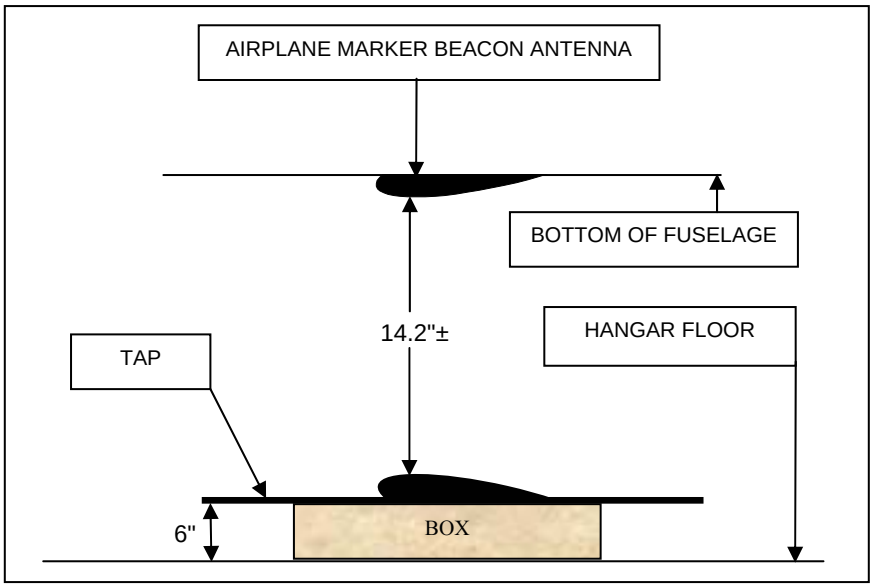

Figure 34 - 30 Location of Test Antenna

8. Repeat step 7 until achieving peak performance from the Marker Beacon receiver and obtaining the lowest possible (highest dBm) signal from the test generator. When performing this adjustment, the minimum triggering level (MTL) of the audio panel should be at least -26 dBm. Further troubleshooting or adjustment of the audio panel should be made if -26 dBm MTL cannot be obtained.
9. When the antenna is properly tuned, turn off the system and avionics master switches, and install the access cover.

c. **Marker Beacon Tuning Adjustments**

NOTE

Never adjust the tuning slug on the TAP antenna. It must remain positioned to the factory default setting.

34-10. INSTRUMENT LANDING SYSTEM – NAV ANTENNA

a. **Nav (ILS) Antenna Removal**

1. The Nav Antenna is located in the vertical stabilizer. See Figure 34 - 31 .
2. Remove the access panel on the vertical stabilizer by removing the screws.
3. Loosen the set screw holding the left element in place.

4. Grasp the antenna rod and twist counterclockwise to unscrew from the antenna body.
5. Repeat step 4 for the antenna on the opposite side of the tail.
6. Remove the screws mounting the Nav antenna body from the bulkhead.
7. Disconnect the antenna cable from the antenna body. On aircraft equipped with the Garmin G1000 system, there is a filter installed between the antenna cable and the antenna.

b. **Nav (ILS) Antenna Installation**

1. Connect the antenna cable to the antenna body. On aircraft equipped with the Garmin G1000 system, connect the antenna cable and filter to the antenna body and secure using heat-shrink wrap; ensuring heat-shrink wrap covers the receptacle, filter, and minimum of 3" of the antenna cable.
2. Fasten the Nav antenna body to the bulkhead.
3. Insert the antenna rods through the holes, and screw clockwise into the antenna body until snug.
4. Tighten the set screw holding the left element in place. (The right element has Loctite on the two threads closest to the antenna rod prior to installation.)
5. Install the access panel to the vertical stabilizer with the screws.

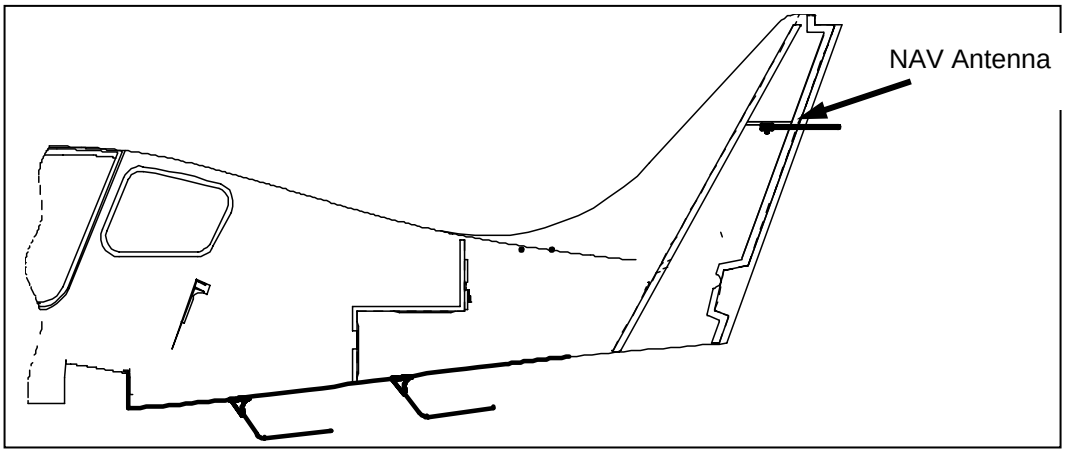

Figure 34 - 31 Nav Antenna Location

34-11. GARMIN AT APOLLO GX50 GPS

a. **GX50 GPS Removal**
   1. Insert an Allen wrench into the hole in the faceplate and turn counterclockwise.
   2. Slide out the GX50 unit out of the rack, and remove the connectors.

b. **GX50 GPS Installation**
   1. Fasten the connector to the GX50 GPS.
   2. Slide the GX50 unit into its slot.
   3. Insert an Allen wrench into the hole on the cover plate, and turn clockwise until the unit is firmly seated in the rack.

c. **GX50 GPS Antenna**
   1. The GX50 GPS antenna is located above the ELT in the space aft of the baggage compartment. See Figure 34 - 32.
   2. To service the antenna, remove the aft baggage bulkhead upholstery plug and the cover plate.

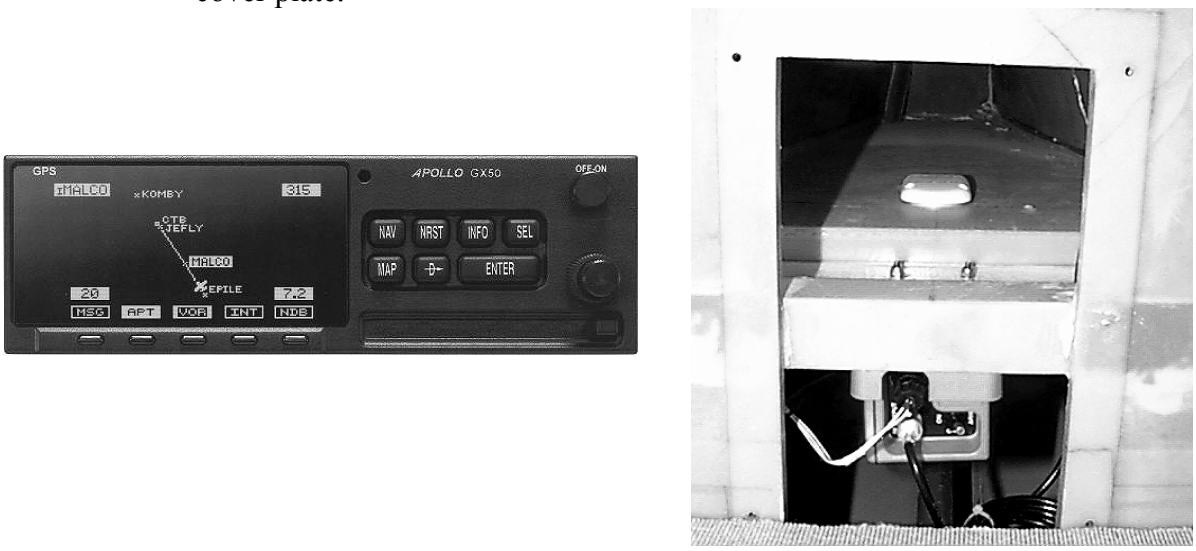

Figure 34 - 32 Apollo GPS GX50 System

34-12. GARMIN GNS 430 GPS System

a. **GNS 430 GPS Removal**
   1. Insert a 3/32 in. Allen wrench into the hole at the bottom of the faceplate, and turn counterclockwise.
   2. Slide out the GNS 430 unit out of the rack, and remove the connectors.

b. **GNS 430 GPS Installation**
   1. Fasten the connector to the GNS 430 unit.
   2. Slide the GNS 430 unit into its slot.
   3. Insert a 3/32 in. Allen wrench into the hole at the bottom of the faceplate, and turn clockwise until the unit is firmly seated in the rack.

c. **GNS 430 Antenna –** The GNS 430 antennas are located above the ELT in the space aft of the baggage compartment as shown in Figure 34 - 32. If the Garmin avionics package is installed, there will be two antennas, one for each unit.
   1. Antenna Removal
      a) Remove the aft baggage bulkhead upholstery plug and the cover plate.

b) Remove the four nuts on each antenna.
c) Lift the antenna and antenna gasket from the antenna box.

2. Antenna Installation
a) Place the gasket on the antenna box, and align the holes.
b) Place the antenna over the gasket with the studs through the holes in the antenna box.
c) Install the four nuts, and torque 20 to 25 in.-lbs.
d) Replace cover plate and aft baggage bulkhead upholstery plug.

34-13. GARMIN AT APOLLO ANNUNCIATOR CONTROL UNIT (ACU) – The following procedures for removal and installation of the GARMIN AT Apollo ACU apply only if the Avidyne Primary Flight Display is not installed in the aircraft.

a. **ACU Remote Removal –** The ACU is located in the instrument panel on the main instrument plate. See Figure 34 - 33 .
1. Remove the glare shield per instructions in Chapter 25, and remove any fittings installed into the back of the ACU.
2. Unfasten the mounting screws on the face of the ACU.
3. Slide the unit out of the instrument panel.

b. **ACU Remote Installation**
1. Install any fittings into the back of the ACU remote.
2. Slide the unit into its slot in the instrument panel, and screw in the mounting screws.
3. Install the glare shield.

c. **ACU Remote Testing**
1. Press 'TEST" to ensure the annunciator lights are all operative. Refer to the GARMIN AT manual for servicing to the unit.
2. If the annunciator lights are operative, but do not operate when the GPS receiver shows it should, check wiring (refer to wiring manual) for continuity.
3. If no apparent wiring or bulb outage is discovered, refer to the GARMIN AT GPS manual for instructions. Only qualified technicians should service this unit.

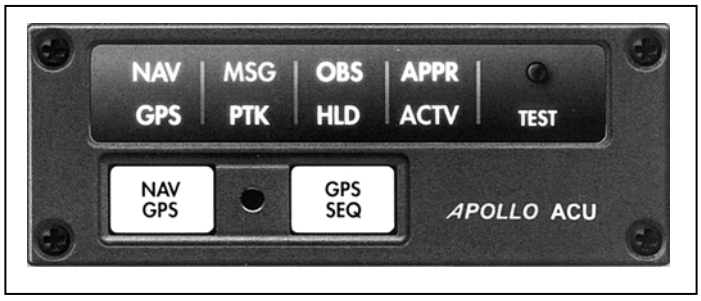

Figure 34 - 33 Apollo ACU

34-14. MID-CONTINENT MD41-1464A ANNUNCIATOR CONTROL UNIT (ACU) – The following procedures for removal and installation of the Mid-Continent MD41-1464A ACU apply only if the Avidyne Primary Flight Display is not installed in the aircraft.

a. **ACU Removal –** The ACU is located in the instrument panel on the main instrument plate. See Figure 34 – 34 for a picture of the ACU.
1. Remove the glare shield per instructions in Chapter 25.
2. Remove the connector at the back of the ACU unit.

3. Remove the two mounting screw on the face of the ACU.
4. Slide the unit out of the instrument panel.

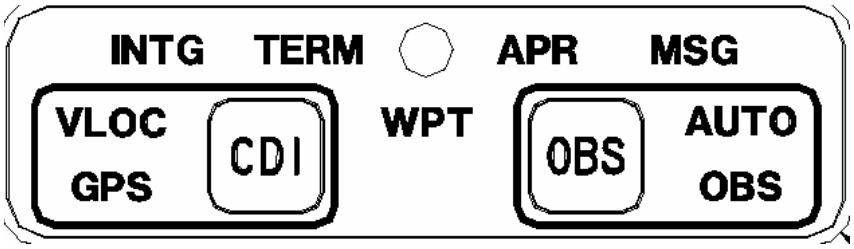

Figure 34 – 34 MD41-1464A ACU

b. **ACU Installation**
   1. Slide the unit into the instrument panel.
   2. Install the two screws securing the ACU to the instrument panel, and tighten snug.
   3. Attach the connector at the back of the ACU unit.
   4. Install the glare shield per instructions in Chapter 25.

c. **ACU Testing**
   1. Press the test switch on the trim panel to test the annunciators on the ACU.
   2. If the annunciator lights are operative, but do not operate when the GPS receiver shows it should, check wiring (refer to wiring manual) for continuity.
   3. If no apparent wiring or bulb outage is discovered, refer to the Garmin GNS 430 system manual for instructions. Only qualified technicians should service this unit.

34-15. GARMIN AT APOLLO SL30 NAV/COMM

a. **SL30 Nav/Comm Removal**
   1. The SL30 is located in the radio rack and can be removed by inserting an Allen wrench into the hole in the faceplate and turning counterclockwise. See Figure 34 - 35 .
   2. Slide the SL30 out of the rack.
   3. Remove the unit.

b. **SL30 Nav/Comm Installation**
   1. Slide the SL30 into its slot in the radio rack
   2. Lock it in place by inserting an Allen wrench into the hole on the cover plate and turning clockwise.

c. **SL30 Nav/Comm Testing –** Other than for checking for loose connections, all testing and service must be accomplished by a qualified radio technician.

d. **SL30 Nav 1 Initial Set-up Procedures**
   Nav 1 interacts with the HSI located in the flight panel, bottom row under the Flight Director. This setup procedure should be done while seated in the left pilot seat with the ACU NAV/GPS button in the NAV position.
   1. Turn the aircraft and avionics power switches ON and the SL30 unit OFF.
   2. On the SL30 unit, press and hold the ◄—► and SYS buttons at the same time and turn the power knob to ON.
   3. Let go of buttons after 3 to 4 seconds.

4. Turn the large knob one click clockwise, and follow step by step directions as given on the unit's screen for calibrating the resolver.
5. When done with the setup, turn the unit off and back on again.
6. Press the NAV button, and adjust the frequency to 110.00.
7. Press the OBS button then rotate the left hand knob with the green/yellow arrow located on the HSI instrument. The screen display on the unit should follow the rotation of the arrows on the HSI. If not, repeat the entire process.

e. **SL30 Nav 2 Initial Set-up Procedures**
   1. Follow steps 1 through 6 for the SL30 Nav 1 Initial Set-up Procedures.
   2. Press the OBS button, then rotate the OBS knob on the NAV indicator instrument. The screen display on the unit should follow the rotation of the arrows on the NAV indicator instrument. If not, repeat the entire process.

Figure 34 - 35 GARMIN AT Apollo SL30 Nav/Comm

34-16. NAVIGATION INDICATOR – The following procedures for removal and installation of the navigation indicator apply only if the Avidyne Primary Flight Display is not installed in the aircraft.

a. Either a Mid-Continent MD 200-306 or a Garmin GI 106A Navigation Indicator will be installed. Removal and installation of both instruments are the same.

b. **Removal of the Navigation Indicator** – The navigation indicator is located in the main plate of the instrument panel. See Figure 34 - 36 .
   1. Turn off the master switch.
   2. Disconnect the main plate from the instrument panel.
   3. Disconnect the connector on the back of the navigation indicator.
   4. Remove the screws connecting the navigation indicator to the instrument panel.

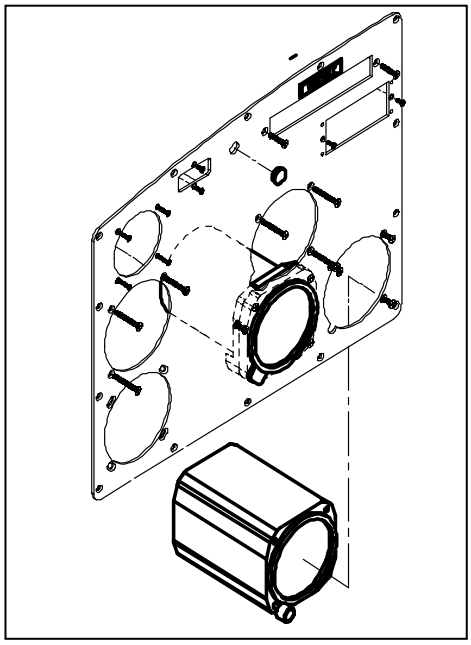

Figure 34 - 36 Navigation Indicator Location

c. **Navigation Indicator Installation**
   1. Place the navigation indicator in its hole in the instrument panel.
   2. Connect the electrical connectors into the back of the navigation indicator.
   3. Refasten the screws connecting the navigation indicator to the main plate of the instrument panel.
   4. Install the main plate of the instrument panel.

34-17. TRANSPONDER

a. Either an Apollo SL70, Garmin GTX 327, or Garmin GTX 330 Transponder will be installed. Removal and installation of the instruments are the same.

b. **Transponder Removal**

The transponder is located in the radio rack.

1. Turn off the master switches.
2. Remove the transponder by inserting an Allen wrench into the hole in the faceplate and turning counterclockwise. See Figure 34 - 37 or Figure 34 - 38.
3. Slide the transponder out of the rack.
4. Disconnect the electrical and antenna connections.

Figure 34 - 37 GARMIN AT Apollo SL70 Transponder

Figure 34 - 38 Garmin GTX 327 and GTX 330 Transponder

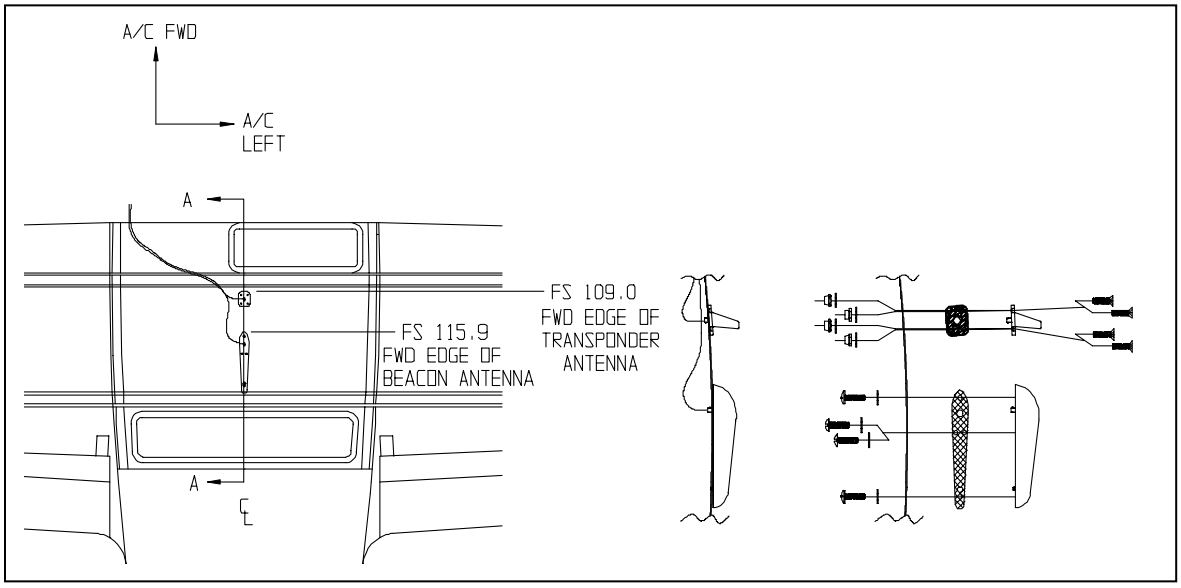

Figure 34 - 39 Transponder Antenna Location

c. **Transponder Installation –** Perform steps 2-5 above in reverse order.

d. **Transponder Antenna Removal** (See Figure 34 - 39)
   1. Turn off the master switches.
   2. Remove the access plate under the center of the wing (aircraft belly area).
   3. Unfasten the antenna cable.
   4. Unfasten the antenna, and remove the antenna and the gasket.

e. **Transponder Antenna Installation**
   1. Turn off the master switches.
   2. Follow step 2-4 above, in reverse order.
   3. Ensure proper electrical bonding of the antenna to the aircraft surface.
   4. Restore electrical power.

f. **SL70 Initial Set-up Procedure**
   1. Turn the aircraft and avionics power switches ON and the SL70 unit OFF.
   2. On the SL70 unit, press and hold the IDENT and ALT buttons at the same time, and turn the power knob to ON while holding the buttons for 3 to 4 seconds.
   3. Rotate the large outer knob to the SL70 CONF page.
   4. Rotate the small inner knob to the ASRC page.
   5. Press the IDENT button (the altitude field will start to flash), and rotate the small inner knob to select SER.

6. Press the IDENT button to save the selection.
7. Rotate the small inner knob to the BAUD page.
8. Press the IDENT button (the BAUD field will start to flash), and rotate the small inner knob to select 1200.
9. Press the IDENT button to save the selection.
10. When setup is complete, return to normal operation by switching the unit's power knob to OFF then back to ON.

g. **GTX 327 and GTX 330 Initial Set-up Procedure –** Refer to the Garmin GTX 327 Transponder Installation Manual, P/N 190-002187-02, or the Garmin GTX 330 Transponder Installation Manual, P/N 190-00207-02 for set-up procedures.

34-18. BENDIX/KING KG 102A REMOTE GYRO AND KI525A HSI INDICATOR – If the Avidyne Entegra Primary Flight Display (PFD) is installed in the aircraft the KG 102A remote gyro and KI525A HIS indicator will not be installed in the aircraft.

a. **KG 102A Remote Gyro Removal –** The KG 102A is located in the avionics shelf under the baggage compartment floor. See Figure 34 - 40 .
   1. Remove the carpet and access panel from the baggage compartment floor.
   2. Loosen the mounting screws that hold the KG 102A in its tray. See Figure 34 - 42 .
   3. Disconnect all connectors to the unit.
   4. Remove the unit.

b. **KG 102A Remote Gyro Installation** (See Figure 34 - 42)
   1. Connect the wiring and connectors to the KG 102A.
   2. Place the KG 102A in its place in the instrument shelf.
   3. Install screws to connect the KG 102A to the tray and torque screws 10 to 12 in.-lbs.
   4. Install the access panel.
   5. Replace the carpet.

c. **KMT 112 Slaving Transmitter Removal** (See Figure 34 - 42)
   1. Remove the access panel on the underside of the left outboard wing**.** See Figure 34 - 40 **.**
   2. Disconnect the electrical connections.
   3. Unfasten the nuts, and remove the unit.

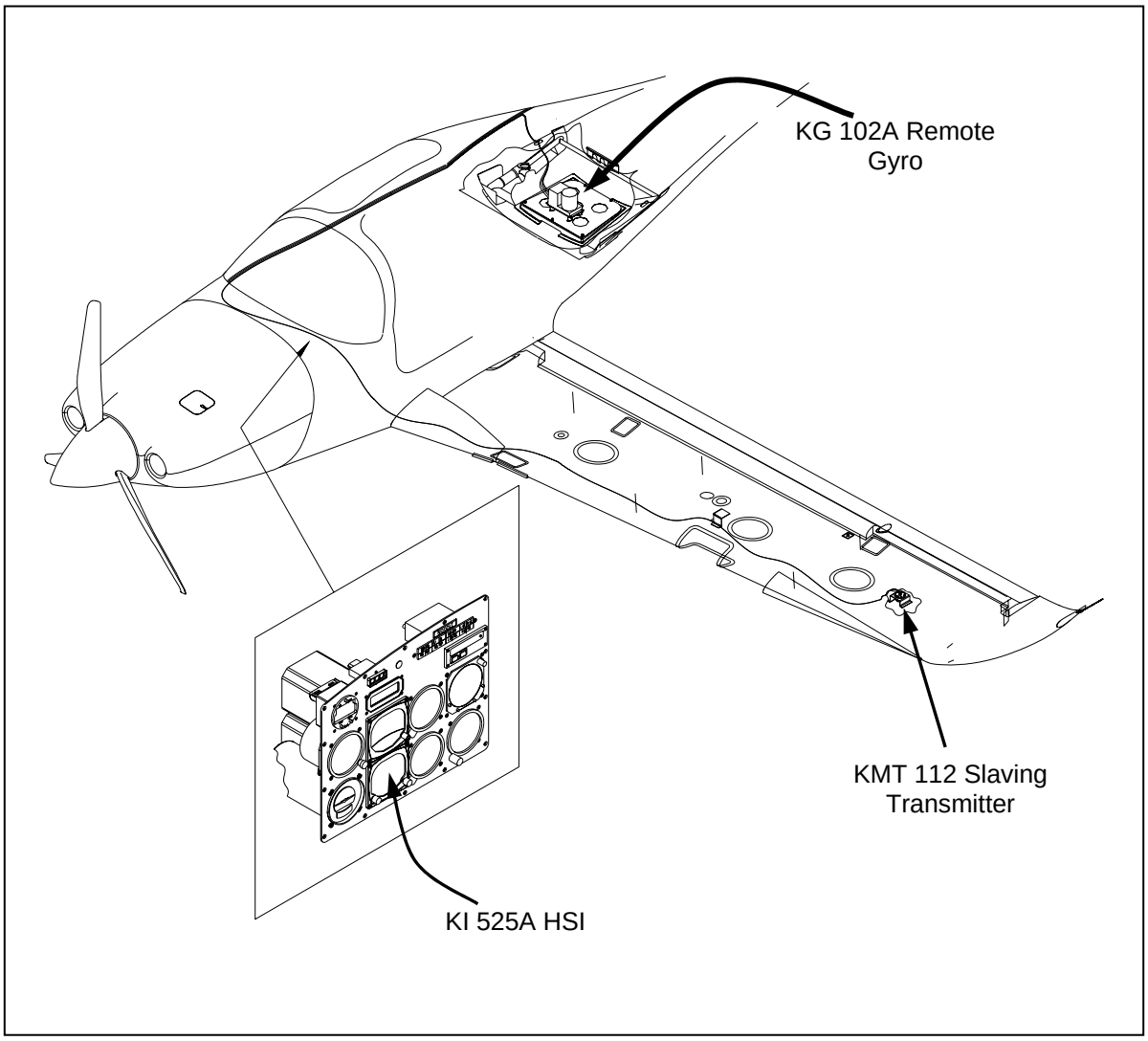

Figure 34 - 40 HSI – Remote Slaving System

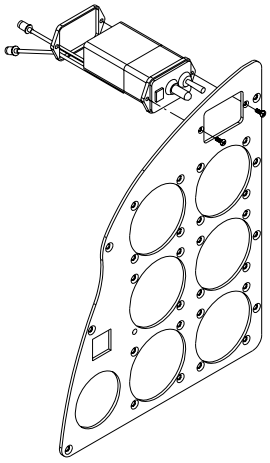

Figure 34 - 41 KA 51B Slaving Control Location

d. **KMT 112 Slaving Transmitter Installation – Use King Installation Manual**
   1. Install the unit with the arrow on top facing forward, using the same hardware or new brass hardware only.
   2. Connect the electrical connectors.
   3. At a compass rose, adjust the unit per manufacturer's manual.
   4. Finger-tighten the nuts. Do not over-tighten.
   5. Install the access panel.

e. **KI 525A Removal** – See Figure 34 - 43 .
   The KI 525A is located in main plate of the instrument panel.
   1. Unfasten the main plate, and carefully slide it out far enough to work on the back side of the KI 525A.
   2. Unfasten the mounting screws that surround the KI 525A, and slide it forward while slipping the mounting plate off the back of the unit.
   3. Disconnect connectors installed into the back of the KI 525A.

f. **KI 525A Installation** – See Figure 34 - 43 .
   1. Slide the unit into the instrument panel, and slide the mounting bracket over the back of the unit.
   2. Install the connectors into the back of the KI 525A.
   3. Fasten the mounting screws through the front of the instrument panel and then into the mounting plate. Torque screws 6 to 8 in.-lbs.

g. **KA 51B Slaving Control Removal** – See Figure 34 - 41.
   The KA 51B is located in the engine instrument plate of the instrument panel.
   1. Unfasten the engine instrument plate, and carefully slide it out far enough to work on the backside of the KA 51B.
   2. Unfasten the mounting screws from the KA 51B, and slide it forward.
   3. Disconnect the connectors installed into the back of the KA 51B.

h. **KA 51B Installation** – See Figure 34 - 41.
   1. Follow steps 2 through 4, above, in reverse order.
   2. At a compass rose, adjust the unit per the manufacturer's manual.

34-19. TERMINAL JUNCTION RAIL

a. **Terminal Junction Rail Removal –** The terminal junction rail is located on the vertical brace on the backside of the instrument panel. See Figure 34 - 44 or Figure 34 - 45.
   1. Remove the glare shield per instructions in Chapter 25.
   2. Remove the two 6-32 flat head screws, self-locking nuts, and washers for S/N 42003 to 42023. Remove the 6-32 flat head screws for S/N 42024 and on.
   3. Remove the terminal junction rail.

b. **Terminal Junction Rail Installation** – Installation is a direct reversal of removal. Torque screws 10 to 12 in.-lbs.

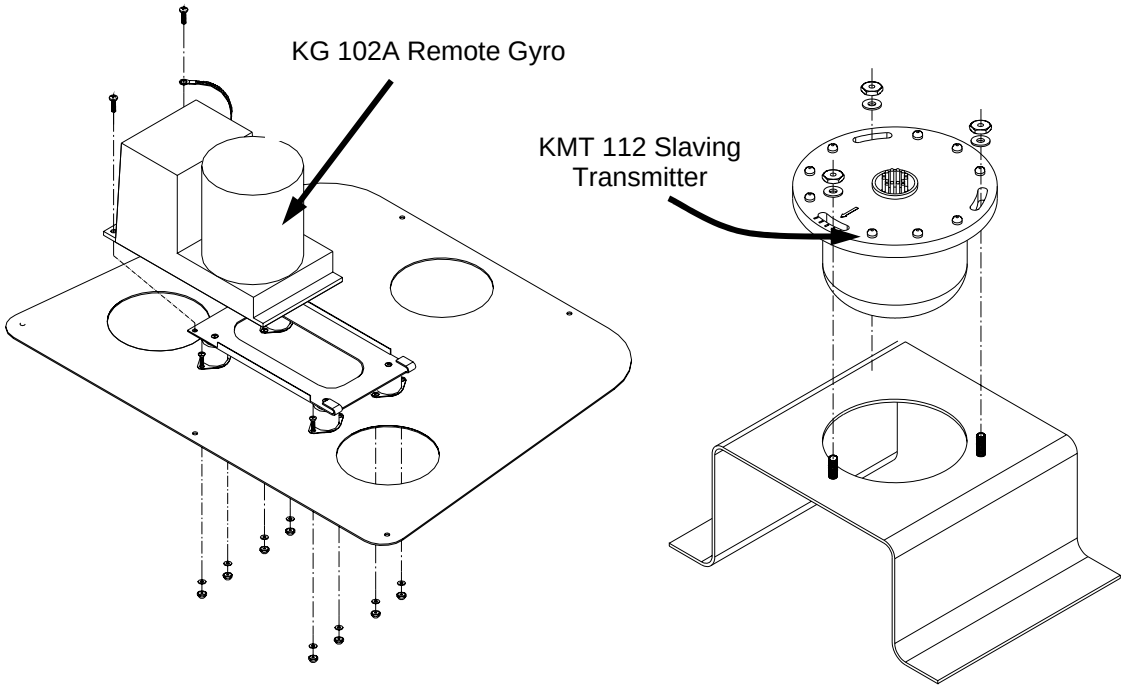

Figure 34 - 42 HSI Remote System Installation

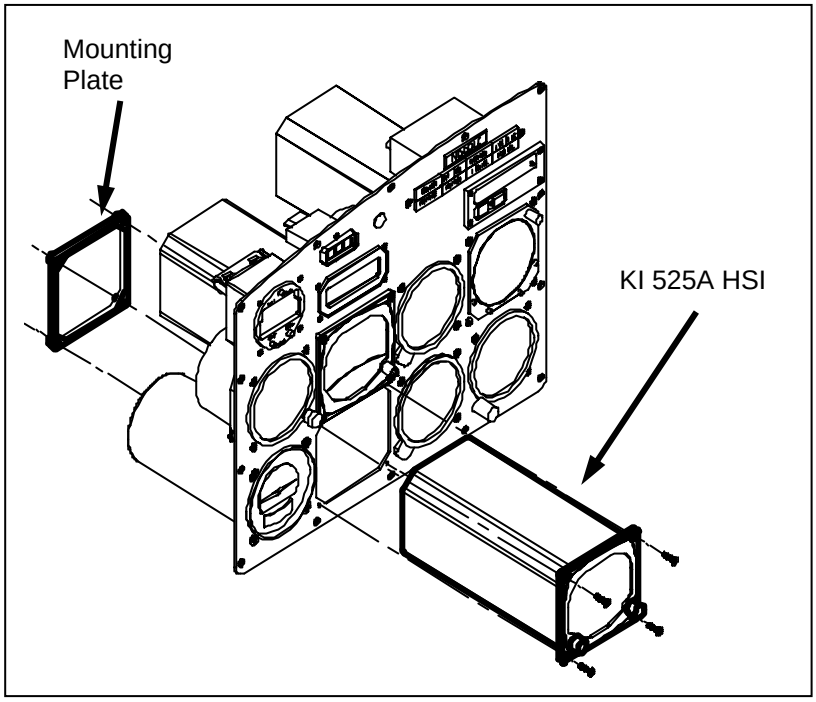

Figure 34 - 43 KI 525A HSI Installation

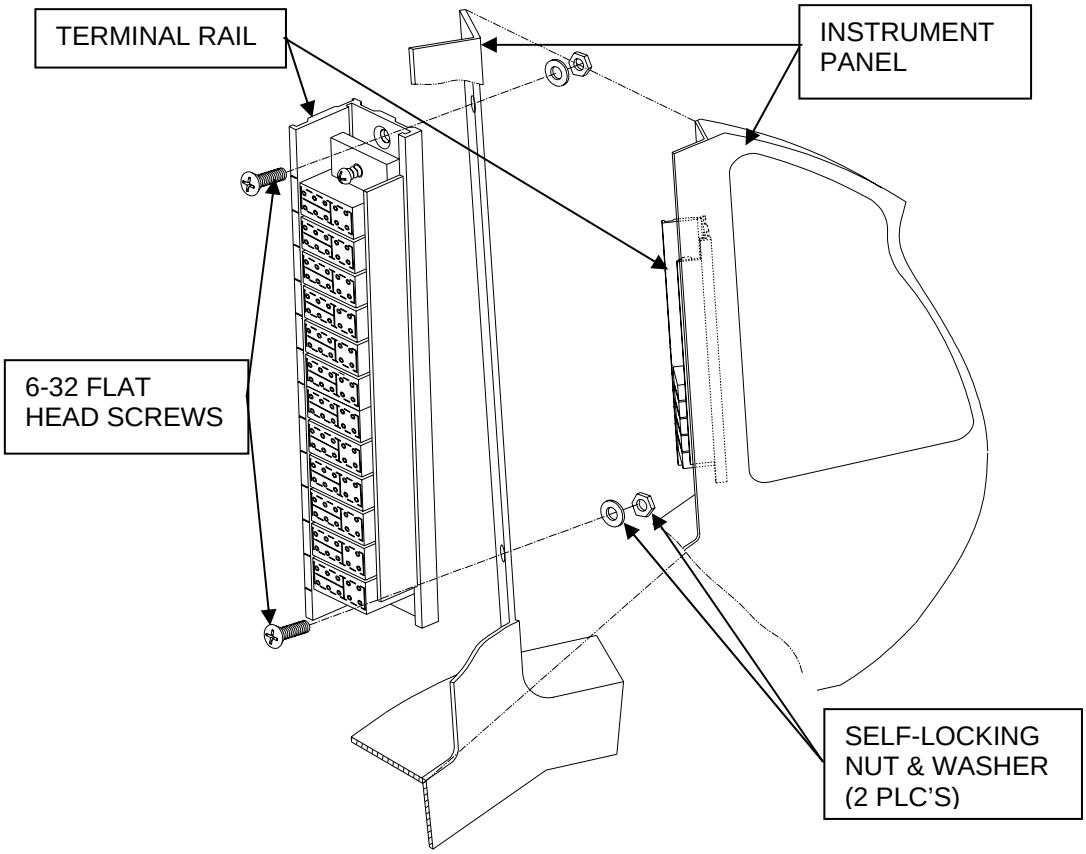

Figure 34 - 44 Terminal Junction Rail (S/N 42003 to 42023)

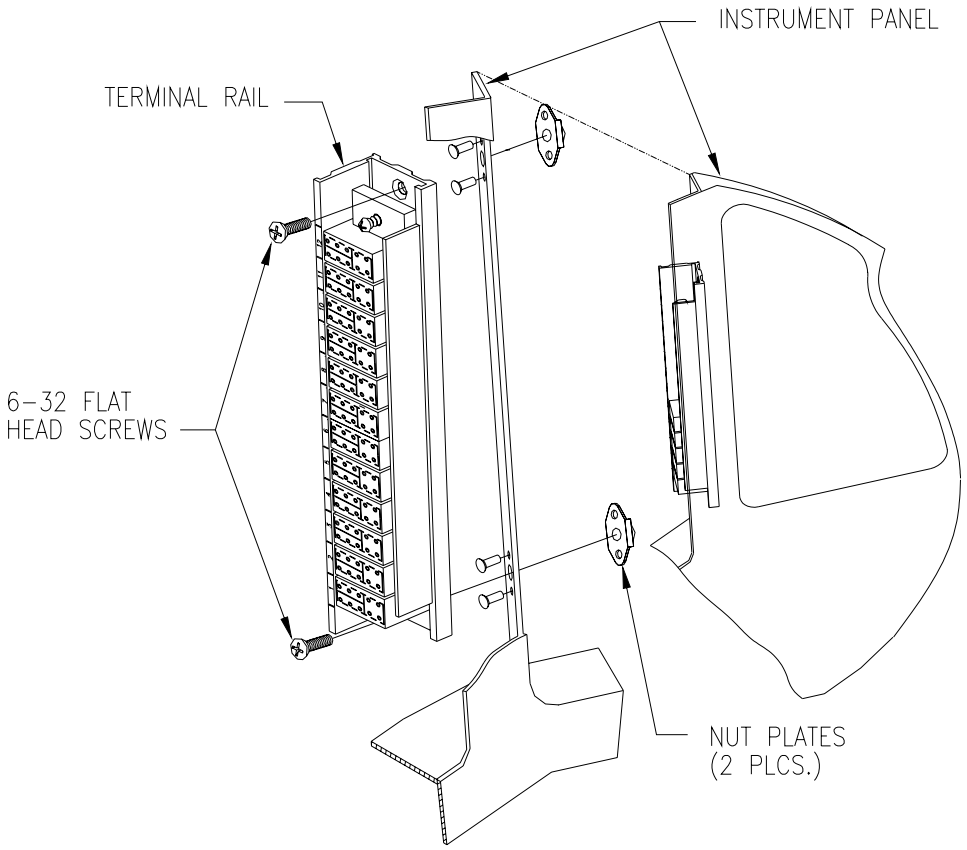

Figure 34 - 45 Terminal Junction Rail (S/N 42024 and on)

34-20. ANTENNA SPLITTER

a. **Basic or Avidyne Option**

1. **Garmin AT Avionics**

a) **Antenna Splitter Removal –** The antenna splitter is located on the inside right side of the radio rack. See Figure 34 - 46. An antenna splitter may not be present depending upon the avionics package installed in the airplane.

1) Remove the center console per instructions in Chapter 25 to gain access to the radio rack.
2) Remove two truss head 6-32 screws, self-locking nuts, and washers.
3) Remove the antenna splitter.

b) **Antenna Splitter (Garmin AT Avionics) Installation –** Installation is a direct reversal of removal. Torque screws snug.

2. **Garmin Avionics**

a) **Antenna Splitter () Removal –** The antenna splitter is located on the inside right side of the radio rack. See Figure 34 - 47

1) Remove the center console per instructions in Chapter 25 to gain access to the radio rack.
2) Remove the two truss head 10-32 screws, self-locking nuts, and washers.
3) Remove the antenna splitter.

b) **Antenna Splitter (Garmin Avionics) Installation –** Installation is a direct reversal of removal. Torque screws snug.

b. **Garmin G1000 Option**

1. **Antenna Splitter Removal –** The antenna splitter is located under the instrument panel to the right of the tower on the co-pilot's side. It is positioned under the LRU rack to the far right.
   a) Access the antenna splitter from the underside of the instrument panel.
   b) Remove the two pan head 10-32 screws.
   c) Remove the antenna splitter.
2. **Antenna Splitter Installation –** Installation is a direct reversal of removal. Torque screws snug.

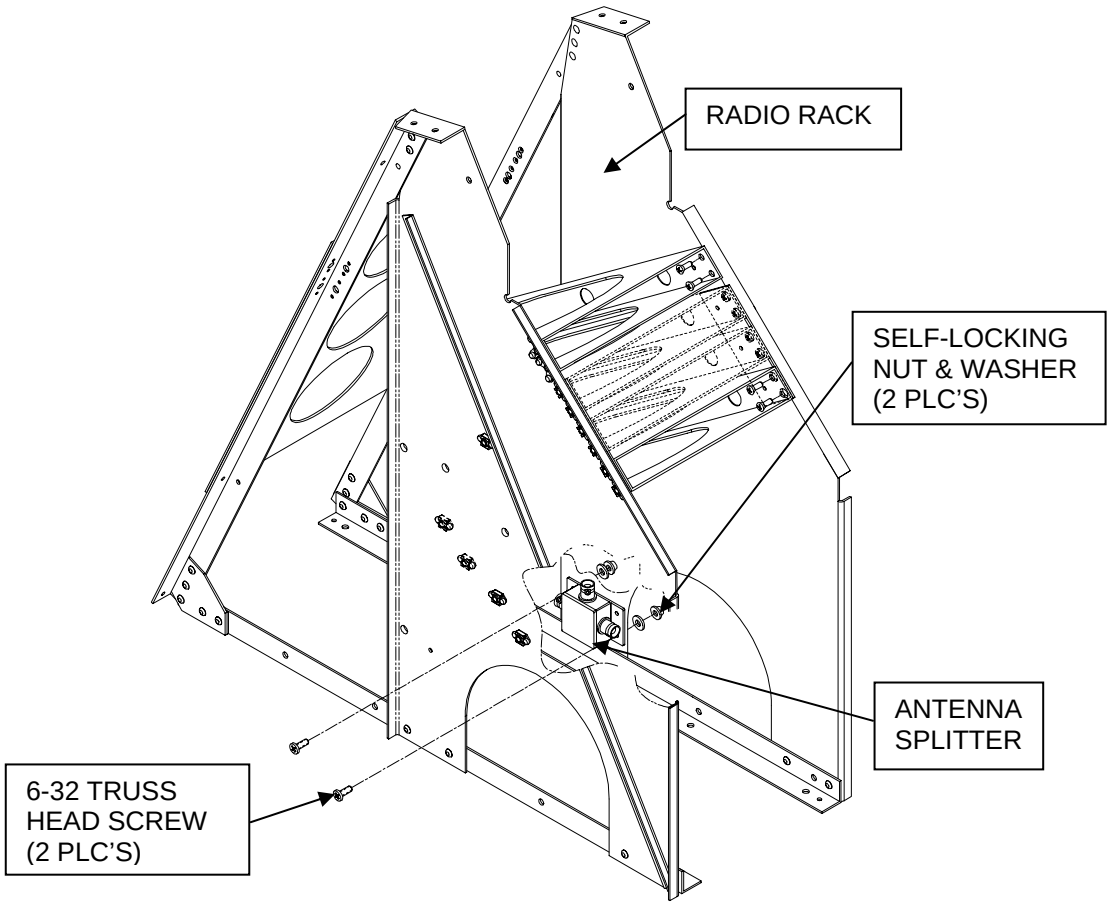

Figure 34 - 46 Antenna Splitter (Garmin AT Avionics)

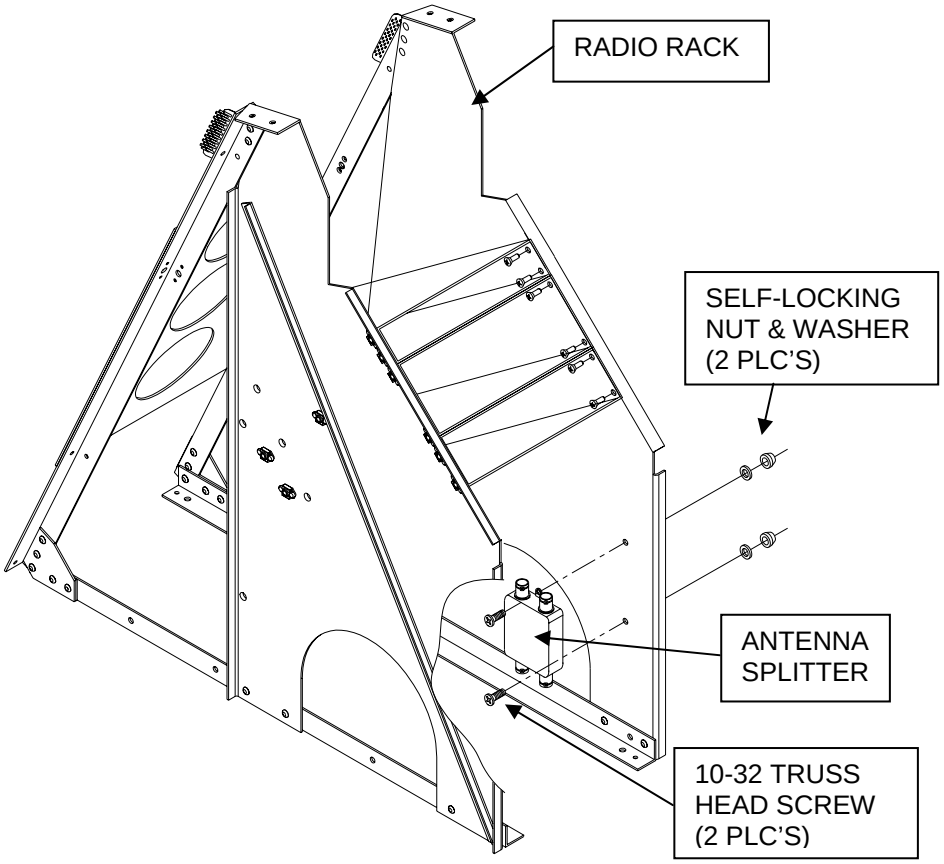

Figure 34 - 47 Antenna Splitter (Garmin Avionics)

34-21. GARMIN AVIONICS COOLING FAN

a. **Basic or Avidyne Option**

1. **Cooling fan Removal –** The Garmin avionics cooling fan is located on the inside of the radio rack. See Figure 34 - 48.
   a) Remove the center console side panels per instructions in Chapter 25 to gain access to the radio rack.
   b) Loosen the tie wrap where the 5/8" hoses are connected to the fan enclosure and disconnect the hoses.
   c) Remove three pan head 6-32 screws, self-locking nuts, and washers.
   d) Unplug and remove the fan.
2. **Cooling Fan Installation –** Installation is a direct reversal of removal. Torque screws 10 to 12 in.-lbs.

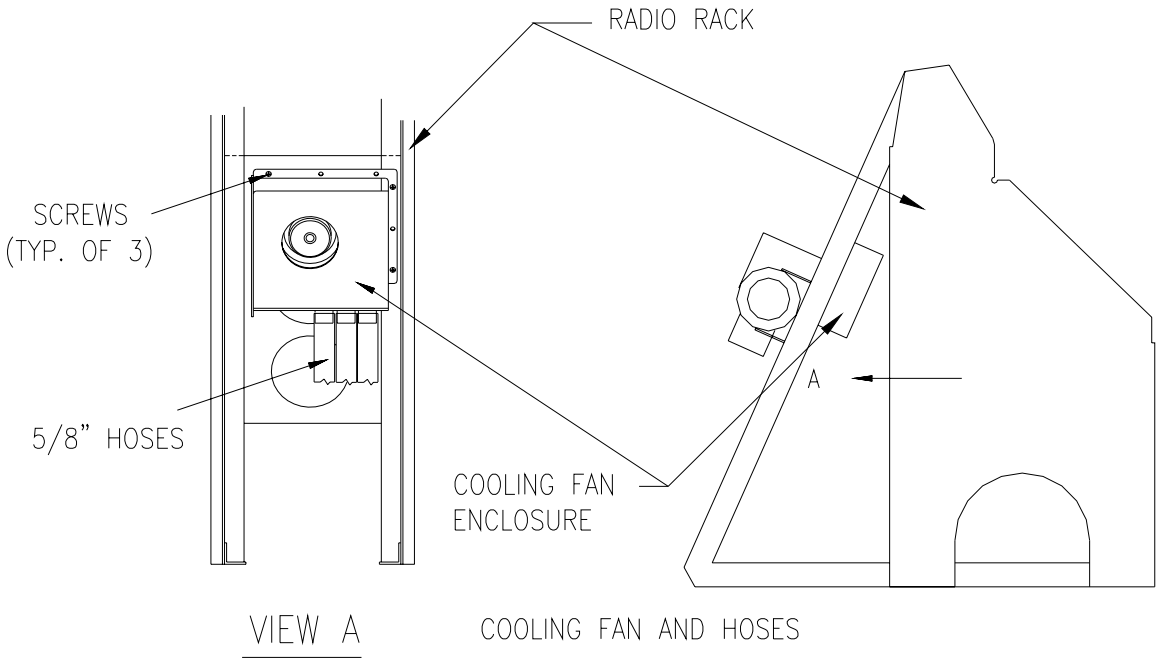

Figure 34 - 48 Garmin Avionics Cooling Fan

b. **Garmin G1000 Option –** There are three cooling fans. One in the LRU rack behind the PFD, one in the LRU rack behind the MFD, and one mounted under the instrument panel on the pilot's side.
   1. **Cooling fan Removal**
      a) Remove the PFD or MFD, as required, per Chapter 34. Access the third fan from the underside of the instrument panel.
      b) Remove six Philips screws attaching the fan and its mounting bracket to the LRU rack or three pan head 6-32 screws attaching the fan to the underside of the instrument panel, as applicable.
      c) Gently remove the fan.
      d) The fans have been removed from the LRU rack, they may be removed from their brackets by removing three Phillips screws.
   2. **Cooling Fan Installation –** Installation is a direct reversal of removal. Torque screws snug.

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34-22. AVIDYNE ENTEGRA PRIMARY FLIGHT DISPLAY (PFD)

a. **Primary Flight Display Removal –** The PFD is located in the left side of the instrument panel. See Figure 34 - 53.
   1. Ensure both the master and avionics switches are in the off position
   2. Loosen the four 5/64" allen head screw fasteners at the four corners of the PFD. The fasteners will not come out of the PFD, only loosen.
   3. Slide the unit out of the instrument panel.
   4. Disconnect the pitot static hoses by pressing the quick release disconnect button on the connector of each hose.
   5. Loosen the two jackscrews on each cable connector attached to the back of the PFD and disconnect the cables.

b. **Primary Flight Display Installation**
   1. Remove the Avidyne EX5000 Multi-Function Display as described below to allow easier access to the PFD.
   2. Attach the cable connectors to the back of the PFD and tighten the jackscrews.
   3. Attach the pitot static hose connectors.
   4. Hold the PFD with one hand at the top front and the other hand supporting the bottom. Slide the PFD into the instrument panel while lifting up on the rear edge and simultaneously pushing the upper edge under the glare shield.
   5. Ensure the PFD mating clip slides over the horizontal support bracket located approximately 8 inches behind the instrument panel.
   6. Ensure that the PFD cables and pitot static hoses are routed away from any moving parts behind the instrument panel.
   7. Secure the PFD to the instrument panel with the four corner mounted 5/64" allen head screw fasteners.
   8. Perform a functional check in accordance with section 34-22.d. System Setup and Checkout.

c. **Troubleshooting Information**

NOTE

For troubleshooting instructions recommending wiring checks, refer to the FlightMax Entegra Primary Flight Display Installation Manual, P/N 600-00080-000 and/or the FlightMax Entegra Series PFD Pilot's Guide, P/N 600-00096-000.

Component	*Trouble*	*Probable Cause*	*Remedy*
Attitude Display	Pitch ladder and horizon removed and replaced with red X	PFD AHRS failure	Replace PFD
Altitude, Airspeed, *Vertical* Velocity Display	Instruments removed and replaced with red X	PFD Air Data Computer failure	Replace PFD
Heading	Heading indication removed and replaced with red X	Break in wire between PFD and Magnetometer	Test and repair wiring.
		Magnetometer failed	Replace Magnetometer/OAT Assembly
		PFD RS-422 or Magnetometer output power failure	Replace PFD
	Excessive heading error	Magnetometer not properly calibrated	Calibrate Magnetometer
Alignment	PFD fails to align after power up	Aircraft was moved during power up alignment	Cycle PFD power and remain stationary during power up alignment
		PFD ADAHRS failure	Replace PFD
OAT	Display indicates red dashes	OAT probe failed	Replace Magnetometer/OAT Assembly
	Excessive temperature error	OAT probe failed	Replace Magnetometer/OAT Assembly
True Airspeed	Display indicates red dashes	OAT probe failed	Replace Magnetometer/OAT Assembly

Component	*Trouble*	*Probable Cause*	*Remedy*
Wind Vector	Display indicates dashes	Winds aloft have been calculated as light and variable or less than 5 knots	Behavior by design, no action required
		PFD not receiving ground speed. Break in wire between PFD and VHF/GPS NAV 1 & VHF/GPS NAV 2	Test and repair wiring
		Aircraft has zero ground speed	Behavior by design, no action required
VHF /GPS NAV 1	PFD Primary Nav, Bearing, or Auxiliary indication displays dashes when GPS 1 or VLOC 1 selected	VHF/GPS NAV 1 not configured properly	Configure unit
		Break in wire between PFD and VHF/GPS NAV 1	Test and repair wiring
		VHF/GPS NAV 1 ARINC 429 Output failed	Replace VHF/GPS NAV 1
		PFD ARINC 429 input failed	Replace PFD
	GPS1 does not slave to PFD course pointer in OBS mode	Break in ARINC 429 wire between PFD and GPS NAV 1	Test and repair wire
		ARINC 429 Receive Input failed	Replace VHF/GPS NAV 1
		ARINC 429 Transmit output failed	Replace PFD
VHF /GPS NAV 2	PFD Primary Nav, Bearing, or Auxiliary indication displays dashes when GPS 2 or VLOC 2 selected	VHF/GPS NAV 2 not configured properly	Configure unit
		Break in wire between PFD and VHF/GPS NAV 2	Test and repair wiring
		VHF/GPS NAV 2 ARINC 429 Output failed	Replace VHF/GPS NAV 2
		PFD ARINC 429 input failed	Replace PFD
	GPS2 does not slave to PFD course pointer in OBS mode	Break in ARINC 429 wire between PFD and GPS NAV 2	Test and repair wire
		ARINC 429 Receive Input failed	Replace VHF/GPS NAV 2
		ARINC 429 Transmit output failed	Replace PFD

Component	*Trouble*	*Probable Cause*	*Remedy*
Autopilot Integration	Airspeed, Heading, or Altitude bugs do not "fill" but autopilot modes still engage.	Break in Annunciator wires between PFD and AP Computer	Test and repair wiring
		Autopilot Computer interface failure	Replace unit
		PFD interface failure	Replace unit
	Autopilot annunciators on the PFD (if equipped) do not illuminate or illuminate erratically.	Break in Annunciator wires between PFD and AP Computer	Test and repair wiring
		Autopilot Computer interface failure	Replace unit
		PFD interface failure	Replace unit
	Autopilot VS mode does not track PFD Vertical Speed setting, but autopilot mode engages.	Break in VS Command wires between PFD and Autopilot Computer	Test and repair wiring
	Autopilot HDG mode does not track PFD Heading setting, but autopilot mode engages.	Break in Heading Datum wire between PFD and Autopilot Computer	Test and repair wiring
	Autopilot HDG mode does not track PFD Heading setting, but autopilot mode engages (con.)	Autopilot Computer interface failure	Replace unit
		PFD interface failure	Replace unit
		PFD heading bug out of alignment	Perform PFD heading bug calibration
	Autopilot NAV mode does not track PFD Nav Course Pointer setting or CDI.	Break in Course Datum and Cross track wires between PFD and Autopilot Computer	Test and repair wiring
		Autopilot Computer interface failure	Replace unit
		PFD interface failure	Replace unit
		Autopilot out of alignment	Perform autopilot alignment in NAV mode
	Autopilot NAV annunciator flashes and FAIL indication appears.	Break in CDI Flag wire between PFD and Autopilot Computer	Test and repair wiring
		Autopilot Computer interface failure	Replace unit
		VHF/GPS NAV 1 or 2 Nav data invalid	Refer to VHF/GPS NAV 1 & 2 Troubleshooting procedure
		PFD interface failure	Replace unit
	Autopilot APR mode does not track PFD CDI or VDI.	Break in CDI and VDI wires between PFD and Autopilot Computer	Test and repair wiring
		Autopilot Computer interface failure	Replace unit
		PFD interface failure	Replace unit

Component	*Trouble*	*Probable Cause*	*Remedy*
Autopilot Integration (Continued)	Autopilot APR mode fails to engage.	Break in Glideslope flag or LOC Switch wires between PFD and Autopilot Computer	Test and repair wiring
		Autopilot Computer interface failure	Replace unit
		PFD interface failure	Replace unit
	Autopilot Computer display indicates vertical speed value	Break in ALT Select wire between PFD and Autopilot	Test and repair wiring
		Autopilot Computer interface Autopilot Integration failure	Replace unit
		PFD interface failure	Replace unit
	Autopilot GPSS mode fails to engage in flight (GPSS mode will not engage during a ground test)	Break in PFD ARINC 429 Transmit or Autopilot Relay Control wire	Test and repair wiring
		Autopilot Relay failure	Replace relay
		Autopilot Computer interface failure	Replace unit
		PFD interface failure	Replace unit
	Autopilot GPSS only follows GPS 1 source	Break in Autopilot Relay Control wire	Test and repair wire
		Autopilot Relay failure	Replace relay
		PFD interface failure	Replace unit
	Flight Director command bars (if equipped) do not align to attitude reference symbol or behave erratically. (Test alignment on in level flight in smooth air with HDG and ALT modes engaged)	PFD out of alignment	Perform PFD heading bug calibration
		Break in pitch / roll command signal wires between PFD and Autopilot computer	Test and repair wiring
		Autopilot computer interface failure	Replace unit
		PFD interface failure	Replace unit

d. **System Setup and Checkout –** Any time the Avidyne PFD or Magnetometer/OAT Sensor Assembly is replaced with another unit, these system setup procedures should be performed to assure that the unit is properly configured for the installation.

Turn on the PFD by applying power to the aircraft. Allow the PFD to initialize. This will take approximately 3 minutes.

1. **System Setup Page Access –** Apply power to all the sensors that interface with the PFD, including: the GPS and the autopilot. System Setup Page will allow you to configure the PFD. The System Setup Page can only be accessed by pressing the line select keys in a specific manner.

The System Setup Page is accessible as follows:

a) Turn on the PFD by applying power to the aircraft via the appropriate aircraft power switches (Avionics Master, etc.) switches.
b) The system will begin its normal start up sequence.
c) Simultaneously press and hold the top left (L1), and 3rd down from the top left (L3) Line Select keys for until the count down timer in the lower left corner of the display indicates zero seconds.
d) At the end of a displayed countdown clock adjacent to the L4 line select key, the System Setup Page will appear.

2. **Aircraft Model Setup –** Perform the following steps to configure the PFD for the correct aircraft model.
   a) Enter the System Setup Page.
   b) Press LSK R4, labeled "A/C Model", as required until the appropriate aircraft model is displayed in the button label. Note that the model field in the middle of the page reflects the model selected. For those aircraft models that have a "+" suffix displayed in the LSK R4 button label, the OAT will be suppressed from the display.
3. **Barometric Units Setting –** Perform the following steps to configure the PFD for desired barometric units.
   a) Enter the System Setup Page.
   b) Press LSK R3, labeled "Perform A/P Cal", to enter the Autopilot Calibration page.
   c) Press LSK R4, labeled "Baro Unit", to cycle through barometric pressure units (inHg, hPa, Mb).
   d) Once the desired units have been selected and appear in the button label, press the "Back to PFD" button (LSK L4).

4. **GNS-430 Nav/Com Setup**
   a) PFD Unit – The ARINC 429 ports on the PFD come hard-wired and pre-assigned. GNS-430 Unit #1 (top unit in radio rack) must be connected to ARINC port 1 (GPS output line) and port 2 (VLOC output line). GNS-430 Unit #2 (if installed) must be connected to ARINC port 3 (GPS output line) and port 4 (VLOC output line). Use the line select keys R1, R2 to configure the PFD in the following manner:
      1) Line Select Key (LSK) R1: GPS1 Port: ARINC 1;
      2) LSK R1: GPS1 Port: NONE, if GNS-430 No. 1 not installed;
      3) LSK R2: GPS2 Port: ARINC 3, if GNS-430 No. 2 installed;
      4) LSK R2: GPS2 Port: NONE, if GNS-430 No. 2 not installed.

      Once complete, verify the displayed configuration in the sensor suite section of the System Setup page accurately reflects actual configuration.
   b) GNS-430 Unit(s)

NOTE

Refer to the GNS 430 users guide for GNS 430 detailed operations.

Place the GNS 430 unit(s) into Maintenance Mode via the following steps:

1) Ensure GNS 430 unit(s) off
2) Depress the "ENT" button as the power to the GNS 430 unit(s) is applied. Continue to depress the ENTR button until the GNS 430 self -test is completed.
3) Depress the ENT button twice to display the "MAIN ARINC 429 CONFIG" page. Configure the GNS 430 unit(s) to have the following mandatory settings:

Main ARINC 429 Configuration Page

Field	*Garmin 1*	*Garmin 2 (if installed)*
IN 1	Low, Sandel EHSI	Low, Sandel EHSI
IN 2	Low, OFF	Low, OFF
Out	Low, GAMA 429 Graphics	Low, GAMA 429 Graphics
SDI	LNAV1	LNAV2

4) Turn the right inner knob until the "MAIN RS232 CONFIG" page is displayed.
5) Configure the GNS 430 unit(s) to have the following mandatory settings:

Main RS232 Configuration Page

Field	*GNS-430 No. 1*		*GNS-430 No. 2 (if installed)*	
	Input	Output	Input	Output
CHNL 1	Off (1)	Aviation	Off (1)	Aviation
CHNL 2	Off (1)	Off (1)	Off (1)	Off (1)
CHNL 3	Crossfill	Crossfill	Off (1)	Off (1)
CHNL 4	Off (1)	Off (1)	Off (1)	Off (1)

(1) Unless other equipment installed (i.e., Stormscope)

6) Turn the right inner knob until the "MAIN INPUTS 2" page is displayed. Verify CDI = "GPS"
7) Turn the right inner knob until the "MAIN DISCRETE OUTPUTS" page is displayed. Verify Discrete Toggle = "APR"
8) Turn the right inner knob until the "VOR/LOC/GS ARINC 429 CONFIG" page is displayed
9) Configure the GNS 430 unit(s) to have the following mandatory settings:

VOR/LOC/GS ARINC 429 Configuration Page

Field	*GNS-430 No. 1*	*GNS-430 No. 2 (if installed)*
Speed	Low – Low	Low – Low
IN 2	VOR/ILS1	VOR/ILS2
DME Mode	Directed Freq. 1	Directed Freq. 2

5. **Autopilot Setup**
   a) **PFD Unit –** Perform the following steps when replacing a PFD.
      1) Enter the System Setup Page (see section 34-22.d) and press the line select key R3, labeled "Perform A/P Cal" and depress line select key R1 " A/P Type" to select the appropriate autopilot.
      2) Select line select key L4 labeled "Back to PFD" and reenter the System Setup Page. Verify the displayed autopilot configuration in the sensor suite section of the System Setup page accurately reflects actual configuration.

   b) **S-Tec System 55x Autopilot Unit –** Perform the following steps when replacing the PFD or Autopilot unit.

      If the autopilot installed in the aircraft is an S-Tec System 55x, ensure it is configured as a KCS-55 compatible unit. Refer to the S-Tec System 55x Installation Manual for details.

      NOTE

      The following steps are to be performed in smooth air.

      1) While NAV (without GPSS) mode is engaged on the System 55x, using a GNS 430 GPS source, insert an appropriately sized screwdriver into the slotted hole in the 55X faceplate
      2) Perform a NAV course alignment by slowly adjusting the screw until the aircraft closely tracks the GPS course. Allow the aircraft to stabilize between adjustments. This may take several minutes and require several adjustments.
      3) Engage the 55x in HDG mode and allow aircraft to stabilize. The heading bug may be offset from the lubber line by several degrees.
      4) Turn OFF all GPS/NAV systems.
      5) Enter the PFD System Setup Page (See section 34-22.d).
      6) Turn ON all GPS/NAV systems.
      7) Select "Perform A/P Cal" on the PFD
      8) Select "Sync HDG". Autopilot should immediately command the aircraft to turn toward the heading bug.
      9) Select "Back to PFD".
      10) Verify HDG mode accurately tracks heading bug.
      11) Engage NAV (without GPSS) mode on the autopilot.

12) Verify NAV mode accurately tracks flight plan (Flight plan or waypoint will need to be entered).
13) A/P calibration is complete.

NOTE

If autopilot does not track the heading bug or NAV mode correctly repeat steps in this procedure.

6. **PFD System Calibrations –** The PFD and Magnetometer require calibrations upon initial installation, when replaced, or bi-annual periodic inspection. The following two aircraft level calibration procedures are listed below:

Calibration	*When Performed*	*Notes*	*Periodic Calibration*
IRU	Initial installation and replacement	Must be performed first upon initial installation with Magnetometer	None
Magnetometer	Initial installation, replacement, PFD replacement, or bi-annual inspection	Must be performed after IRU calibration upon initial installation or PFD replacement	Every 24 months

NOTE

The IRU calibration must be completed prior to the Magnetometer calibration.

a) IRU Calibration Procedure

Approximate duration: 15 Minutes

Required Equipment: Digital Level, Resolution: 0.25 °, Accuracy: +/- 0.25°

Recommended Personnel: One avionics technician

Procedure:

1) Avionics power applied.
2) Allow the PFD to align (approximately 3 minutes) until the ADAHRS countdown timer expires and the Gyro Warm-up box is removed.
3) Wait in the aligned state for an additional 10 minutes.
   Enter System Setup mode on the PFD by following steps in section 34-22.d.
4) Press the IRU Install line select key (L2). The IRU Installation page is displayed.
5) Place a level 2X4-like piece of wood between the two door jams, or place an inclinometer on top (or bottom) of the PFD Bezel, aligned to display roll angle.
6) Place a digital level on the baggage compartment doorsill of the airplane

7) Follow the directions on the IRU installation page.

NOTE

The only two axis that are changed via installation calibration input are the pitch axis and roll axis.

NOTE

The pitch attitude and roll attitude as determined by the digital level should each have a resolution of at least ¼ degree.

8) Press the Install IRU line select key (R4) when proper pitch and roll is displayed.
9) After the display indicates the IRU calibration is done, cycle power on the PFD in order for the IRU calibration to take effect.

b) Magnetometer Calibration Procedure

Approximate duration:	20 minutes (assuming airplane is free to rotate 360 degrees without magnetic disturbances).
Required equipment:	Installed PFD Installed magnetometer Compass: Resolution: 1 ° , Accuracy: +/- 1 °
Recommended personnel:	2 avionics technicians (one in the cockpit to push required bezel buttons, one on the wing to push the aircraft to new headings)

Procedure:

NOTE

Ensure the IRU installation calibration procedure has been completed.

1) Airplane doors closed.
2) Flaps in retracted position.
3) Engine off.
4) Airplane in level flight attitude.
5) Install the compass onto the top surface of the left tail surface.

NOTE

A compass rose with 30 degree positioning increments may be used in lieu of a tail-mounted compass.

6) Position lubber line to be parallel with longitudinal axis of the airplane.
7) Position compass to be approximately ½ chord length back from leading edge of tail and 43 inches outboard from fuselage.
8) Master switch in ON position. (Consider attaching an external power cart until ready for step 11.) PFD may already be on and aligned from IRU install. If so, skip to step 11. All other aircraft equipment shall be operating.
9) Allow the PFD to align (approximately 3 minutes) until the ADAHRS countdown timer expires and is removed from the HSI.
10) Wait in the aligned state for an additional 10 minutes.
11) Enter System Setup mode on the PFD by following steps in section 34-22.d.
12) Press the Perform Mag Cal line select key (L1). The Magnetometer Calibration page is displayed.
13) Align centerline of airplane on magnetic north heading as indicated by tail-mounted compass (or compass rose) by pushing the aircraft as required. Ensure this is being done clear of magnetic disturbances in the pavement or immediate vicinity.
14) Follow the directions on the Magnetometer Calibration page (press the Calibrate Heading button and wait until "Done" is displayed).
15) Align centerline to airplane on consecutive 30° headings as indicated by the tail-mounted compass (or compass rose) by pushing the aircraft as required and continue following the directions of the Magnetometer Calibration page until complete.

NOTE

Pressing (R3) "Redo Previous" provides undo capability.

NOTE

Aborting the calibration process (by exiting the Magnetometer Calibration Page or shutting off power to the PFD) prior to completion will result in the loss of stored calibration parameters. Restarting the calibration process from the beginning would be required.

16) Press the "Back to PFD" line select key (L4) when complete.
17) Align the centerline of the airplane with consecutive 90° headings as indicated on the tail-mounted magnetic compass (or compass rose) and verify the accuracy of the PFD heading display is within +/- 4 degrees of the tail mounted magnetic compass (or compass rose).
18) Calibration complete.

7. Post Installation Check
   a) Lighting Check – Rotate the "INST" instrument lighting control rheostat through its full range of motion. Ensure the PFD LED lights around all bezel keys and knobs come on and match the MFD LED lighting.
   b) GPS NAV-COMM Check

NOTE

This procedure assumes both GNS-430s are installed.

1) PFD Nav Button– Ensure GPS1 displayed. Observe that the GNS-430 No. 1 CDI button label displays GPS.
2) GNS-430 No. 1 CDI button– Press once and observe CDI button displays VLOC label. Observe the PFD Nav button changes to VLOC1.
3) GNS-430 No. 1 CDI button– Press once and observe the CDI button label displays GPS. Note that the PFD Nav button changes to GPS1.
4) PFD Nav button– Press once and observe VLOC1 is displayed. Observe the GNS-430 No. 1 CDI button changes to VLOC.
5) PFD Nav button– Press once and note GPS2 is displayed. Observe the GNS-430 no. 2 CDI button displays GPS.
6) GNS-430 No. 2 CDI button– Press once and observe the CDI button label displays VLOC. Observe the PFD Nav button changes to VLOC2.
7) GNS-430 No. 2 CDI button– Press once and observe the CDI button label displays GPS. Observe the PFD Nav button changes to GPS2.
8) PFD Nav button– Press once and observe VLOC2 is displayed. Observe the GNS-430 No. 2 CDI button changes to VLOC.

NOTE

Two-way communication between the PFD and both GNS-430 units is demonstrated by the procedure above.

8. Pitot-Static Leak Check – Perform a pitot-static leak check, as described in section 34-2, anytime time the pitot-static ports are disconnected from the PFD.

e. **Primary Flight Display Testing and Operation –** Refer to the Avidyne FlightMax Entegra Primary Flight Display Installation Manual and FlightMax Entegra Series Pilot's Guide for testing and operation procedures.

34-23. AVIDYNE ENTEGRA EX5000 MULTI-FUNCTION DISPLAY (MFD)

a. **Multi-Function Display Removal –** The MFD is located in the right side of the instrument panel. See Figure 34 - 53 or Figure 34 - 54.
   1. Ensure both the master and avionics switches are in the off position.
   2. Loosen the 5/64" allen head screw fasteners at the four corners of the MFD. The Fasteners should not come out of the MFD, only loosen.
   3. Slide the unit out of the instrument panel.

4. Loosen the two jackscrews on the cable connector located on the back of the MFD and disconnect the cable.

b. **Multi-Function Display Installation –** Perform steps 1 – 3 above in reverse order. During installation ensure that the MFD cable is routed away from any moving parts behind the instrument panel.

c. **Troubleshooting Information** -The Avidyne MFD incorporates a message bar located at the bottom of the display. Messages are generated by the system and displayed on the message bar and are helpful in troubleshooting system problems. The following figures present the messages that are generated by each application.

Message	*Meaning*
Nav Source: Not Responding	No RS 232 or ARINC 429 GPS data is being received. Check wiring and Comm port setting.
Nav Source: Position Data Not Valid	Data is being received from the external GPS. However, insufficient information is available to determine position. Aircraft could be in the hanger or the GPS may not have determined its “fix” or location.
Nav Source: Position Data Valid	The MFD is receiving valid position data from the GPS.
Nav Source: Data Format Error	Data is being received, however the MFD does not recognize the data as the format selected. May indicate a baud rate or receiver type error.
Nav Source: No Port selected	The COMM port setting in the GPS setup dialog is set to NONE. To correct this, choose the port to which the GPS is connected, typically COM2.
Nav Source: Reconnecting	This message is displayed when data between the MFD and the GPS is being synchronized.
Nav Source: Can’t Open Port	Another device is configured for the same port. Check the Setup page for all devices. Typically the GPS is configured for RS 232 Port 2 or ARINC 429 port 1. A second GPS can only be configured for ARINC port 2.
Heading Data is NOT VALID	Heading data is no longer available from the GPS. Will only appear if the GPS is being used as the heading source.
Heading Data is VALID	Heading data has been restored.

Figure 34 - 49 GPS/FMS Messages

Message	*Meaning*
Traffic Sensor is OFF (TAS)	Traffic data is not being received.
Traffic Sensor is in Stand-By (TAS)	The Skywatch sensor has been placed in Standby mode.
Traffic Sensor is in Self-Test (TAS)	The Skywatch sensor has been placed in Self-Test mode.
Traffic Sensor is Operating Normally (TAS)	Verification that Traffic data is valid.
TCAD Altitude Unavailable (TCAD)	Occurs when altitude data has been lost from the TCAD sensor.

Figure 34 - 50 Traffic Messages

Message	*Meaning*
Lightning Sensor is Operating Normally	Verification that strike data to the MFD is valid.
Lightning Sensor in Demo Mode	Demo mode has been selected as the operating mode from the Lightning Setup Page.
Lightning Sensor is in Noise-Monitor Mode	Noise-Monitor mode has been selected as the operating mode from the Lightning Setup Page.
Lightning Sensor in Test Mode	Test mode has been selected as the operating mode from the Lightning Setup Page.
Lightning Sensor ERROR	The sensor system has reported an error that may mean current data is incomplete or erroneous. The error may clear.
Lightning Sensor has FAILED	The sensor system has reported an error that may mean current data is incomplete or erroneous. The error will not clear until power is removed from and reapplied to the sensor system.
Lightning Sensor is OFF	Communication of strike data from the Lightning sensor to the MFD has been lost.
Lightning Ahead	The Lightning Ahead option has been checked on the Lightning Setup page. Displayed when a Lightning Ahead condition exists.
Lightning Heading Source Failed	Heading data is no longer available from the WX-500. Strike data may still be valid. Will only appear if the WX-500 is being used as the heading source.
Lightning Heading Source OK	Heading data has been restored.
Stuck mic-PLEASE CHECK	Check the comm transmitters for indication of a stuck mike.
Verify Antenna Location (run Setup)	There may be an inconsistency between the antenna location jumper setting and the software configuration. This message should only appear during installation.

Figure 34 - 51 Lightning Messages

Message	*Meaning/Action*
Datalink Sensor Data Is Invalid	The EX5000 has received unreadable satellite data.
Datalink Sensor Configuration Error	The EX5000 is improperly configured for datalink. o Requires factory servicing. Refer to Factory Service Policies section of this manual.
Datalink Sensor Is NOT Communicating	The EX5000 is experiencing a communication failure with the internal Satellite transceiver. o Requires factory servicing. Refer to Factory Service Policies section of this manual.

Figure 34 - 52 Datalink Messages

d. **System Setup and Checkout –** Any time the Avidyne MFD is removed and sent to the factory for service, or is replaced with another unit, these system setup procedures should be performed to assure that the unit is properly configured for the installation. This section should also be referenced if it is suspected that the unit may be mis-configured.

Turn on the MFD by applying power to the aircraft battery and avionics bus master switches. The green bezel key backlighting should illuminate to full intensity when the Instrument Lights dimmer control is turned fully counterclockwise. As the dimmer control is turned clockwise, the bezel key lighting should dim fully, and then increase in intensity.

1. **Maintenance Mode Access –** Upon initial system power-up, the MFD will execute a start-up sequence. During this time, the screen will initially display the FlightMax logo, followed by a blank screen until a blue screen with the text "Initializing-Please Wait…" is displayed. The startup sequence is finished when the text "Press any bezel key to Continue…" is displayed. The initialization screen displays the loaded software version, along with NavData and Obstacle database expiration dates.

   Apply power to all sensors that interface with the MFD such as GPS navigator, lightning sensor, and traffic sensor. The MFD configuration is accomplished in the setup pages that are accessed through the Maintenance Mode, which is accessed as follows:

   a) Upon completion of system initialization, rotate the page knob to the *Setup* page.
   b) Simultaneously press and hold the top and middle left-hand bezel keys for at least 5 seconds until the maintenance page appears.

2. **GPS Interface, Setup and Checkout –** The GPS is interfaced to the MFD by either of two methods selected at the time of initial installation: ARINC 429 or RS 232. The ARINC 429 interconnect requires that a shielded pair connect the MFD to the GPS navigator. The RS 232 interconnect requires that a single shielded wire connect the MFD to the GPS navigator. Refer to the FlightMax EX5000/EX3000 Installation Manual, P/N 600-00073 and/or the FlightMax EX5000 Pilot's Guide, P/N 600-00102-000 for wiring diagram information. If it is not known which interface is utilized in a particular installation, select ARINC 429. If this selection is not correct, the message *Nav Source: Not Responding* will be displayed on the Map page after system start-up, in which case the RS 232 selection should be configured on the MFD.

NOTE

ARINC 429 is the recommended configuration.

From the Maintenance Page, press the *GPS Setup* bezel key to select the GPS Setup page.

a) ARINC 429 Setup – When the ARINC 429 interconnect is used, GPS 1 and GPS 2 are normally connected to the MFD. To select ARINC 429, highlight each configuration line and select the appropriate MFD settings as shown below:

Configuration Item	*MFD Setting*	*Comment*
Receiver	GAMA 429 Format	
Port	ARINC 1	Select ARINC 2 for GPS 2
Speed	Low	

Press the *Save* bezel key. The MFD must be restarted for the settings to take effect. Assure the Garmin GNS 400 series navigator is configured appropriately. To display the ARINC 429 Configuration Page, apply power to the GPS while pressing and holding the ENT key. Release the ENT key when the display activates. After the data base pages, the first page displayed is the MAIN ARINC 429 CONFIG page. While in the configuration mode, pages can be selected by ensuring the cursor is off and rotating the small right knob.

To change data on the displayed configuration page, press the small right knob (CRSR) to turn on the cursor. Turn the large right knob to change between data fields. Turn the large or small right knob to change a field that the cursor is on. Once the desired selection is made, press the ENT key to accept the entry.

The Garmin GNS 400 series navigator should be configured as follows:

Configuration Item	*Speed*	*Data*
IN 1	As Required	As Required
IN 2	As Required	As Required
OUT	Low	GAMA 429 Graphics w/Int

b) RS 232 Setup – When the RS 232 interconnect is used, only GPS 1 is connected to the MFD. To select RS 232, highlight each configuration line and select the appropriate MFD settings as shown below:

Configuration Item	*MFD Setting*
Receiver	King/Aviation Format
Port	COM 2
Baud	9600

Press the *Save* bezel key. The MFD must be restarted for the settings to take effect.

Assure that the Garmin GNS 400 series navigator is configured appropriately. Enter the configuration mode as described in section 34-23.d.2(a) and display the

MAIN RS232 CONFIG Page. Verify that the Channel 1 RS 232 output is set to *Aviation*.

c) GPS Checkout – After verifying the MFD GPS settings, the aircraft should be located where the GPS receiver can receive valid GPS satellite signals and generate a valid position. After the MFD has been restarted, the message *Nav Source: Position Data Valid* and the current aircraft position should be depicted on the MFD Map display. Refer to Section 34-23.c *Troubleshooting Information* for resolution of any error messages.

3. **Lightning Sensor Interface and Setup –** The WX-500 Stormscope lightning sensor interface is optional. Interconnect details are provided in the FlightMax EX5000/EX3000 Installation Manual, P/N 600-00073 and/or the FlightMax EX5000 Pilot's Guide, P/N 600-00102-000. The MFD communicates to and from the WX-500 using RS 232 communications.

a) Lightning Sensor Setup – To configure Lightning, access Maintenance Mode and select *Lightning Setup*. Highlight each configuration item and select the appropriate MFD settings as shown below:

Configuration Item	*MFD Setting*	*Comment*
Sensor	WX-500	
Operating Mode	Weather	
Port	COM1	
Stab Type	Synchro	Enter *GPS* if no other stabilization is provided.
Enable Stabilization?	Check	Leave unchecked if no stabilization is provided.
Enable Lightning Ahead Warning?	Check	Leave unchecked if not desired.
Antenna on Top?	Check	Leave unchecked if WX-500 antenna is mounted on bottom.

Press the *Save* bezel key. The MFD must be restarted for the settings to take effect.

b) Lightning Sensor Checkout – After the MFD has been restarted, the message *Lightning Sensor is Operating Normally* should be displayed in the message bar. If not, refer to Section 34-23.c, *Troubleshooting Information,* for resolution of any error messages.

Select the Map display and select *Heading/Track Up*. Select the *Strike* Lightning Mode. From the MFD *Setup* page, push the *Lightning Test* bezel key. Verify that a test strike is depicted on the Map display at approximately 18 NM at 45 degrees relative bearing to the aircraft nose.

Refer to Section 34-23.c *Troubleshooting Information* for resolution of any error messages.

4. **Traffic Sensor Interface, Setup and Checkout –** The SkyWatch traffic sensor interface is optional. Interconnect details are provided in the FlightMax EX5000/EX3000 Installation Manual, P/N 600-00073 and/or the FlightMax EX5000 Pilot's Guide, P/N 600-00102-000. The MFD uses an ARINC 429 data connection to receive data from the SkyWatch TRC 497. The MFD provides mode control of the SkyWatch through discrete signals to the Softkey 1 and Softkey 4 inputs on the TRC 497.

   a) Traffic Setup – To configure SkyWatch, access Maintenance Mode as described in Section 34-23.d.1 and select *Traffic Setup*. Highlight each configuration item and select the appropriate MFD settings as shown below:

Configuration Item	*MFD Setting*	*Comment*
Sensor	TAS	
Port	ARINC3	
External Controller?	Unchecked	
Heading Available?	Unchecked	Check if heading data is supplied to the SkyWatch and is available to the MFD.

Press the *Save* bezel key. The MFD must be restarted for the settings to take effect.

   b) Traffic Sensor Checkout – After the MFD has been restarted, the message *Traffic Sensor is Operating Normally* should be displayed in the message bar. If not, refer to Section 34-23.c, *Troubleshooting Information,* for resolution of any error messages.

Select the Map display and select *Heading/Track Up*. Select the *Normal* Traffic mode. From the MFD *Setup* page, push the *Traffic Test* bezel key. Verify that the traffic self-test pattern is depicted on the Map display as follows:

Symbol Type	*Relative Bearing*	*Approx Range*	*Relative Altitude*
Other	30°	3.5 nm	1000 Feet Below
Other	330°	3.6 nm	1000 Feet Above
TA	270°	2 nm	200 Feet Below

The message *Traffic Sensor is in Self-Test* should appear in the message bar.

Refer to Section 34-23.c *Troubleshooting Information* for resolution of any error messages.

5. **Map Setup –** The Avidyne MFD has the capability to overlay Traffic and Lightning information onto the Map display. To utilize this feature, Map requires Heading or Track information.

The MFD can receive HEADING data from one of following sources:

a) Garmin 400 series GPS (via ARINC 429).

b) BF Goodrich Stormscope (via RS 232). This is the most common method.

c) BF Goodrich Skywatch (via ARINC 429) with software level 1.6 or higher.

d) TRACK is the actual direction the aircraft is moving relative to the earth's surface, and is obtained from the GPS.

To configure the Map heading source, access Maintenance Mode as described in Section 34-23.d.1 and select *Map Setup*. Use the right hand knob to cycle through the selections: None (Use GPS Track), FMS/GPS, Stormscope, and SkyWatch. When the appropriate setting is selected, press the *Save* bezel key. The MFD must be restarted for the setting to take effect.

6. **Datalink Setup –** The EX5000 is pre-configured for datalink operation. Accessing the Datalink Info page from the System Info utility, which is entered from the Maintenance Mode menu, gives details of the configuration. Verify the presence of SC Serial Number and version information on this screen.

7. **Datalink Reception Confirmation –** ORCOMM satellite network transmits very low power VHF signals that the EX5000 must receive. If the antenna is not properly installed or if there is excessive electromagnetic interference (such as by nearby radio transmitter or inadequately grounded electronics) the system will not achieve successful reception. The following steps may be taken to assess system performance.

   a) Bring the aircraft to an area that has as few obstacles to line-of-sight viewing to the southern horizon as possible and orient the aircraft heading south.

   b) From Datalink Info table monitor for up to five minutes under the following conditions; other avionics off, avionics on, engines at idle, engines at takeoff power. With a satellite in view the signal strength and quality should peak above the following values.

      - "Signal strength" is greater than -118 dBm
      - "Estimated SNR (EbNo)" is greater than 10 dB
      - "Segment errors: less than 10%

   If the above reception levels are not achieved the following cause and corrective actions may apply.

      - The antenna field of view is obstructed. Try moving or rotating the aircraft. (if rotation works the antenna location on the aircraft may not be optimal.)
      - There is a local source of electromagnetic interference.
      - Try shutting off any nearby noise sources (such as VHF radios, alternators and magnetos). Once identified, have an authorized service center check wiring, shield terminations and component chassis grounding of the offending system.
      - Relocated the aircraft away from potential nearby sources.

- Check electrical connections to ensure there is no improper grounding.
- There is poor satellite coverage. Try again after fifteen minutes.
- The antenna and cable are not properly installed.
- The antenna may not be properly connected to the ground plane.
- The cable may not be properly connected to the antenna or EX5000.

Consult www.avidyne.com for any additional guidance.

e. **Maintenance –** Other than the periodic servicing task discussed below, maintenance of the MFD is based on condition only; no other periodic maintenance is required.

1. **Recommended periodic scheduled servicing tasks –** The MFD contains a 3-volt lithium battery that maintains CMOS memory on an internal processor board and should be replaced after 10 years of service, or when CMOS memory fails to retain configuration data, whichever occurs first. The Avidyne MFD must be returned to an authorized FAA repair station to perform this maintenance function. Failure of the CMOS memory is indicated by the message "WARNING: CMOS battery failure. Check database expiration date. Press any bezel key to continue" on a blue background prior to system boot up. After system boot up, the MFD will function normally, but will not have retained the system date and time. If the system is interfaced to the GNS 430 through ARINC 429, it will acquire current date and time from the GNS 430 when valid satellite data is received.

f. **Multi-Function Display Testing and Operation –** Refer to the Avidyne FlightMax Entegra Multi-Function Display Installation Manual and FlightMax Entegra Series Pilot's Guide for testing and operation procedures.

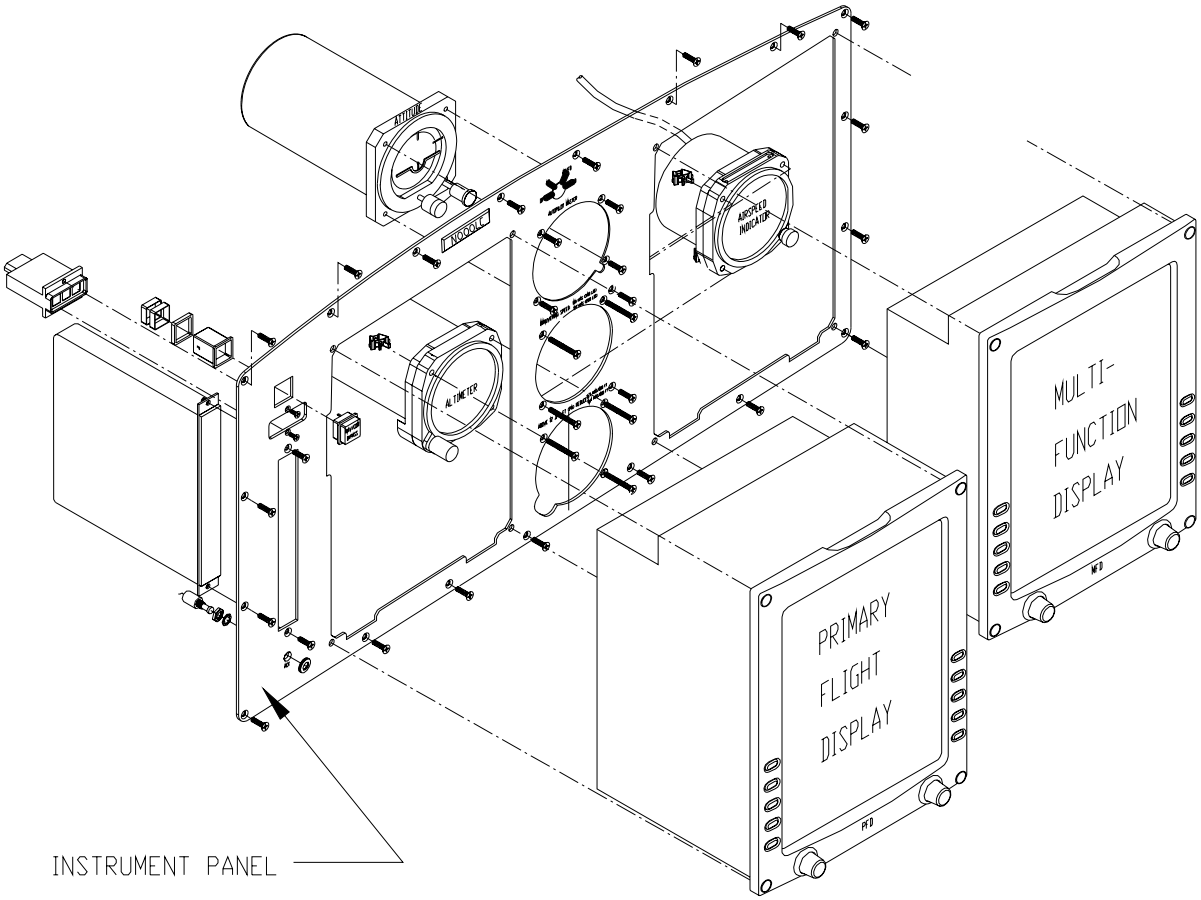


Figure 34 - 53 Avidyne Entegra Primary Flight Display and EX5000 Multi-Flight Display Installations

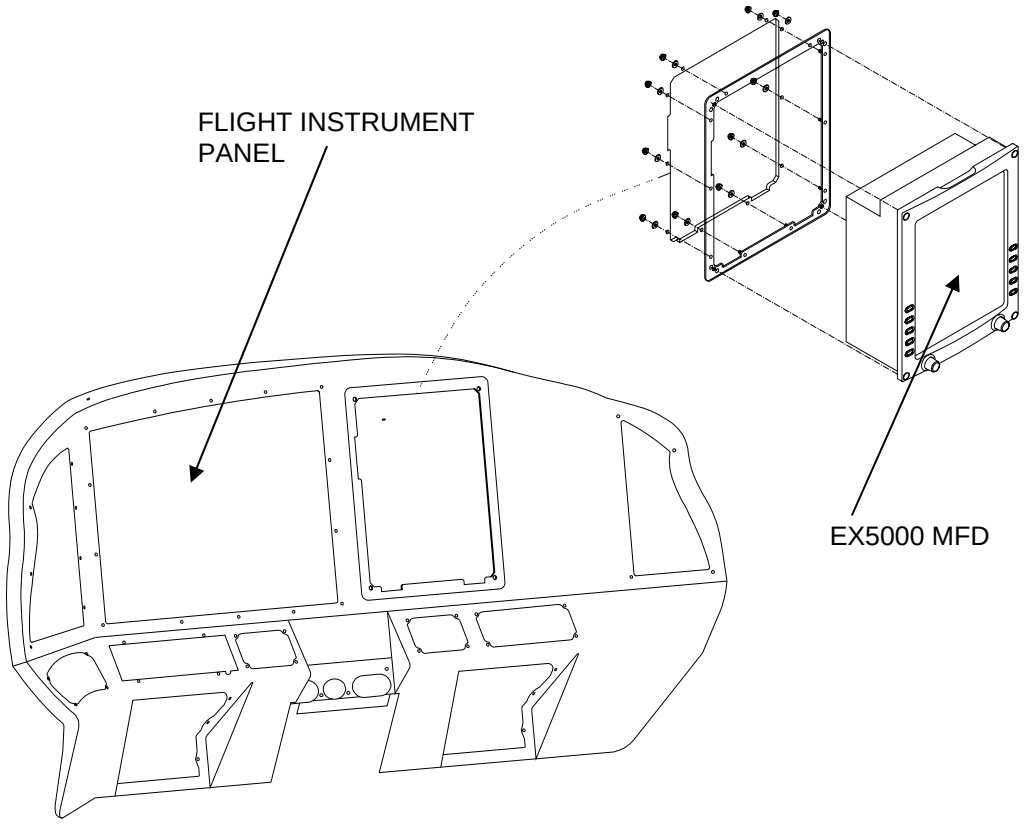


Figure 34 - 54 Avidyne Entegra EX5000 Multi-Flight Display Installation Without PFD Option.

34-24. AVIDYNE MAGNETOMETER/OAT SENSOR ASSEMBLY

a. **Magnetometer/OAT Sensor Assembly Removal**

1. Ensure that the PFD power and avionics master switches are in the off position.
2. Pull the PFD circuit breakers.
3. Locate the magnetometer access panel located underneath the left wing, approximately 4 feet from the wing tip. This assembly replaces the KMT 112 slaving transmitter (see Figure 34 - 40).
4. Remove the OAT probe nut from the mounting hole located in the magnetometer access panel.
5. Remove the magnetometer access panel.
6. Remove the wire tie and cable tie securing the magnetometer connector assembly.
7. Disconnect the connector assembly.
8. Remove the nuts and washers securing the magnetometer to the aircraft mounted standoffs and remove the magnetometer/OAT sensor assembly from the aircraft.

b. **Magnetometer/OAT Sensor Assembly Installation –** Perform steps 1 – 8 above in reverse order. Tighten nuts snug by removing all looseness and add 1/4 turn more.

CAUTION

Ensure the magnetometer is installed with the arrow and letters "FWD" pointing forward.

c. **Magnetometer/OAT Sensor Calibration Procedures –** See Section 34-22.d.

34-25. AVIDYNE ENGINE INSTRUMENT SENSOR INTERFACE UNIT (SIU)

a. **Sensor Interface Unit Removal –** The SIU is located on the avionics shelf in the avionics bay under the baggage compartment floor. See Figure 34 - 55
   1. Ensure both the master and avionics switches are in the off position
   2. Remove the carpet and access panel from the baggage compartment floor.
   3. Disconnect connectors J900 and J901 from the unit.
   4. Remove the four mounting screws that hold the SIU to the instrument shelf. See Figure 34 - 56.
   5. Remove the unit and the four rubber isolators.

b. **Sensor Interface Unit Installation**
   1. Installation is a direct reversal of removal. Hand tighten screws snug.
   2. Ensure connectors J900 and J901 are connected to the appropriate socket on the SIU.
   3. Install the access panel.
   4. Replace the carpet.

c. **SIU Checkout –** Use the MFD to checkout the SIU.
   1. Apply power to the SIU and MFD. Upon initial system power-up, the MFD will execute a start-up sequence. During this time, the screen will initially display the FlightMax logo, followed by a blank screen until a blue screen with the text "Initializing-Please Wait…" is displayed. The startup sequence is finished when the text "Press any bezel key to Continue…" is displayed. The initialization screen displays the loaded software version, along with NavData and Obstacle database expiration dates.
   2. Press any bezel key. The Initial Fuel page is displayed next.
   3. Press the Fuel Done button to exit.

d. **Avidyne Entegra Multi-Function Display (MFD) Configuration –** Refer to the Avidyne EX5000 Installation Manual. The MFD configuration is accomplished in the setup pages that are accessed through the Maintenance Mode, which is accessed as follows:
   1. Upon completion of system initialization, rotate the page knob to the Setup page.
   2. Simultaneously press and hold the top and middle left-hand bezel keys for at least 5 seconds until the maintenance page appears.

Configuration Item	*Selections*
Sensor Type	Not installed, Avidyne SIU
Port	Com 1, Com 2, Com *, Com 4* * Default port
Aircraft Model	Cessna 350

3. Depress bezel key next to the Sensor Type legend until "Avidyne SIU" is displayed
4. Press the Save bezel key. The MFD must be restarted for the settings to take effect.

e. **Engine Sensor Checkout –** Refer to the Avidyne EX5000 Installation Manual and EX5000 Pilot's Guide for testing and operation procedures.

1. After the MFD has been restarted, view the Engine page to verify data communication.
2. Operate the aircraft engine to confirm correct operation of all sensors. The following messages are generated by the Engine application:

Message	*Meaning*
Engine Sensor Unit Not Operating	No RS 232 Engine Sensor Unit data is being received by the MFD.
Engine Sensor Unit Operating Normally	Verification that Engine Sensor Unit data is valid.

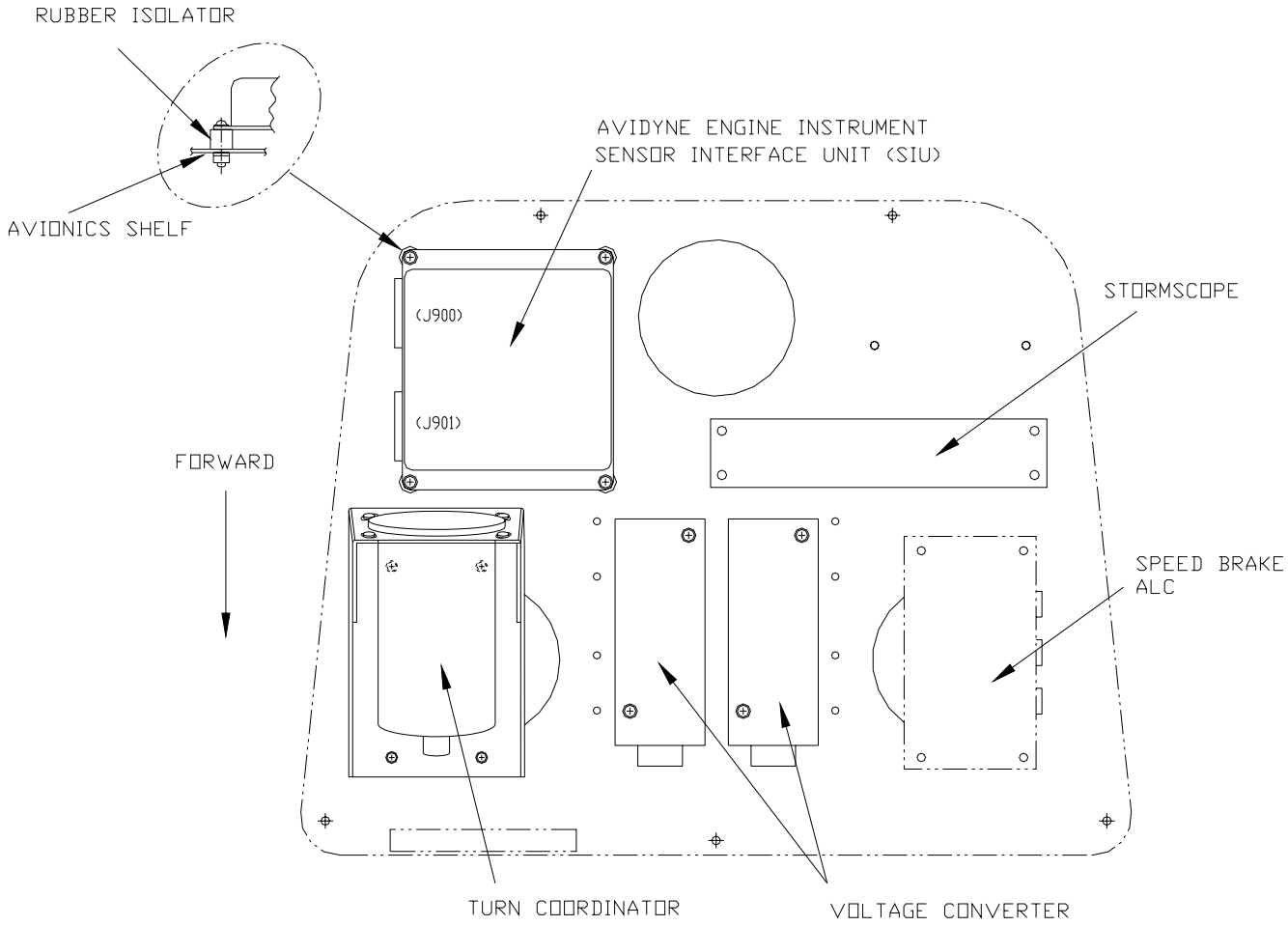


Figure 34 - 55 Avionics Shelf Configuration for Avidyne Entegra Equipment

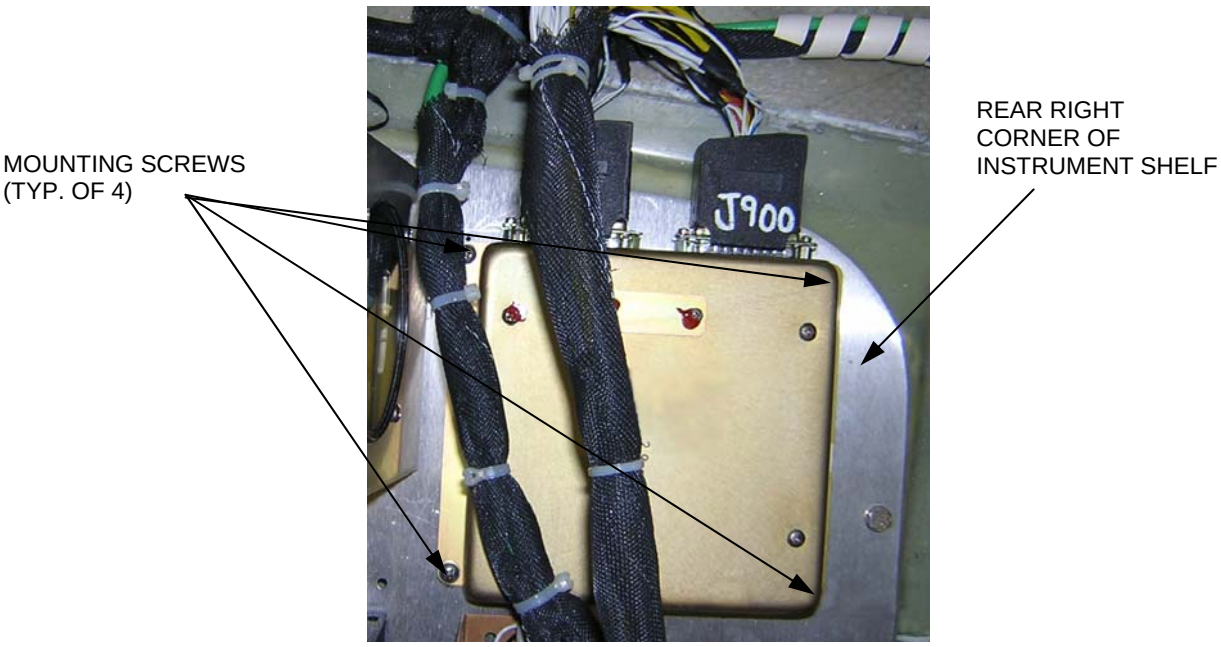


Figure 34 - 56 Engine Instrument Sensor Interface Unit

34-26. VOLTAGE CONVERTERS FOR AVIDYNE ENTEGRA PFD – The voltage converters are located on the avionics shelf in the avionics bay under the baggage compartment floor. The voltage converters are only installed in aircraft S/N 42001 to 42500; they are not required on aircraft S/N 42501 and on. See Figure 34 - 55.

a. **Voltage Converter Removal**
   1. Ensure both the master and avionics switches are in the off position
   2. Remove the carpet and access panel from the baggage compartment floor.
   3. If the XM weather receiver is installed, remove per 0.
   4. Unplug the converter.
   5. If applicable, remove all four voltage converter mounting screws and the XM weather receiver support bracket, otherwise remove the two voltage converter mounting screws.
   6. Remove the converter(s).

b. **Voltage Converter Installation –** Installation is a direct reversal of removal. Hand tighten screws snug.

34-27. VOLTAGE CONVERTER FOR AVIDYNE ENTEGRA EX5000 MFD – The voltage converter is located inside the radio rack at the base. See Figure 34 - 57. The voltage converter is only installed in aircraft S/N 41002 to 41500; it is not required on aircraft S/N 41501 and on.

a. **Voltage Converter Removal**
   1. Ensure both the master and avionics switches are in the off position.
   2. Remove the center console per instructions in Chapter 25 to gain access to the radio rack.
   3. Remove the connector.

4. Remove the four hex head bolts attaching the power converter plate to the bottom of the radio rack and the floor.
5. Lift out the converter.
6. Remove heat shrink material (if present) on the nuts and washers and remove the two 8-32 x 2.5 in. screws holding the voltage converter to the mounting plate.

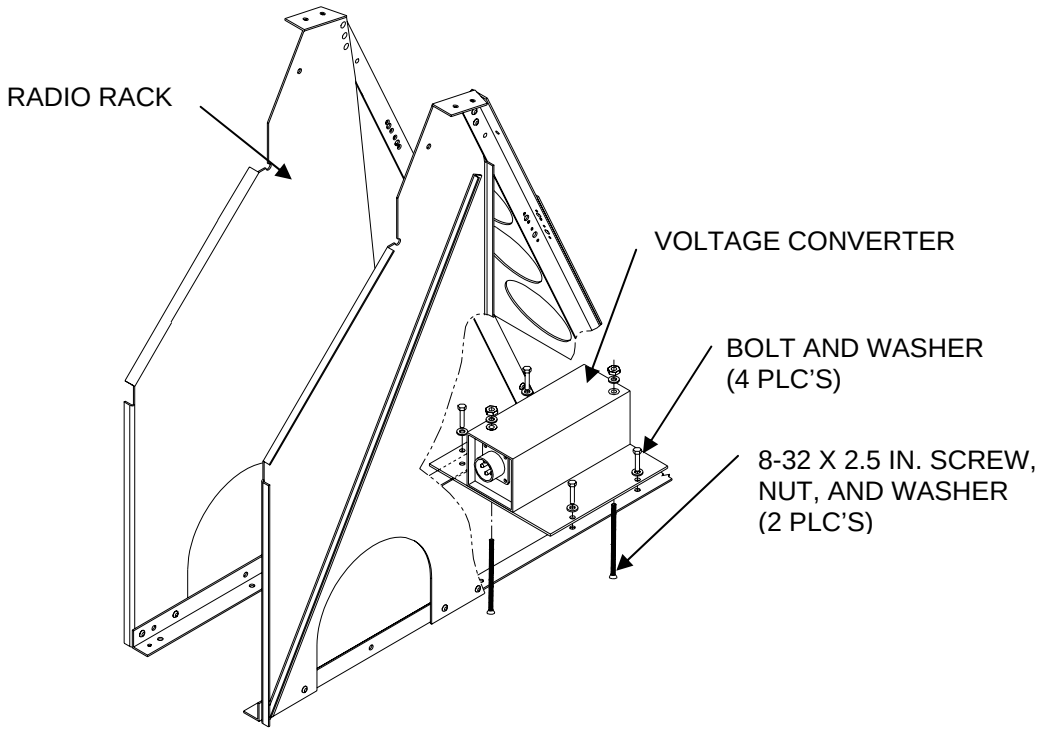


Figure 34 - 57 EX5000 Voltage Converter Installation (S/N 42001 to 42500)

b. **Multi-Function Display Power Converter Installation**

1. Perform steps 1-6 above in reverse order.
2. Torque the 8-32 x 2.5 in. screws 20 to 25 in.-lbs. Apply new heat shrink material.
3. Torque the four mounting plate bolts 30 to 36 in.-lbs.

34-28. GARMIN G1000 SYSTEM – This section describes how to remove and replace G1000 equipment in the Cessna 350. After removal and replacement, LRUs must be configured and tested per the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

CAUTION

When removing and/or replacing any G1000 component, always ensure that aircraft power is off. Unplug any auxiliary power supplies.

Before removing any G1000 LRU, it is required that the technician verify the LRU software part number and version against the Required Equipment List RB011002.

To check an LRU software part number and/or version:

1. Start the G1000 system in configuration mode as described in Section 3.4 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.
2. The System Status page shows a list of LRUs in the LRU window. Activate the cursor and highlight the LRU window.
3. Use the FMS knob to scroll through the list in the window and select the desired LRU.
4. The software part number and version is displayed in the DATA window. Compare this to the data in the Required Equipment List.

NOTE

If a faulty LRU is not reporting its software version and part number, check aircraft maintenance logs for last software version loaded and verify against the Required Equipment List. The Software Manifest page may also be used to check part numbers and versions.

Figure 34 - 58 System Status Page (Configuration Mode)

a. **TROUBLESHOOTING**

For troubleshooting procedures for the Garmin G1000 System see the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

b. **GDU 1040 PFD and GDU 1042/GDU 1044 MFD**

Two Garmin GDUs are installed in the instrument panel. The PFD is a GDU 1040 and the MFD is a GDU 1042/GDU 1044 (required for Wide Area Augmentation System (WAAS) utilization). Both displays provide control and display of nearly all functions of the G1000 integrated cockpit system. The displays are located side-by-side, with the GMA 1347 Audio Panel located in-between.

Both displays are installed in the instrument panel using built-in 1/4-turn fasteners. Each display uses a single Garmin 62-pin connector. Electrical power to the PFD is from the essential bus, whereas the MFD receives power from the Avionics bus.

On the panel, traditional attitude, altitude, and airspeed indicators are provided as backup instruments. These are mounted in a vertical configuration to the left of the PFD.

(GDU 1040 PFD shown)

Figure 34 - 59 GDU 1040 PFD/GDU 1042 MFD/GDU 1044 MFD

1. **Removal**
   a) Using a 3/32nd hex tool, rotate all four 1/4-turn fasteners counter-clockwise until they reach their stops.
   b) Carefully remove the display from the panel.
   c) While supporting the display, disconnect the connector.
2. **Replacement**
   a) Visually inspect the connector and pins for signs of damage. Repair any damage. While supporting the display, connect the connector to the rear of the unit.
   b) Carefully insert the display into the panel cutout, ensuring that all four 1/4-turn fasteners align with the corresponding holes.
   c) Seat the display in the panel cutout. Do not use excessive force while inserting the display.
   d) Once seated, rotate all four 1/4-turn fasteners clockwise to lock the display to the panel.
   e) Load software, configure, and test the MFD and/or PFD according to Section 6.1 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

c. GMA 1347 Audio Panel

The Garmin GMA 1347 Audio Panel is a digital audio panel with integrated marker beacon receiver. The GMA 1347 provides control of all cockpit intercom/mic systems as well as NAV/COM/ILS audio. The unit also provides display reversion mode control through a large red button. Power is received from the Avionics bus; consequently the unit only powers up when the avionics master switch is turned on. The GMA 1347 interfaces with the existing marker beacon antenna as well as existing mic and phone jacks.

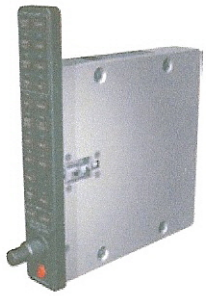


Figure 34 - 60 GMA 1347 AUDIO PANEL

1. **Removal**
   a) Using a 3/32nd hex tool, turn the hex nut counter-clockwise until the GMA 1347 is unlocked from its location.
   b) Carefully remove the GMA 1347 from its rack.
2. **Replacement**
   a) Visually inspect the connectors using a flashlight to ensure there are no bent or damaged pins. Repair any damage.
   b) Gently insert the GMA 1347 into the rack until the locking tab engages the rack.
   c) Begin to turn the hex nut clockwise. This draws the unit into the rack until seated. Do not over-tighten the nut.
   d) Load software, configure, and test the GMA 1347 according to Section 6.2 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

d. GIA 63/GIA 63W INTEGRATED AVIONICS UNIT

Two Garmin GIA 63/GIA 63W IAUs provide VHF COM, VHF NAV, GPS NAV and other various navigation functions, as well as control logic for the Garmin GFC 700 autopilot. GIAs provide a communication interface to all other G1000 LRUs in the system. The GIA 63W is required for Wide Area Augmentation System (WAAS) utilization. Only one type of GIA either GIA 63 or GIA 63W will be installed. Both GIAs are located directly behind the MFD and installed into their respective LRU racks. The #1 GIA is powered through the essential bus and thus immediately powers up when the left battery or right battery master switch is turned on. The #2 GIA receives power through the Avionics bus and powers up when the avionics master switch is turned on. Both GIA 63/GIA 63Ws interface to the following equipment:

- .... VOR/LOC/Glideslope Antenna System
- .... VHF COM 1 & 2 Antennas

- GPS Antennas
  WAAS: GA 35 (#1 GIA only); GA 37 (#2 GIA only)
  Non WAAS: Two GA 56 (one for #1 GIA, the other for #2 GIA)
- WX-500 StormScope, if installed (#2 GIA Only)
- GFC 700 Autopilot
- Ryan TCAD, if installed (#2 GIA Only)
- CO Detector (#2 GIA Only)

Figure 34 - 61 GIA 63/GIA 63W Integrated Avionics Unit

1. **Removal**
   a) Remove the MFD per paragraph 34-28.a.
   b) Unlock the GIA 63/GIA 63W handle by loosening the Phillips screw on the handle.
   c) Pull the handle upward to unlock the GIA 63/GIA 63W. Gently remove the unit from the rack.
2. **Replacement**
   a) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
   b) Gently insert the GIA 63/GIA 63W into its rack. The handle should engage the dogleg track.
   c) Press down on the GIA 63/GIA 63W handle to lock the unit into the rack.
   d) Lock the handle to the GIA 63/GIA 63W body using the Phillips screw.
   e) Reinstall the MFD per paragraph 34-28.a.
   f) Load software, configure, and test the GIA 63/GIA 63W(s) according to Section 6.3 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

e. GEA 71 ENGINE/AIRFRAME UNIT

The Garmin GEA 71 Engine/Airframe Unit provides engine/airframe data to the G1000 system. Data received from transducers/sensors is processed, then sent via RS-485 to a GIA 63/GIA 63W, and subsequently to the GDU 1042/GDU 1044 MFD. The GEA 71 is connected to both GIA 63/GIA 63Ws for redundant communications. In display reversionary mode, engine instrumentation is displayed on the PFD as well. The GEA is located directly behind the PFD in its LRU rack. Power is received from the essential bus.

The GEA interfaces to the following:

- Manifold Pressure Sensor
- Tachometer Sensor
- Oil Pressure Sensor And Low Pressure Switch
- Oil Temperature Sensor
- Cylinder Head Temperature (CHT) Sensors
- Exhaust Gas Temperature (EGT) Sensors
- Turbine Inlet Temperature (TIT) Sensors *(Cessna 400 only)*
- Fuel Level Sensors (Four)
- Low Fuel Switches (L And R)
- Fuel Selector Position Switches
- Fuel Flow Sensor
- Electrical System Sensors And Alternator Inop Switches
- Open Door Detection Switch
- Aileron And Elevator Position Sensors
- Oxygen System Sensors And Switches
- Starter Engage System

Figure 34 - 62 GEA 71 Engine/Airframe Unit

1. **Removal**
   a) Remove the PFD per paragraph 34-28.a.
   b) Unlock the GEA 71 handle by unscrewing the Phillips screw.
   c) Pull the handle upward to unlock the GEA 71.
   d) Gently remove the GEA 71 from its rack.
2. **Replacement**
   a) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
   b) Gently insert the GEA 71 into the rack. The handle should engage the dogleg track.
   c) Press down on the handle to lock the unit into place.
   d) Lock the handle to the GEA 71 body using the Phillips screw.
   e) Reinstall the PFD per paragraph 34-28.a.
   f) Load software, configure, and test the GEA 71 according to Section 6.4 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

f. GTX 33 MODE S TRANSPONDER

The Garmin GTX 33 provides Mode A, C, and S altitude and position reporting information from the G1000 system. The unit is mounted directly behind the PFD in its LRU rack. Power is received from the Avionics Bus. The GTX 33 sends data via RS-232 directly to a GIA 63/GIA 63W. Information is then sent to the PFD, where the pilot can control the transponder. The GTX 33 is connected to both GIA 63/GIA 63Ws for redundant communications. The GTX 33 interfaces with the existing transponder antenna.

Test according to Title 14 CFR §§ 91.411 and 91.413 as well as Part 43 Appendix F.

Figure 34 - 63 GTX 33 Transponder

1. **Removal**
   a) Remove the PFD per paragraph 34-28.a.
   b) Unlock the GTX 33 handle by loosening the Phillips screw on the handle.
   c) Pull the handle upward to unlock the GTX 33. Gently remove the unit from the rack.
2. **Replacement**
   a) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
   b) Gently insert the GTX 33 into its rack. The handle should engage the dogleg track.
   c) Press down on the GTX 33 hand to lock the unit into the rack.
   d) Lock the handle to the GTX 33 body using the Phillips screw.
   e) Reinstall the PFD per paragraph 34-28.a.
   f) Load software, configure, and test the GTX 33 according to Section 6.4 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

g. GDC 74A AIR DATA COMPUTER

The Garmin GDC 74A provides digital air data to the G1000 system. The unit is mounted to its rack directly behind the PFD. Power is received from the Essential Bus. The GDC 74A connects to existing pitot/static ports. Air data is sent to the PFD for display. Four redundant data paths exist to the PFD, MFD, and both GIAs.

Test according to Title 14 CFR §§ 91.411 and 91.413 as well as Part 43 Appendix F. The static system and altimeter must be verified every 2 years in accordance with 14 CFR §§

91.411 and Part 43 Appendix E. Testing procedures per Section 6.6 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

Figure 34 - 64 GDC 74A Air Data Computer

1. **Removal**
   a) Remove the PFD per paragraph 34-28.a.
   b) Disconnect the pitot/static plumbing from the rear of the unit. Disconnect the single connector.
   c) Loosen each thumbscrew on the hold-down clamp and remove the clamp.
   d) Carefully remove the unit from its mount.
2. **Replacement**
   a) Place the unit in the mounting tray.
   b) Position the locking clamp and fasten using the thumbscrews.
   c) Connect the pitot/static plumbing.
   d) Inspect the connector and pins for damage. Repair any damage. Connect the connector to the unit.
   e) Reinstall the PFD per paragraph 34-28.a.
   f) Load software, configure, and test the GDC 74A according to Section 6.6 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

h. GTP 59 OAT PROBE

The Garmin GTP 59 OAT Probe provides the GDC 74A with air temperature data. The OAT probe is mounted to the bottom side of the left wing.

1. **Removal**
   a) Remove the OAT probe cover assembly from the underside of the left wing by removing the Phillips screws.
   b) Disconnect the connector.
   c) Use a deep-socket to hold the probe in place on the outside of the cover. On the inside of the cover, loosen the GTP 59 mounting nut and remove the GTP 59.
2. **Replacement**
   a) Replacement is the reverse of removal.
   b) No configuration is required for the GTP 59. Test according to Section 6.6.3 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

i. GRS 77 ATTITUDE AND HEADING REFERENCE SYSTEM (AHRS)

The Garmin GRS 77 AHRS provides attitude and heading information to the G1000 system. The unit is mounted on the center of the cockpit floor just aft of the firewall. Power is received from the Essential Bus. The GRS 77 interfaces with and provides power to the GMU 44 Magnetometer. The GRS 77 supplies attitude and heading information directly to the PFD, MFD, and to both GIAs.

The GRS 77 utilizes an Earth magnetic field model which is updated once every five years. The update is expected to be available from Garmin by July 1 of each of the following years: 2005, 2010, 2015, and every five years thereafter.

Figure 34 - 65 GRS 77 AHRS

1. **Removal**
   a) Remove the instrument panel tower per Chapter 25.
   b) Disconnect the AHRS connector.
   c) Remove the four Phillips thumbscrews with a screwdriver and set them aside.
   d) Gently lift the GRS 77 from the mounting plate. (If the supports for the mounting plate are removed, the GRS 77 must be re-calibrated. See Section 6.7.3 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03. If the mounting plate is removed and reinstalled in exactly the same position, the GRS 77 does not need to be recalibrated.)
2. **Replacement**
   a) Place the GRS 77 on the mounting plate, ensuring the orientation is correct.
   b) Fasten the unit to the plate using the Phillips thumbscrews.
   c) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage. Connect the connector to the GRS 77.
   d) Load software, configure, and test the GRS 77 according to Section 6.7 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.
   e) Reinstall the instrument panel tower per Chapter 25.

j. GMU 44 MAGNETOMETER

The GMU 44 provides horizontal and vertical magnetic field information to the GRS 77 AHRS. This allows heading to be calculated and provides assistance during AHRS alignment. The GMU 44 is mounted at the left wingtip.

Figure 34 - 66 GMU 44 Magnetometer

1. **Removal**
   a) Remove left wing access cover by unscrewing the Phillips screws.
   b) Unscrew the three screws, or remove the three self-locking nuts and washers that hold the GMU 44 to its mounting rack.
   c) Carefully lift the GMU 44 from the rack.
   d) Disconnect the wiring harness.
2. **Replacement**
   a) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage. Connect the wiring harness to the GMU 44.
   b) Lower the GMU 44 into the rack and secure with three Phillips screws or three self-locking nuts and washers, as applicable. Tighten screws and/ or nuts snug by removing all looseness and add 1/4 turn more.
   c) Reinstall left wing access cover.
   d) Load software, configure, and test the GMU 44 according to Section 6.7 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

k. GDL 69A DATA LINK RECEIVER

The GDL 69A is an XM Satellite Radio data link receiver with the addition of XM Satellite Radio audio entertainment. For display of weather information and control of audio channel and volume, the GDL 69A is interfaced to the GDU 1042/GDU 1044 via an Ethernet link. Audio volume and channel changes may also be controlled with remotely mounted switches located in the center console. The GDL 69A is also interfaced to a Garmin audio panel for amplification and distribution of the audio signal. The GA 55 XM Satellite Radio antenna receives the XM Satellite Radio data signal and passes it to the GDL 69A. On aircraft equipped for Wide Area Augmentation System (WAAS), the GA 55 is not installed and the XM Satellite Radio data signal is received by the GA 37 (combined GPS and XM satellite radio) antenna located on the antenna bay behind the hat rack.

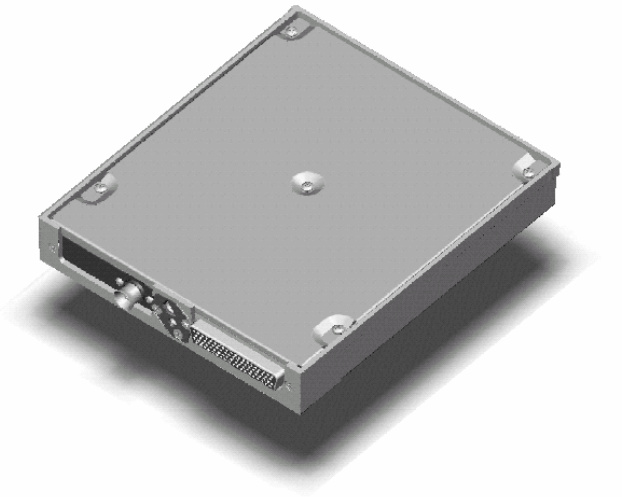


Figure 34 - 67 GDL 69A Data Link Receiver

1. **Removal**
   a) Remove the PFD per paragraph 34-28.a.
   b) Unlock the GDL 69A handle by unscrewing the Phillips screw.
   c) Pull the handle upward to unlock the GDL 69A.
   d) Gently remove the GDL 69A from its rack.
2. **Replacement**
   a) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
   b) Gently insert the GDL 69A into the rack. The handle should engage the dogleg track.
   c) Press down on the handle to lock the unit into place.
   d) Lock the handle to the GDL 69A body using the Phillips screw.
   e) Reinstall the PFD per paragraph 34-28.a.
   f) Load software, configure, and test the GDL 69A according to Section 6.8 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

l. GCU 476 REMOTE KEYPAD

The GCU 476 interfaces with the GDU 104x PFD/MFD. The GCU 476 Remote Keypad provides alphanumeric, softkey, and flight planning function keys used to interface with the G1000. In addition to alphanumeric, softkey, and flight planning function keys the GCU 476 provides COM/NAV tuning capabilities. The GCU 476 mounts on the center console using a single jackscrew.

Figure 34 - 68 GCU 476 Remote Keypad

1. **Removal:**
   a) Remove the setscrew securing the keypad with a 3/32nd hex tool and carefully remove keypad.
   b) While supporting the keypad disconnect the connector.
   c) Remove the unit.
2. **Replacement:**
   a) Connect the connector to the back of keypad.
   b) Gently place keypad into position.
   c) Secure with setscrew using a 3/32nd hex tool. Do not over-tighten.
   d) Reload configuration files and verify software version.

m. **GFC 700 AUTOPILOT SYSTEM –** See Chapter 22.

n. **GA 35 Antenna (WAAS)**

The GA 35 antenna is one of the components of the Garmin G1000 System that allows utilization of the Wide Area Augmentation System (WAAS). The antenna is mounted to a leveling wedge on top of the aircraft and forward of the vertical stabilizer at FS 185.51. The leveling wedge is permanently affixed to the aircraft.

1. **Antenna Removal**
   a) Remove the aft center overhead light panel (ACOLP) per instructions in Chapter 25.
   b) Disconnect the antenna cable from the connector on the base of the antenna. Keep the antenna connector clean with a plastic cover cap or other means of protection.
   c) Remove the RTV 162 white silicone sealant around the base of the antenna and covering the four screws attaching the antenna to the leveling wedge.
   d) Stow the braided wire/terminal from the antenna.
   e) Remove the antenna.
2. **Antenna Installation**
   a) Remove all old RTV-162 silicone from the leveling wedge to antenna mounting surface and apply a fresh layer.
   b) Unstow the braided wire/terminal.

c) Place the antenna on the wedge and install the mounting screws. Ensure the braided wire/terminal is aligned to the antenna mounting hole. Torque 17 to 21 in-lbs. Wipe away any excess RTV-162.
d) Fill the voids over the screw heads with RTV-162 and smooth it to the contours of the antenna.
e) Connect the antenna cable to the connector on the base of the antenna.
f) Install the aft center overhead light panel (ACOLP) per instructions in Chapter 25.

o. GA 37 Antenna (WAAS)

The GA 37 is a combined GPS and XM satellite radio antenna, and is one of the components of the Garmin G1000 System that allows utilization of the Wide Area Augmentation System (WAAS). The antenna is mounted on the antenna box behind the hat rack.

1. **Antenna Removal**
   a) Remove the aft baggage bulkhead upholstery plug and the cover plate.
   b) Disconnect the GPS antenna cable and the XM satellite radio data cable from the connectors on the base of the antenna. Keep the antenna connectors clean with a plastic cover cap or other means of protection.
   c) Remove the four screws, self-locking nuts, and washers attaching the antenna to the antenna box.
   d) Lift the antenna and the O-ring from the antenna box.
2. **Antenna Installation**
   a) Place the O-ring on the antenna.
   b) Place the antenna on the antenna box.
   c) Install the fours screws, self-locking nuts, and washers. Torque 28 to 34 in-lbs.
   d) Connect the GPS antenna cable and the XM satellite radio data cable.
   e) Replace the cover plate and aft baggage bulkhead upholstery plug.

p. **CONFIGURATION MODULE REMOVAL AND REPLACEMENT**

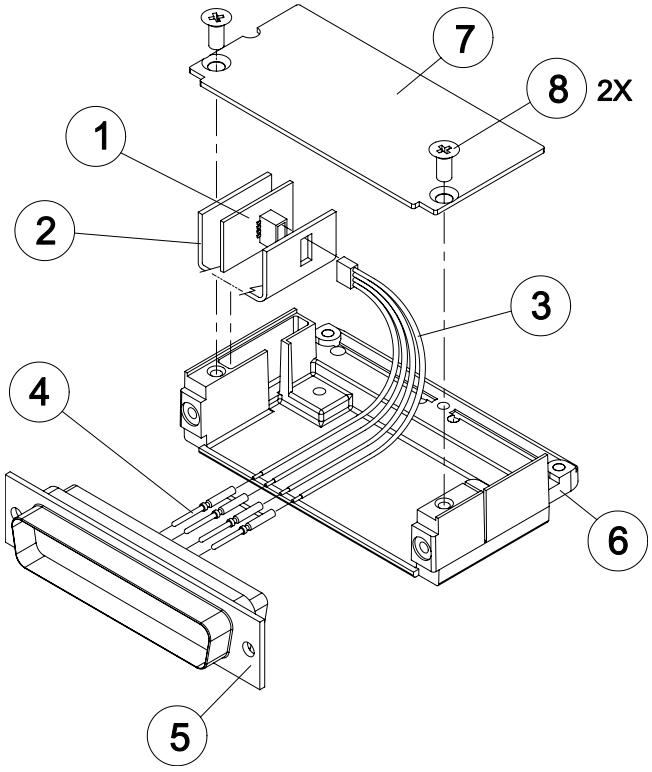


Figure 34 - 69 Configuration Module Installation

1. **Removal** (See Figure 34 - 69**)**
   a) Disconnect connector from LRU.
   b) Remove 2 screws (8) from cover (7) and remove cover.
   c) Unplug connector from configuration module (1).
   d) Remove configuration module.
2. **Replacement** (See Figure 34 - 69)
   a) Inspect connector for damaged pins (4).
   b) Place configuration module (1) in position.
   c) Insert connector into configuration module (1).
   d) Assembly of the connector is the reverse of disassembly.
3. **If the GRS 77 AHRS Configuration Module is replaced –** All three GRS 77/GMU 44 calibration procedures must be performed. See Section 6.6.3 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.
4. **If GDC 74 Configuration Module is replaced –** Configuration settings must be reloaded to the GDC 74A. See Section 6.5.2 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.
5. **If the Master Configuration Module is replaced**
   a) Start the G1000 system in configuration mode.
   b) Go to the Configuration Upload Page on the PFD.
   c) Press the UPDT CFG softkey.

6. **If both the PFD and Master Configuration Module are replaced –** The G1000 system (except GRS 77/GMU 44 and GDC 74A) must be re-configured. See Section 6.1.1 of the Garmin G1000 System Maintenance Manual, P/N 190-00577-03.

q. GEA 71 BACKSHELL THERMOCOUPLE REMOVAL AND REPLACEMENT

1. **Removal** (See Figure 34 - 70)
   a) Remove GEA 71 per paragraph 34-28.e.
   b) Remove GEA connector backplate.
   c) Remove connector J701 (5) from the backplate.
   d) Remove cover (6) from the backshell.
   e) Unscrew thermocouple from boss on backshell. Extract the thermocouple pins from the connector.
2. **Replacement** (See Figure 34 - 70 and Figure 34 - 71)
   a) Crimp pins (2) onto each of the thermocouple wires (1). Ensure that pre-stripped wire length is 1/8" prior to crimping.
   b) Insert newly crimped pins and wires into the appropriate connector housing location (4) as specified by the 350/400 avionics wiring diagram of Garmin G1000 System Maintenance Manual, P/N 190-00577-03.
   c) Place thermocouple body (1) onto the backshell boss (5). Place the thermocouple as shown in Figure 34 - 70 so that the wires exit towards the bottom of the backshell.
   d) Fasten thermocouple tightly to backshell using the provided screw (3).
   e) Fasten cover (6) to backshell using the provided screws (7).

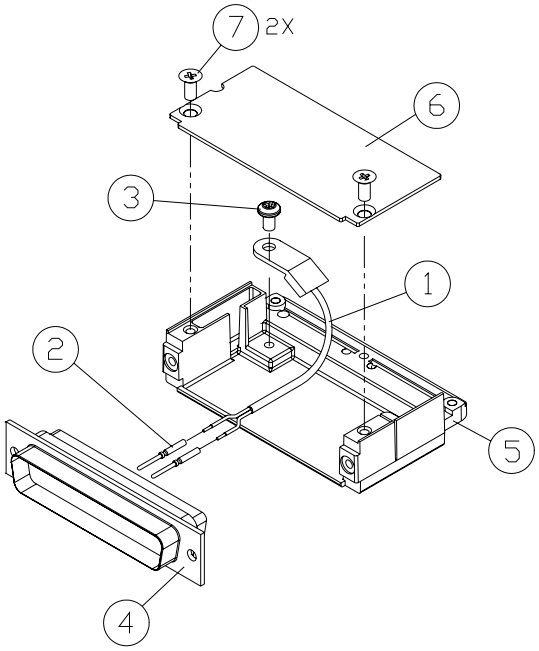


Figure 34 - 70 GEA Backshell Thermocouple

Item #	Description	Qty. Needed	Garmin Part Number
1	3” Thermocouple, K type	1	925-L0000-00
2	Pins #22 AWG	2	336-00021-00
3	Screw	1	211-60234-08

Figure 34 - 71 Thermocouple Kit (011-00981-00)

r. **INSPECTION**

1. **Annual**

a) Power up the aircraft by turning on the left & right batteries / alternators, start the engine and then turn on the Avionics Master Switch. – ensure the cross tie switch is OFF, A/P Trim Master ON
b) Wait approximately one minute for the system to initialize.
c) Verify the system obtains attitude and altitude, and no functions on the PFD or MFD are inoperative. Inoperative functions are “X” out in red.
d) Allow the system to acquire GPS position. Verify that both GIAs have GPS position via the AUX page.
e) Ensure the system does not generate any failures or alerts.
f) Verify all engine parameter and temperature sensors are operational.
g) Verify aircraft charging system is operation, both left and right bus.
h) Turn off the left alternator and verify the G1000 annunciates a failure of the left alternator and plays an aural message “Left Alternator Out”.
i) Monitor the full screen engine page on the MFD, turn off the left battery switch. Verify the left battery voltage drops off and the G1000 annunciates a failure of the left bus; restore the left battery and alternator switches to the on position.
j) Turn off the right alternator and verify the G1000 annunciates a failure of the right alternator and plays an aural message “Right Alternator Out”.
k) Monitor the full screen engine page on the MFD, turn off the right battery switch. Verify the right battery voltage drops off and the G1000 annunciates a failure of the right bus; restore the right battery and alternator switches to the on position.
l) Verify the selected full tank value position, agrees with the indicator on the MFD (full screen EIS, or engine strip). Switch fuel tanks, and verify the indication on the G1000 follows. Move the fuel selector between tanks momentarily and verify the G1000 annunciates “fuel valve” along with an aural warning.
m) Leave the avionics switch on, and turn off the engine with the mixture control (lean to cut-off)
n) Unlatch the pilot door, and verify the G1000 annunciates “door”, latch the pilot door, and repeat for the copilot door and the aft cargo door.
o) If speed breaks are installed actuate the speed breaks and verify the G1000 annunciates an advisory to indicate the speed breaks are deployed.
p) If the rudder hold option is installed actuate the hold function and verify the G1000 annunciates an advisory to indicate the rudder is being held.
q) Turn off the avionics switch, ensure the G1000 enters reversionary mode (composite engine / PFD)

r) Run the elevator trim to the full up position, and verify the trim indicates full up. Repeat for down.
s) Run the aileron trim to the full right position, and verify the indication. Repeat for left.
t) Turn off the trim master switch and verify elevator and aileron trim are disabled and G1000 annunciates a pitch trim failure.

2. **1000 Hour –** Perform a visual inspection every 1000 airframe hours in conjunction with the next scheduled inspection and check for corrosion, damage, or other defects for each of the items listed below. Replace any damaged parts as required. Inspection may require the temporary removal of a unit or units to gain access to connectors.
   a) **GDU 1040 PFD and GDU 1042/ GDU 1044 MFD** – Remove the PFD & MFD. Inspect the mounting surface and connector for corrosion, heavy oxidation, or other damage.
   b) **GMA 1347 Audio Panel** – Inspect the GMA 1347 unit, rack, and connectors for corrosion or other defects.
   c) **GIA 63/GIA 63W Integrated Avionics Unit (Qty 2)** – Inspect the GIA 63/GIA 63W #1 and #2 unit, rack, and connectors for corrosion or other defects.
   d) **GEA 71 Engine/Airframe Unit** – Inspect the GEA 71 unit, rack, and connectors for corrosion or other defects.
   e) **GTX 33 Mode S Transponder** – Inspect the GTX 33 unit, rack, and connectors for corrosion or other defects.
   f) **GDL 69A XM Radio Data Link Receiver** – Inspect the GDL 69A unit, rack, and connectors for corrosion or other defects.
   g) **GRS 77 AHRS** – Inspect the GRS 77 unit, rack, and connector for corrosion or other defects.

34-29. WX-500 STORMSCOPE PROCESSOR – The WX-500 Stormscope Processor is located on the avionics shelf under the baggage compartment floor; it's location on the avionics shelf depends on other equipment installed on the aircraft.

a. WX-500 Stormscope Processor and Tray Removal (see Figure 34 - 72 or Figure 34 - 73)
   1. Ensure both the master and avionics switches are in the off position.
   2. Remove the carpet and access panel from the baggage compartment floor.
   3. Disconnect the wiring connectors.
   4. Remove the locking nut on front of the mounting tray.
   5. Slide out the stormscope from the mounting tray.
   6. Stormscope Mounting Tray Removal (if required)
      a) S/N 42001 to 42500) – Remove the four 10-32 screws and spacers attaching the tray to the avionics shelf. Remove the tray.
      b) S/N 42501 and on – Remove the XM Weather Receiver per 34-6.a. Remove the three 10-32 screws attaching the tray to the avionics shelf. Remove the tray.
b. WX-500 Processor and Tray Installation (see Figure 34 - 72) – Installation is a direct reversal of removal. Hand tighten screws snug.

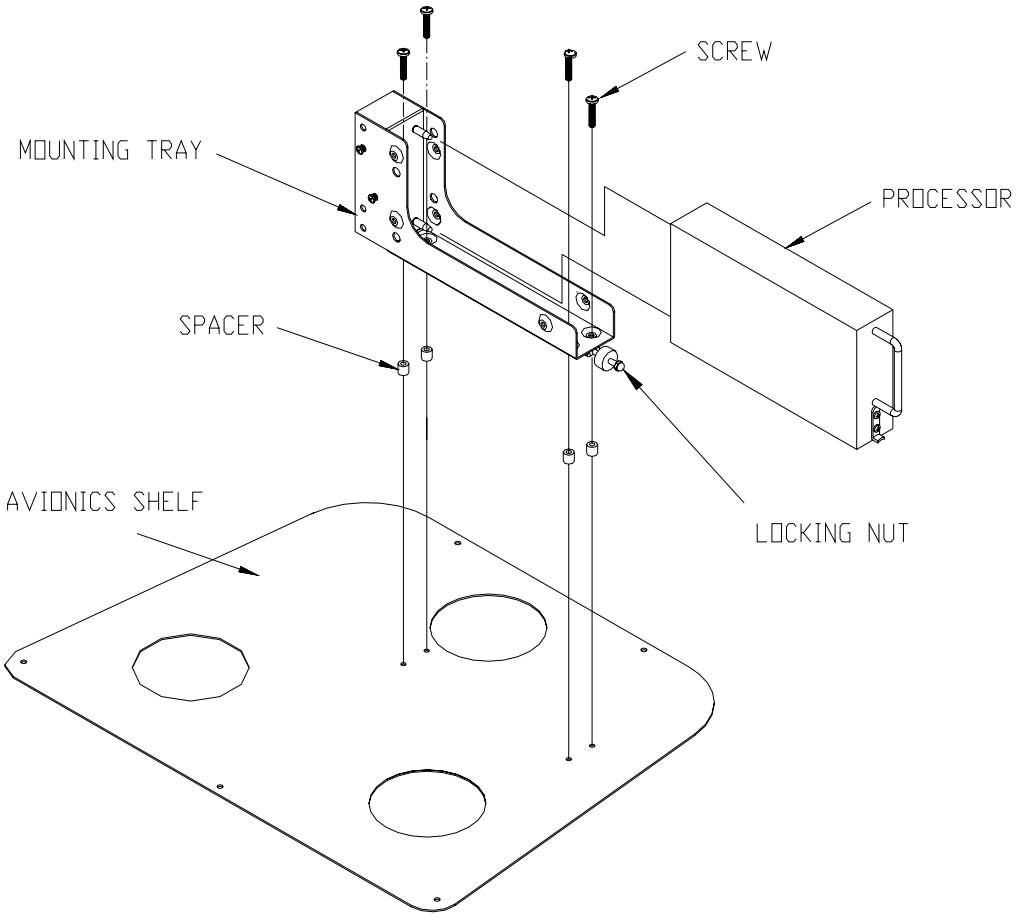


Figure 34 - 72 Stormscope Processor and Mounting Tray (S/N 42002 to 42500)

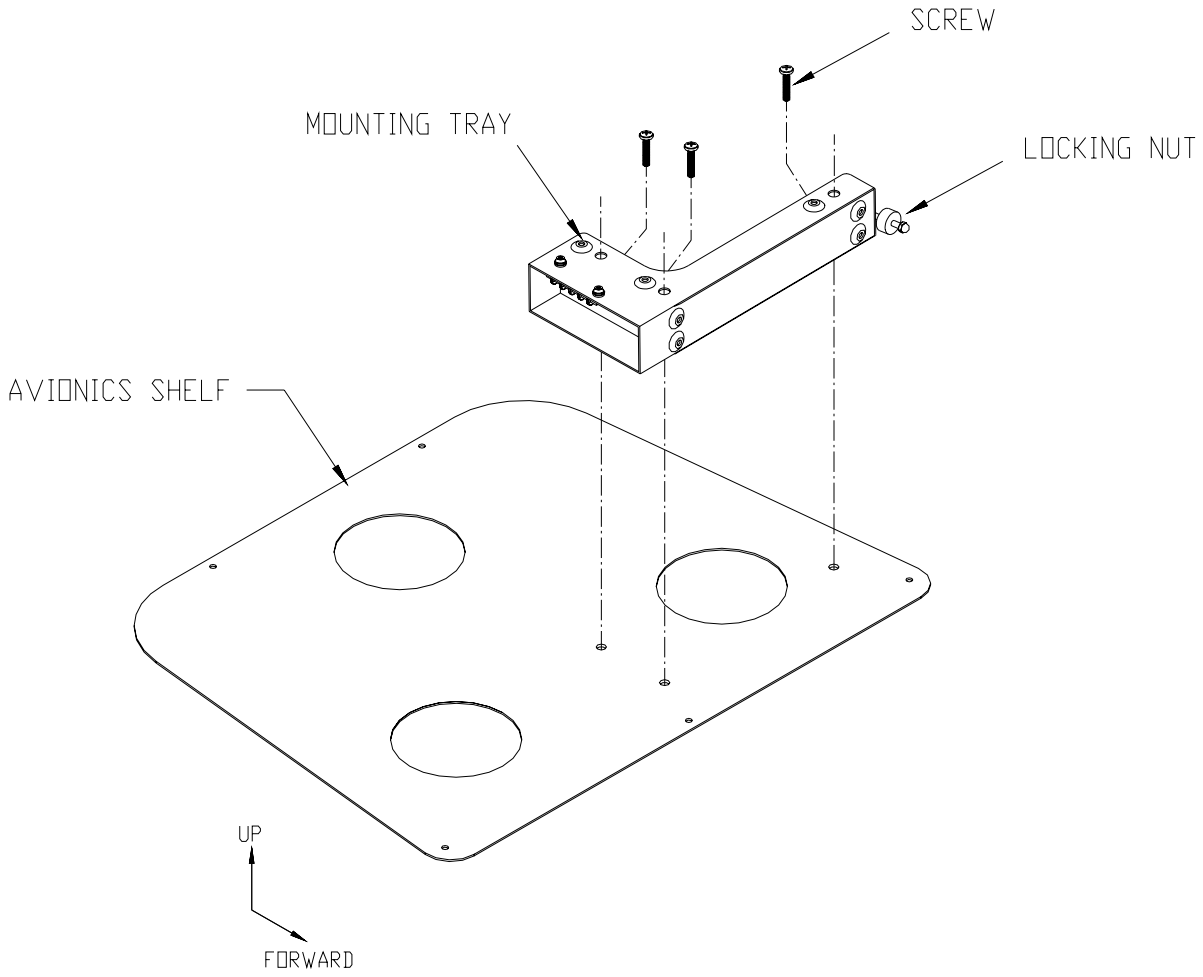


Figure 34 - 73 Stormscope Mounting Tray (S/N 42501 and on)

34-30. WX-500 STORMSCOPE ANTENNA – The WX-500 Stormscope Antenna is located under the hat rack.

a. WX-500 Stormscope Antenna Removal (see Figure 34 - 74)
   1. Ensure both the master and avionics switches are in the off position.
   2. Remove the hat rack per Chapter 25 to gain access to the baggage bulkhead access panel. Remove the baggage bulkhead access panel.
   3. Remove the antenna wiring connector.
   4. Remove RTV silicon caulk around antenna base, if present.
   5. Remove two screws attaching the antenna and gasket to the fuselage and antenna plate.
   6. If necessary, remove two screws attaching the antenna plate to the fuselage. If these screws do not go through the gasket, they must be removed.

b. WX-500 Stormscope Antenna Installation (see Figure 34 - 74)
   1. If necessary, install two screws attaching the antenna plate to the fuselage. Ensure these screws go through the gasket. Torque 30 to 36 in.-lbs.

2. Install two screws attaching the antenna and gasket to the fuselage and antenna plate. Torque 30 to 36 in.-lbs.
3. Apply and smooth RTV silicon caulk around the antenna base.
4. Attach antenna wiring connector.
5. Replace the baggage bulkhead access panel and the hat rack per Chapter 25.

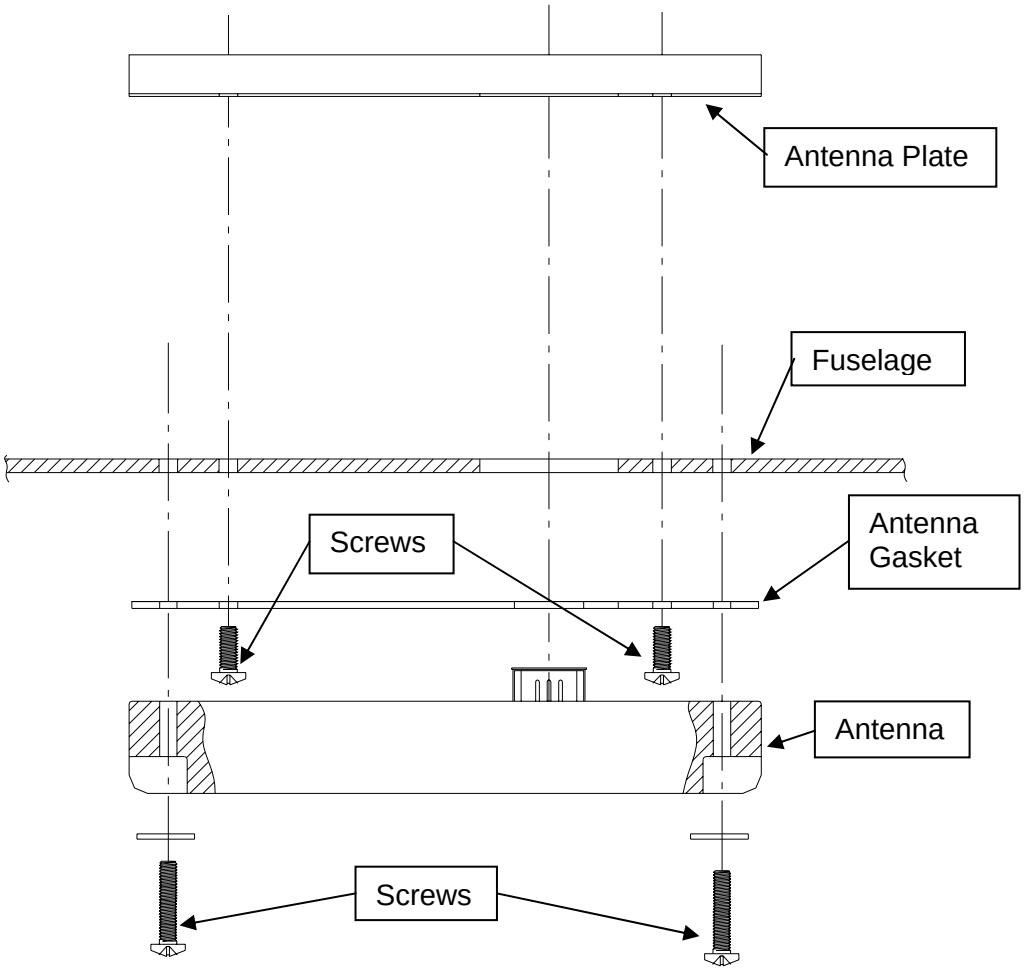


Figure 34 - 74 Stormscope Antenna Installation

34-31. MX20 MULTI-FUNCTION DISPLAY (OPTIONAL)

a. **Display Removal** – The map displays are located in the instrument panel, and the controller is located on the display unit (see Figure 34 - 75).
   1. Insert a $^{3}/_{32}$ in. hex wrench through the hole at the bottom center of the unit you want to remove.
   2. Turn the wrench counterclockwise at least 5 turns, and gently pull the unit away from the instrument panel until free.

b. **Display Installation**
   1. Slide the unit into the tray and insert a $^{3}/_{32}$ in. hex wrench through the hole at the bottom center of the unit you want to remove.

2. Turn the wrench clockwise while pushing on the unit. Do not over-torque the locking screw.

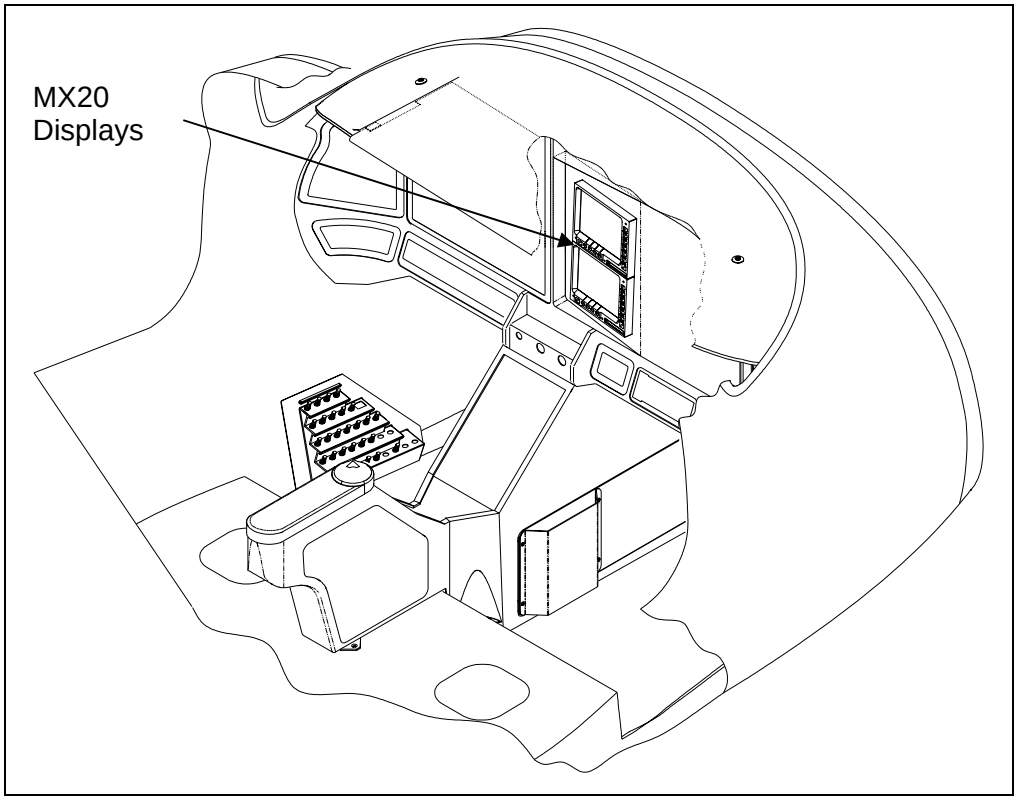
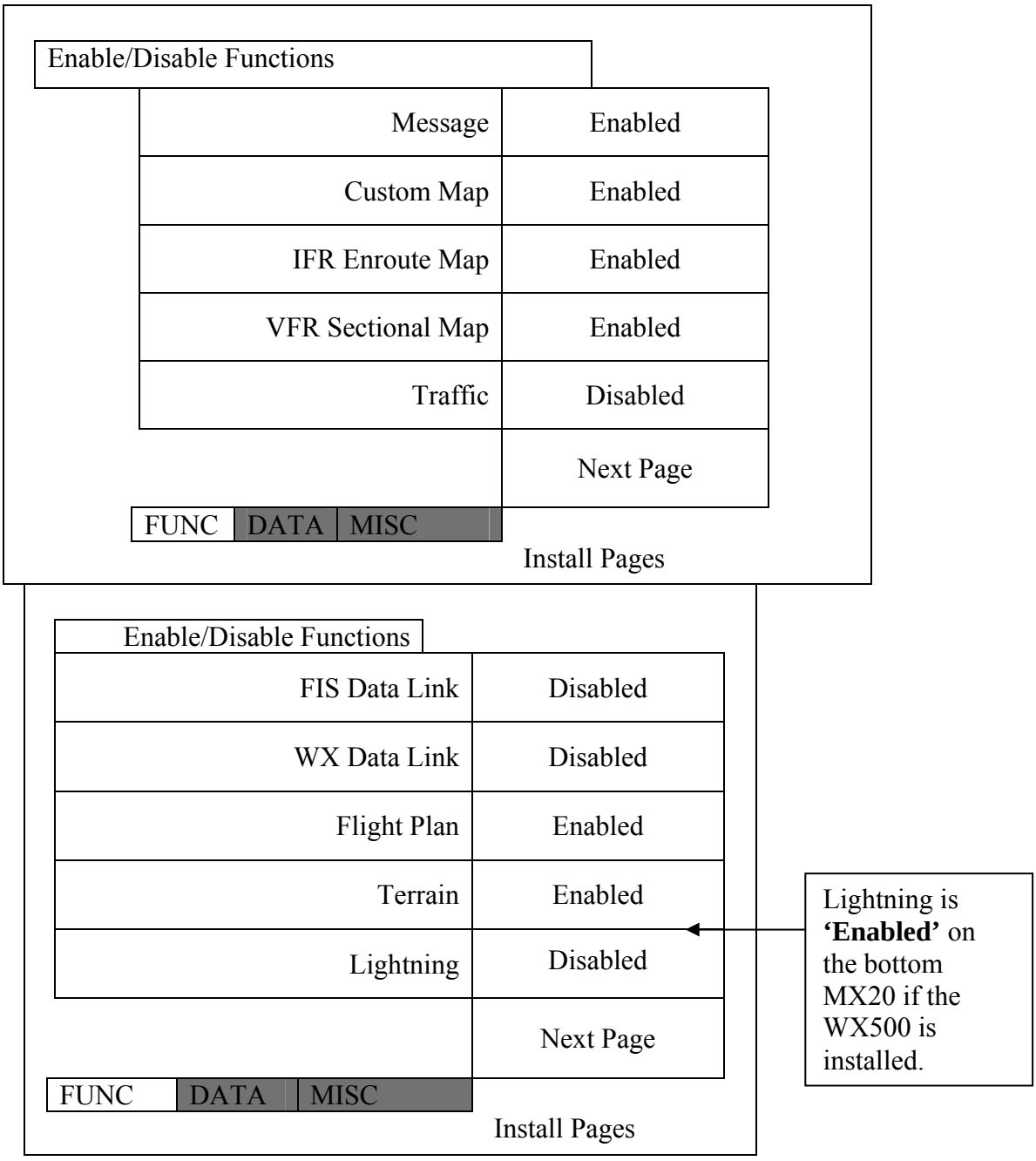
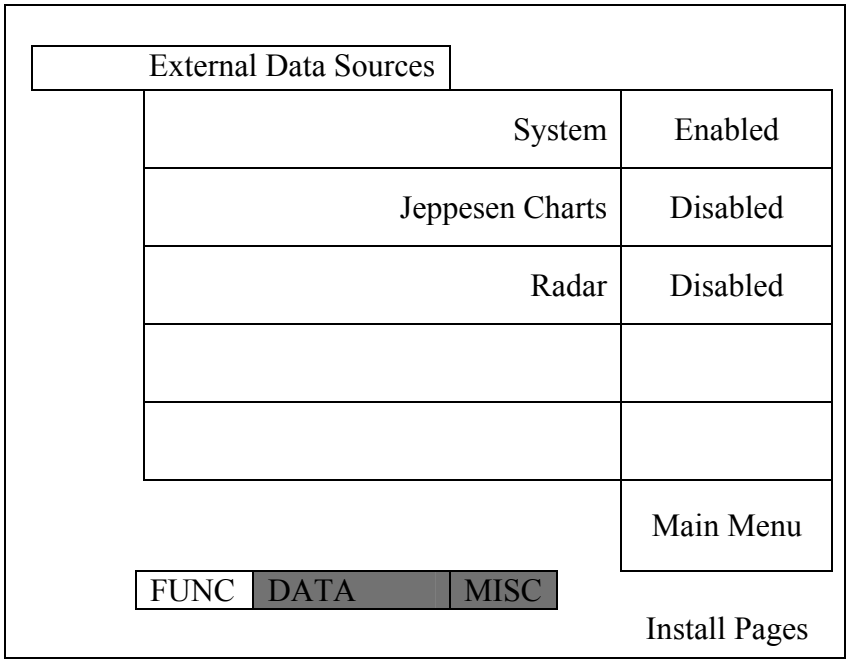


Figure 34 - 75 MX20 Displays

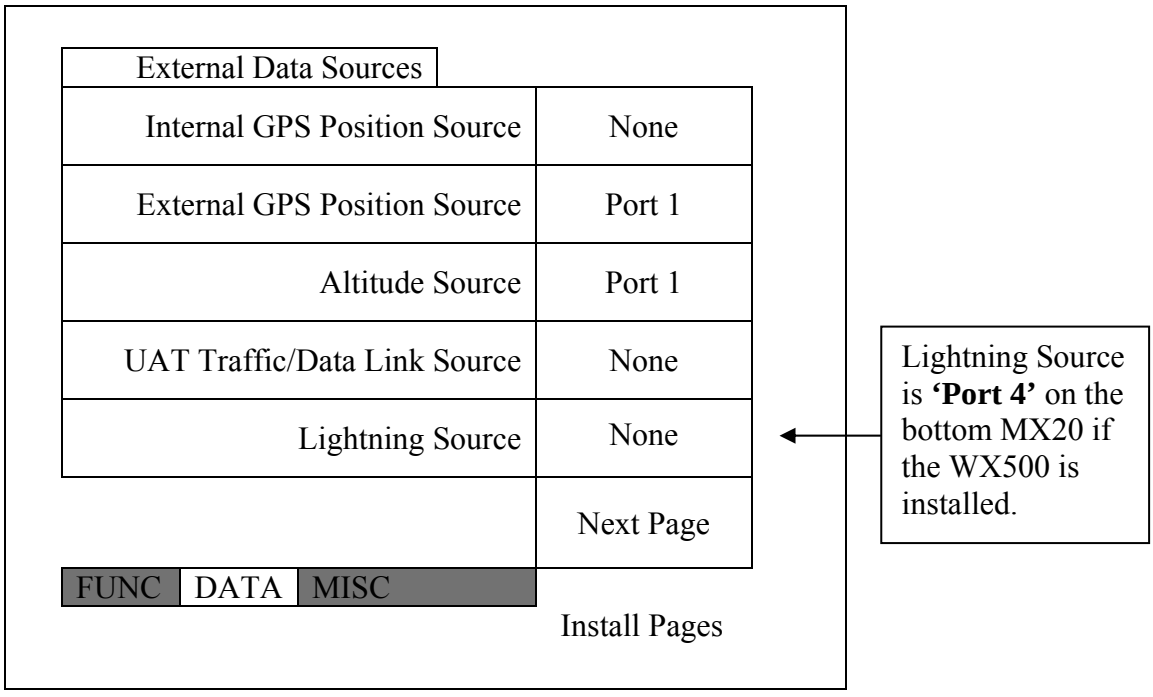
c. **Initial Set-up Procedure**

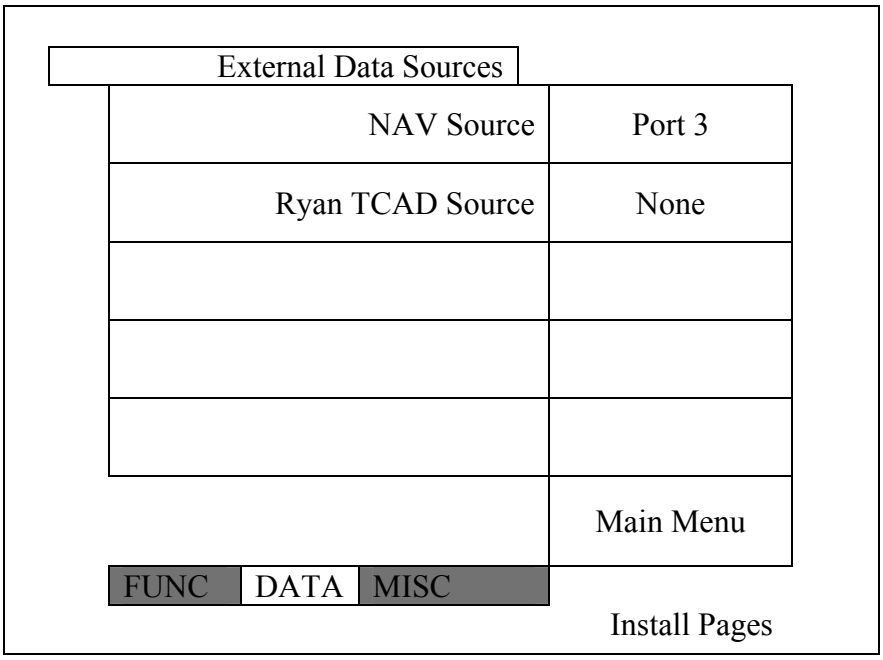
1. Turn the aircraft power and MX20 power to ON.
2. Immediately after the self-test of the unit is complete, press line select keys **1**, **4**, and **6** in sequence. Do not press any other buttons before starting the sequence. (Line select keys are located vertically along the right hand side of the MX20 unit below the rotary power knob, and are numbered in order with the top key as 1.
3. Depress the FN function key, one or more times until INSTL is displayed above the smart keys.
4. Select the INSTL function by pressing the smart key directly below the INSTL label on the screen. The Enable/Disable Functions will be the first screen to appear and FUNC will be highlighted above the smart keys.
5. Set the Enable/Disable Functions per the table below by pressing the line key next to the function label.


Enable/Disable Functions
Message
Enabled
Custom Map
Enabled
IFR Enroute Map
Enabled
VFR Sectional Map
Enabled
Traffic
Disabled
Next Page
FUNC
DATA
MISC
Install Pages
Enable/Disable Functions
FIS Data Link
Disabled
WX Data Link
Disabled
Flight Plan
Enabled
Terrain
Enabled
Lightning
Disabled
Next Page
FUNC
DATA
MISC
Install Pages
Lightning is **'Enabled'** on the bottom MX20 if the WX500 is installed.

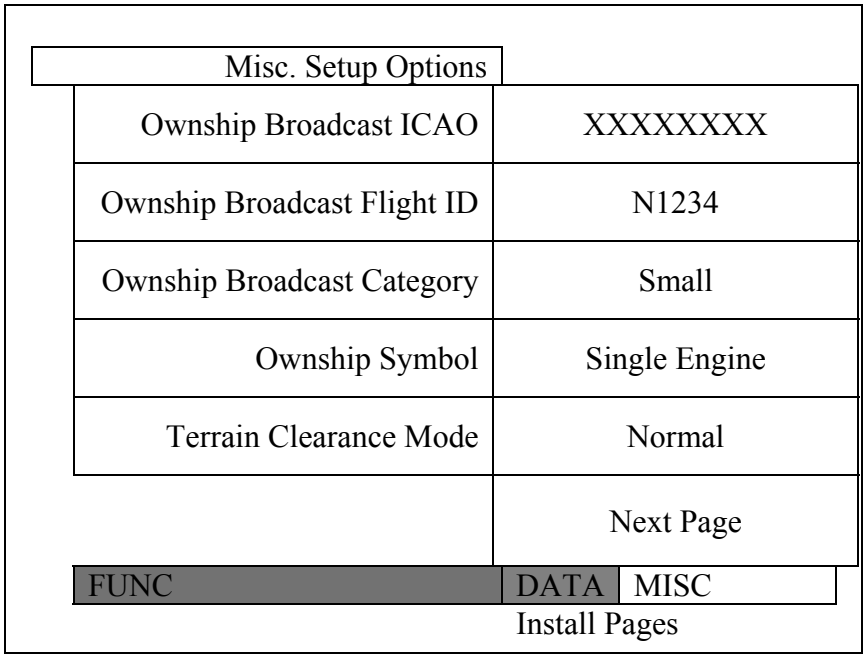


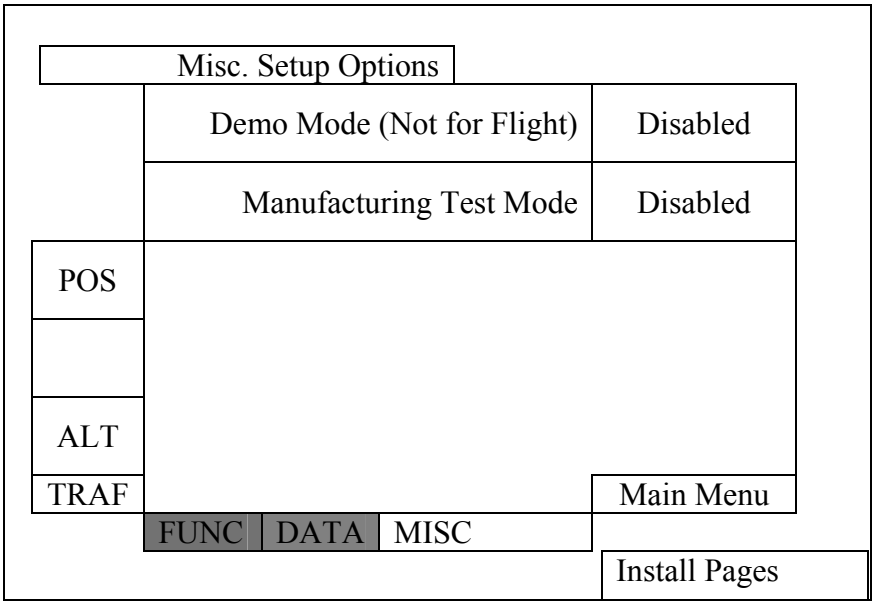
6. Press the smart key below the DATA label while still in the INSTL function. Set the External Data Sources per the table below by pressing the line key next to the function label.





7. Press the smart key below the MISC label while still in the INSTL function. Leave everything on these pages at factory defaults as shown below.





NOTE

After making final setting adjustments, do not power off for 30 seconds.

8. Press Enter/Menu to confirm all settings.
9. After the 30-second waiting period, turn the MX20 power off to apply all configuration settings.

34-32. ORBCOMM ANTENNA (Basic or Avidyne Option Only) – The ORBCOMM antenna is not installed with the Garmin G1000 System.

a. **ORBCOMM Antenna Removal**

1. The ORBCOMM antenna is located on top of the aircraft and forward of the vertical stabilizer. See Figure 34 - 76 and Figure 34 - 77.
2. Remove the aft center overhead light panel (ACOLP) per instructions in Chapter 25.
3. Disconnect the antenna cable from the connector on the base of the antenna. Keep the antenna connector clean with a plastic cover cap or other means of protection.
4. Remove the RTV 162 white silicone sealant around the base of the antenna and covering the four machine screws attaching the antenna to the fuselage.
5. With an assistant within the aircraft immobilizing the self locking nuts, remove the screws mounting the ORBCOMM antenna to the fuselage.
6. Remove the antenna.
7. Discard the antenna gasket.

b. **ORBCOMM Antenna Installation**

1. If necessary, remove all paint and primer from the entire contact area of the antenna gasket to expose the lightning mesh.
2. Position the ORBCOMM antenna and antenna gasket (PIC Wire and Cable part # PGP103-02 or PVF110-10) on top of the fuselage.
3. Attach the antenna to the fuselage using the four machine screws, washers and nuts. Have an assistant inside the fuselage immobilize the nuts while the screws are tightened.
4. Do not exceed 15 in-lbs of torque and tighten the hardware in a progressive pattern.
5. Allow the antenna gasket to cure for 24 hours per the gasket manufacturer's instructions.
6. Fill the voids around the antenna base and over the screw heads with RTV white silicone sealant conforming to Mil-A-46146. Smooth the sealant to the contours of the antenna base.
7. Connect the antenna cable to the connector on the base of the antenna.
8. Install the aft center overhead light panel (ACOLP) per instructions in Chapter 25.

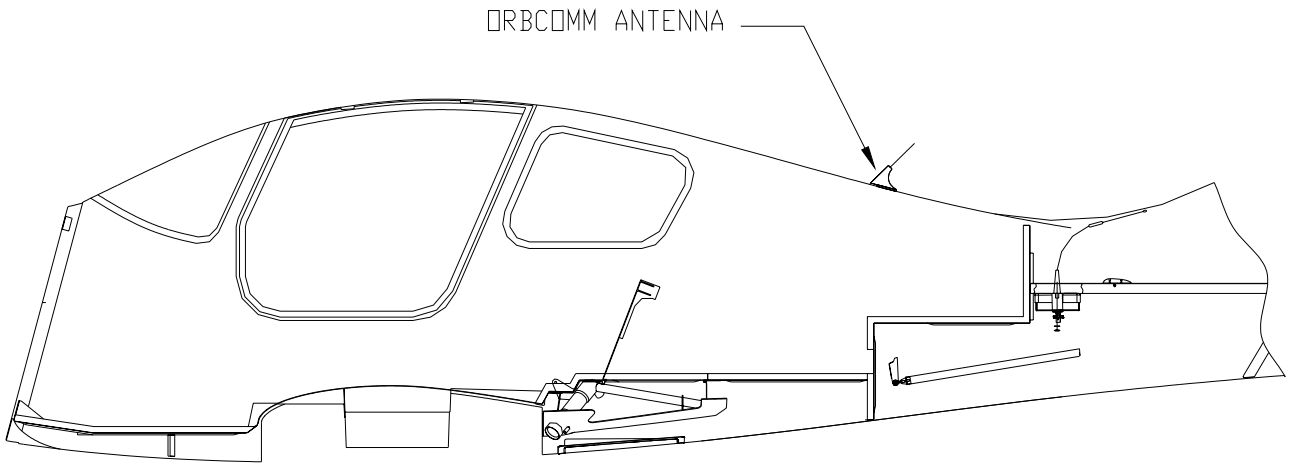


Figure 34 - 76 ORBCOMM Antenna Location

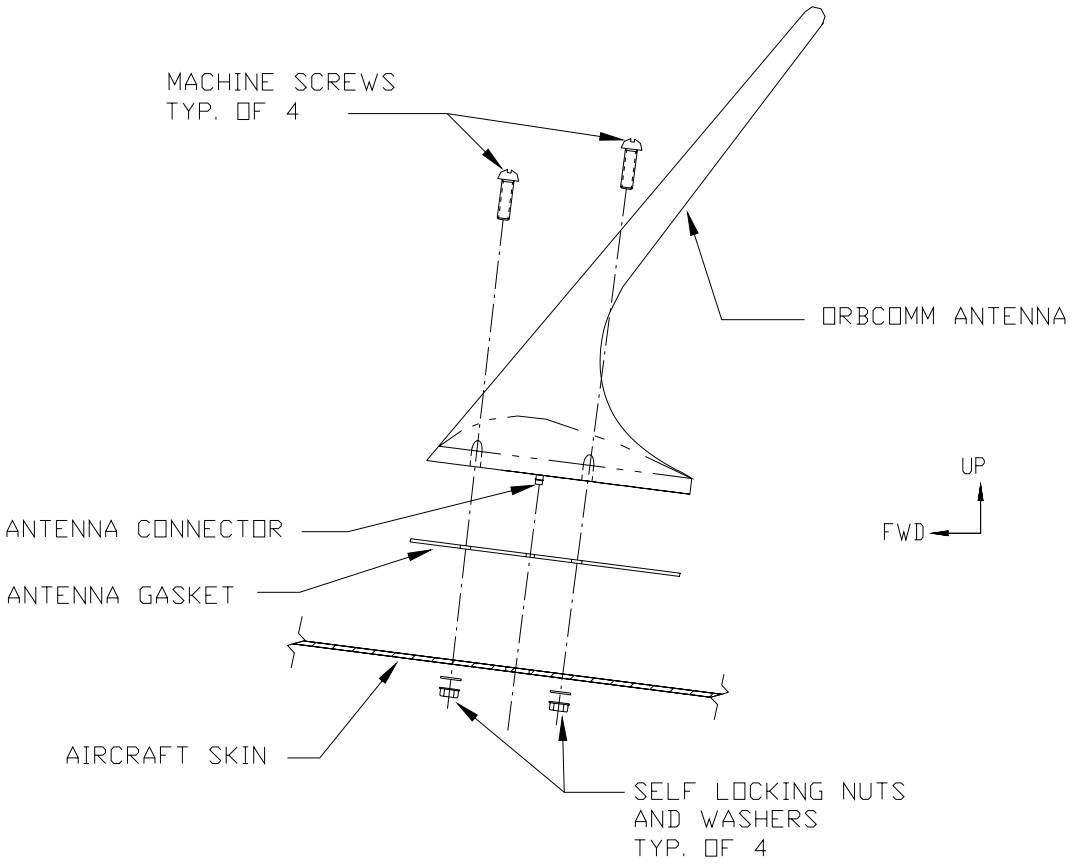


Figure 34 - 77 ORBCOMM Antenna

34-33. RYAN TCAD PROCESSOR

The Ryan TCAD Processor is located on the avionics shelf under the baggage compartment floor. See Figure 34 - 78.

a. **Processor Removal**
   1. Ensure both the master and avionics switches are in the off position.
   2. Remove the carpet and access panel from the baggage compartment floor.
   3. Disconnect all connectors to the unit.
   4. Loosen the two mounting screws that hold the processor onto the tray railings in the mounting bracket attached to the avionics shelf.
   5. Slide the processor off the tray railings and remove the unit.

b. **Processor Installation**
   1. Perform steps 1-4 above, in reverse order.
   2. Perform a functional check of the MFD in accordance with Section 34-23.d System Setup and Checkout.

34-34. RYAN TCAD ANTENNA

a. **Antenna Removal**
   1. Top Antenna – The top antenna is located on top of the aircraft and forward of the vertical stabilizer. See Figure 34 - 80 and Figure 34 - 81.
      a) Remove the aft center overhead light panel (ACOLP) per instructions in Chapter 25.
      b) Disconnect the cabling from the base of the antenna. Keep the antenna connectors clean with a plastic cap or other means of protection.
      c) Remove the RTV 162 white silicone sealant around the base of the antenna and covering the four machine screws attaching the antenna to the fuselage.
      d) With an assistant within the aircraft immobilizing the self locking nuts, remove the screws mounting the antenna to the fuselage.
      e) Remove the antenna.
      f) Discard the antenna gasket.
   2. Bottom Antenna – The bottom antenna is on the aileron bellcranks and crossover assembly access panel located on the bottom of the aircraft. See Figure 34 - 80 and Figure 34 - 81.
      a) Remove the aileron bellcranks and crossover assembly access panel.
      b) Disconnect the cabling from base of the antenna. Keep the antenna connectors clean with a plastic cap or other means of protection.
      c) Remove the RTV 162 white silicone sealant around the base of the antenna and covering the four machine screws attaching the antenna to the fuselage.
      d) Remove the screws mounting the antenna to the access panel.
      e) Remove the antenna.
      f) Discard the antenna gasket.

b. **Antenna Installation**
   1. Top Antenna
      a) If necessary, remove all paint and primer from the entire contact area of the antenna gasket to expose the lightning mesh.
      b) Position the antenna and antenna gasket (PIC Wire and Cable part # PTS109-01) on top of the fuselage.

c) Attach the antenna to the fuselage using the four machine screws, washers and nuts. Have an assistant inside the fuselage immobilize the nuts while the screws are tightened.
d) Do not exceed 15 in-lbs of torque and tighten the hardware in a progressive pattern.
e) Allow the antenna gasket to cure for 24 hours per the gasket manufacturer's instructions.
f) Fill the voids around the antenna base and over the screw heads with RTV white silicone sealant conforming to Mil-A-46146. Smooth the sealant to the contours of the antenna base.
g) Connect the antenna cables to the connectors on the base of the antenna.
h) Install the aft center overhead light panel (ACOLP) per instructions in Chapter 25.

2. Bottom Antenna
a) If necessary, remove all paint and primer from the entire contact area of the antenna gasket to expose the lightning mesh.
b) Position the antenna and antenna gasket (PIC Wire and Cable part # PTS109-01) on the aileron bellcranks and crossover assembly access panel.
c) Attach the antenna using the four machine screws.
d) Do not exceed 15 in-lbs of torque and tighten the hardware in a progressive pattern.
e) Allow the antenna gasket to cure for 24 hours per the gasket manufacturer's instructions.
f) Fill the voids around the antenna base and over the screw heads with RTV white silicone sealant conforming to Mil-A-46146. Smooth the sealant to the contours of the antenna base.
g) Connect the antenna cables to the connectors on the base of the antenna.
h) Install the aileron bellcranks and crossover assembly access panel.

34-35. RYAN TCAD TRANSPONDER COUPLER

The Ryan TCAD transponder coupler is located on the fuselage behind the instrument panel on the co-pilot's side for the basic or Avidyne option, and on the center wing aft spar behind the rear seats footwell for the Garmin G1000 option.

a. **Basic or Avidyne Option**
1. **Transponder Coupler Removal**
a) Ensure both the master and avionics switches are in the off position.
b) Remove the co-pilot's side kidney panel per Chapter 25
c) Disconnect the cabling to the coupler.
d) Depending upon the mounting configuration, there are either two screws, washers and nuts; or only two screws to be removed for detachment of the coupler from the mounting bracket. See Figure 34 - 82 and Figure 34 - 83.
2. **Transponder Coupler Installation** – Installation is the direct reverse of removal.

b. **Garmin G1000 Option**
1. **Transponder Coupler Removal**
a) Ensure both the master and avionics switches are in the off position.
b) Remove the rear seats footwell per Chapter 25
c) Disconnect the cabling to the coupler.

d) Remove two screws, washers and nuts attaching the coupler to the spar. See Figure 34 - 84.

2. **Transponder Coupler Installation** – Installation is the direct reverse of removal.

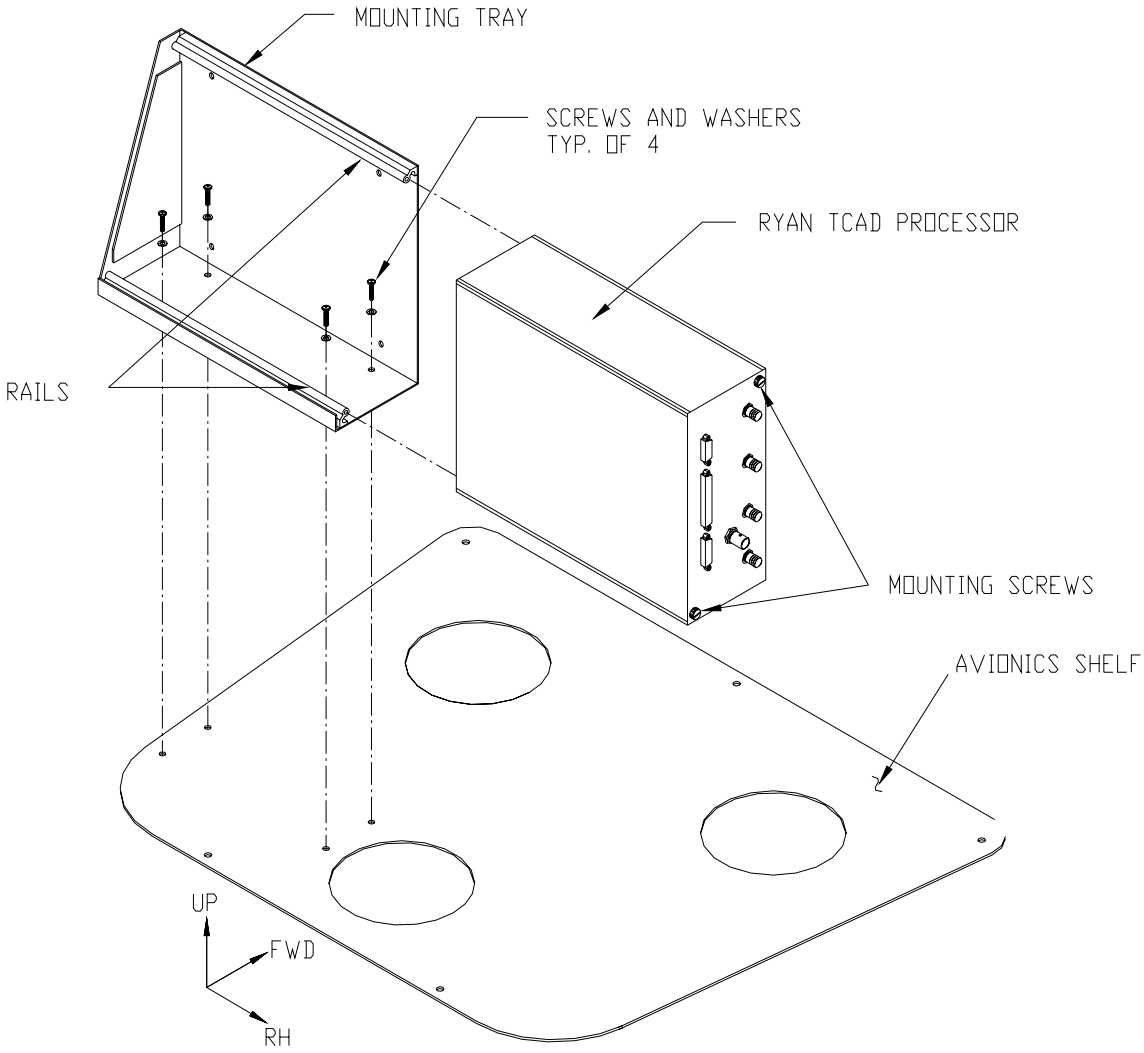


Figure 34 - 78 Ryan TCAD Processor (S/N 42001 to 42500)

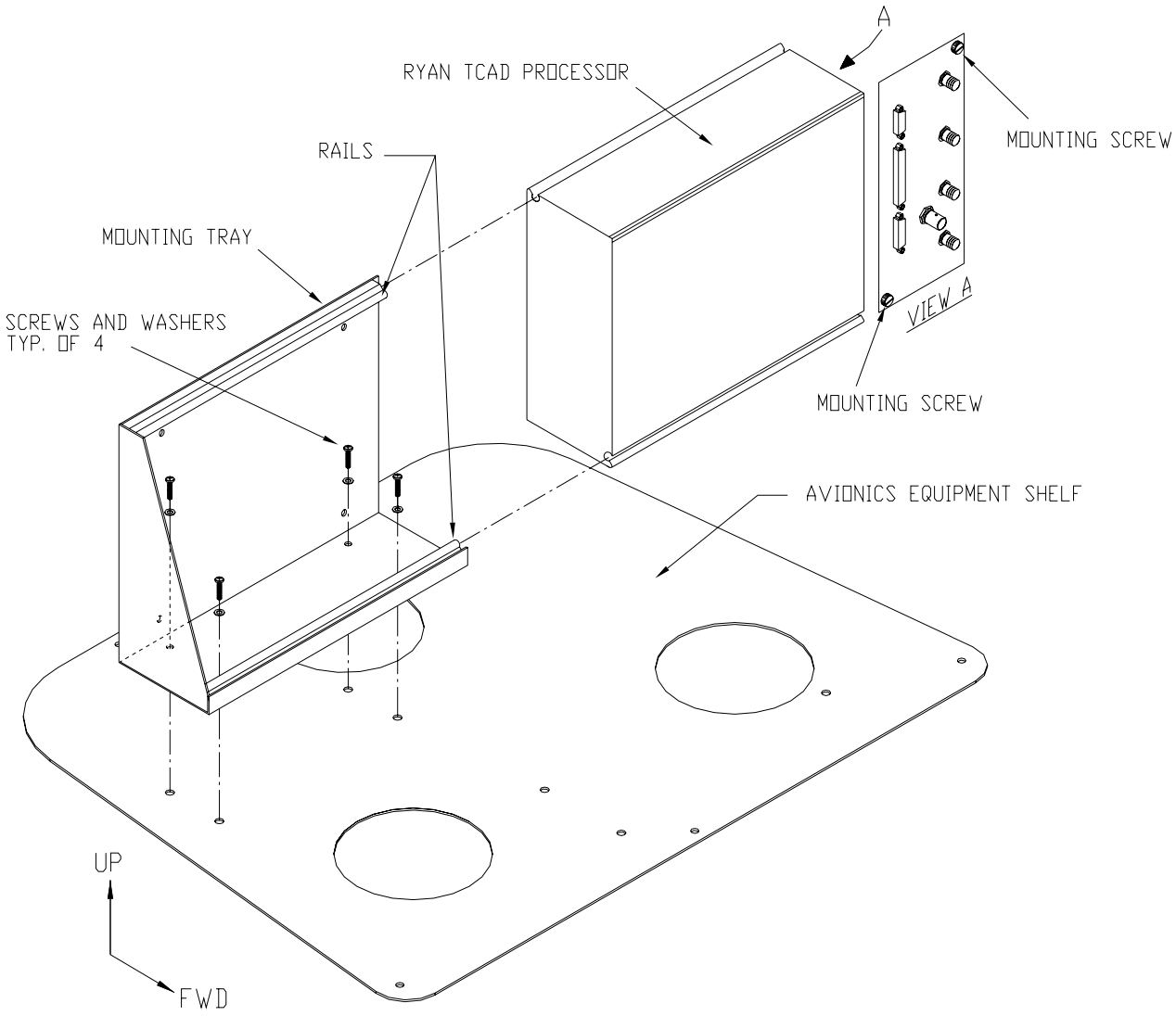


Figure 34 - 79 Ryan TCAD Processor (S/N 42501 and on)

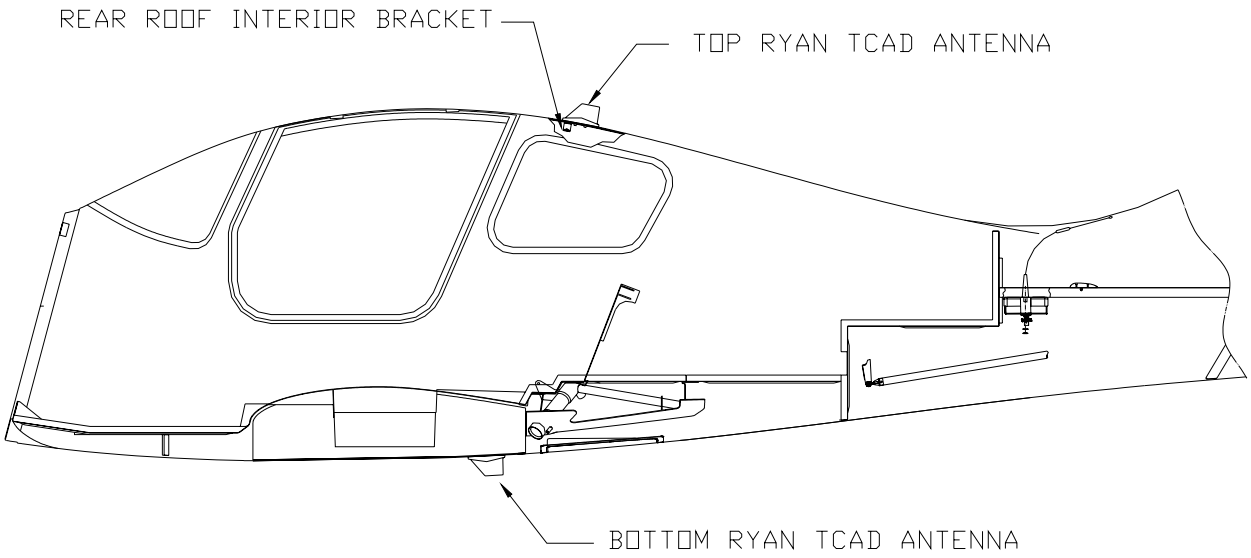


Figure 34 - 80 Ryan TCAD Antenna Locations

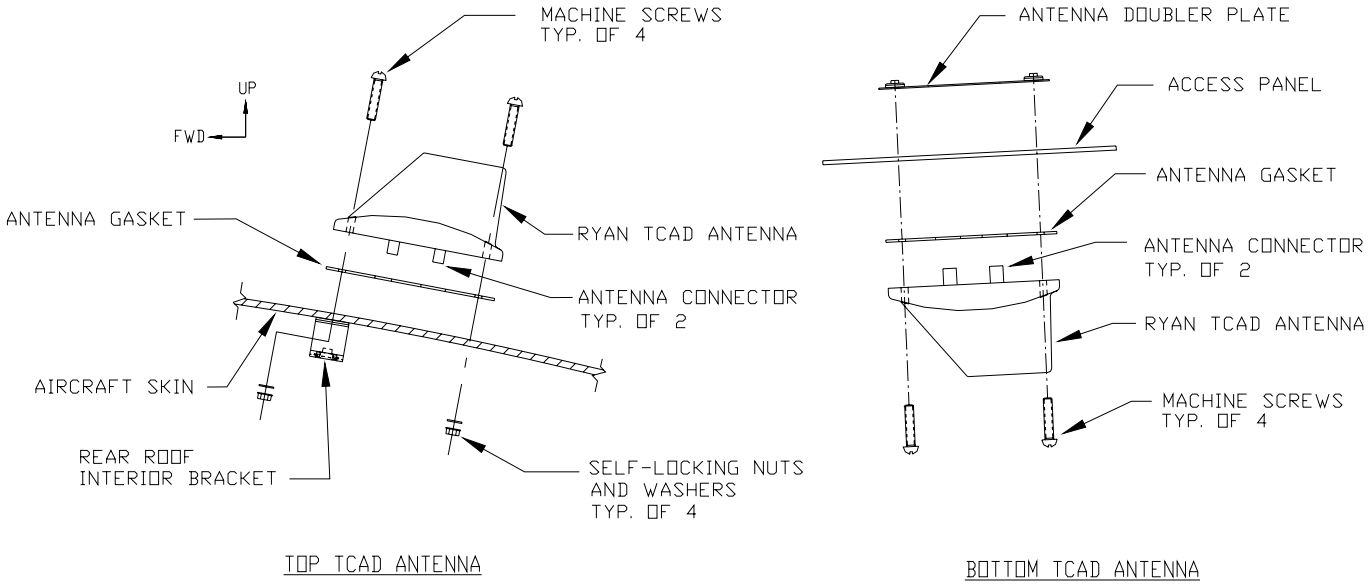


Figure 34 - 81 Ryan TCAD Antennas

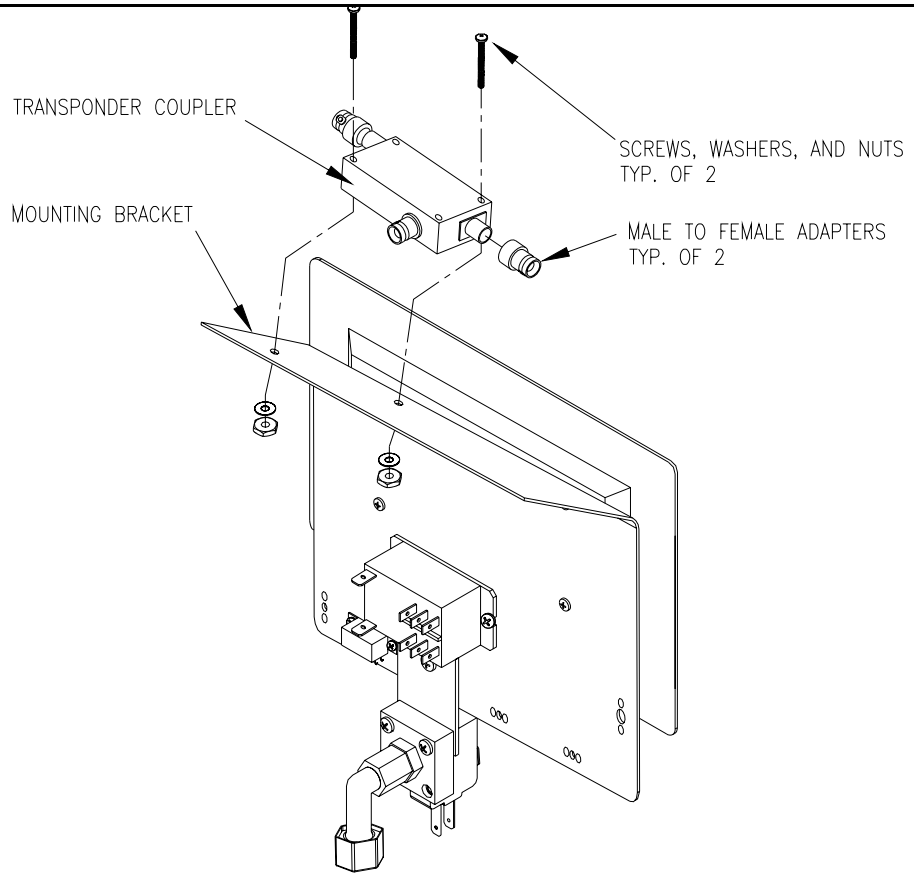


Figure 34 - 82 Ryan TCAD Transponder Coupler

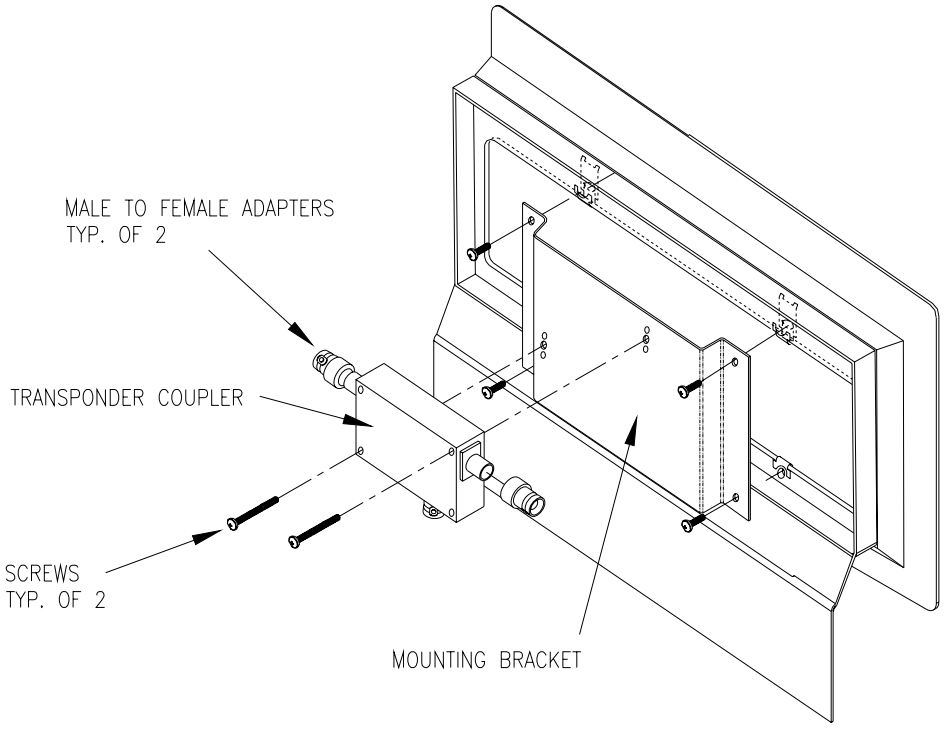


Figure 34 - 83 Ryan TCAD Transponder Coupler

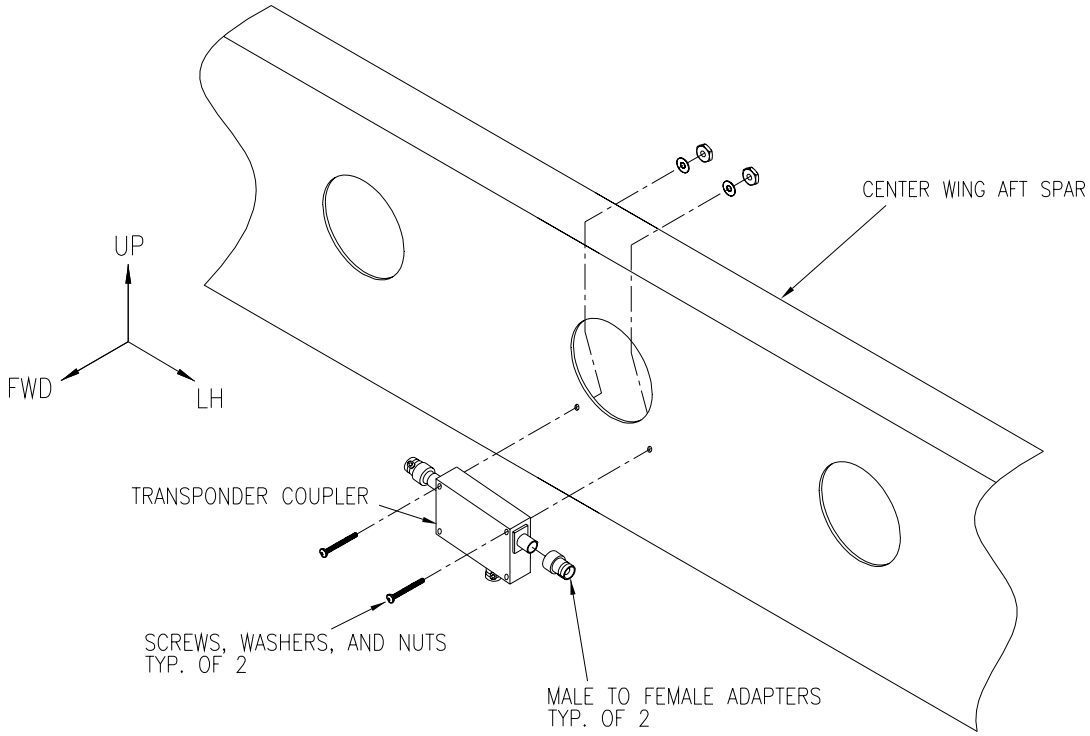


Figure 34 - 84 Ryan TCAD Transponder Coupler (Garmin G1000 Option)

34-36. XM WEATHER RECEIVER

The XM Weather Receiver is located on the avionics shelf under the baggage compartment floor. See Figure 34 - 85.

a. **Receiver Removal**
   1. Ensure both the master and avionics switches are in the off position.
   2. Remove the carpet and access panel from the baggage compartment floor.
   3. Disconnect all connectors to the unit.
   4. Remove the four screws and washers that hold the receiver onto the mounting bracket attached to the top of the two PFD voltage converters.
   5. Remove the receiver.

b. **Receiver Installation**
   1. Perform steps 1-4 above, in reverse order.
   2. Perform a functional check of the MFD in accordance with Section 34-23.d. System Setup and Checkout.

34-37. XM WEATHER ANTENNA

a. **Antenna Removal** – The antenna is located behind the instrument panel on the co-pilot's side. See Figure 34 - 88 and Figure 34 - 89.
   1. Remove the glare shield per instructions in Chapter 25.
   2. Disconnect the antenna cable from the connector on the base of the antenna. Keep the antenna connector clean with a plastic cover cap or other means of protection.
   3. Remove the screws, washers, and self-locking nuts mounting the antenna to the mounting bracket.

4. Remove the antenna.

b. **Antenna Installation**

1. Perform steps 1-4 above, in reverse order.

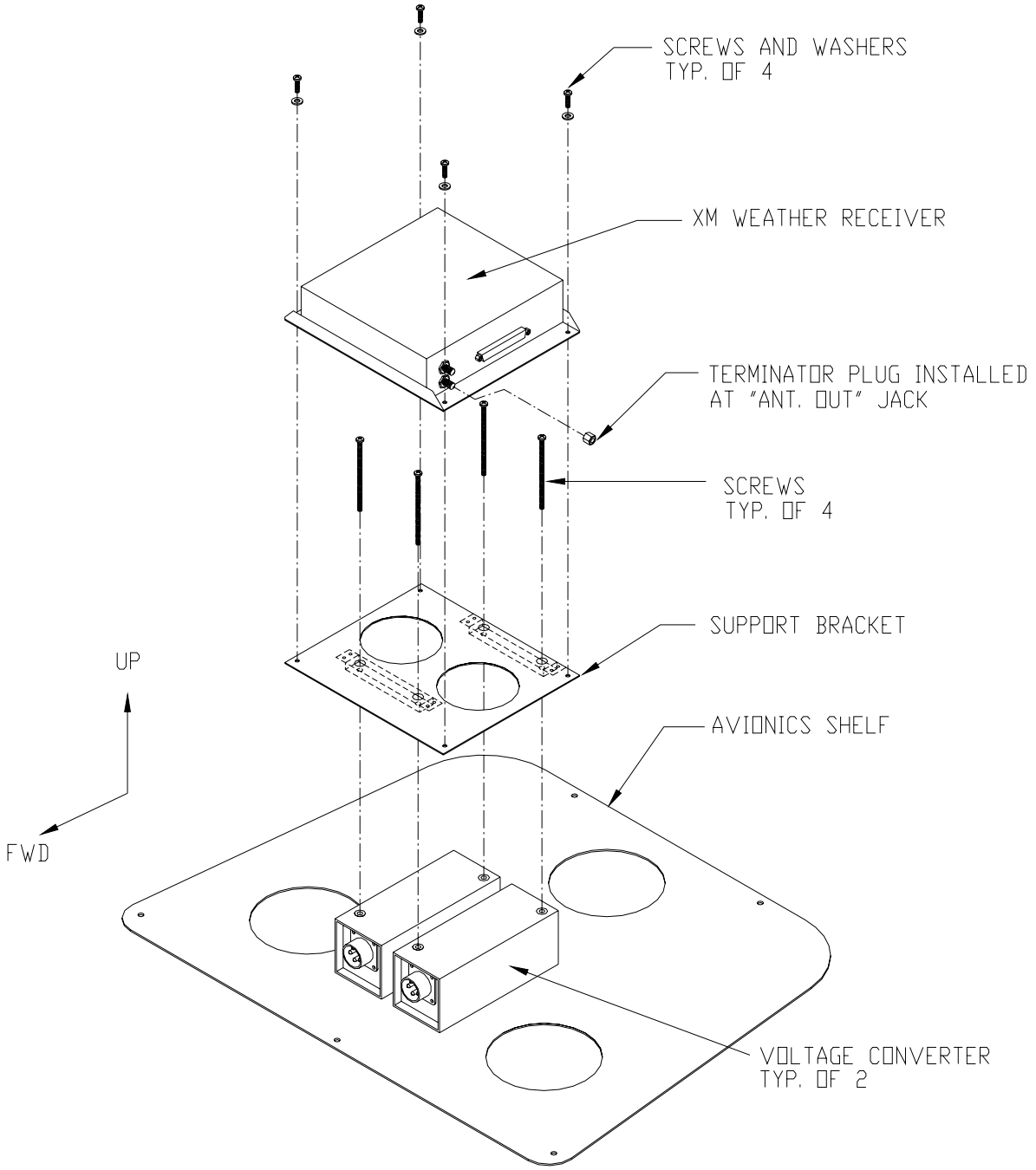


Figure 34 - 85 XM Weather Receiver (42001 to 42500)

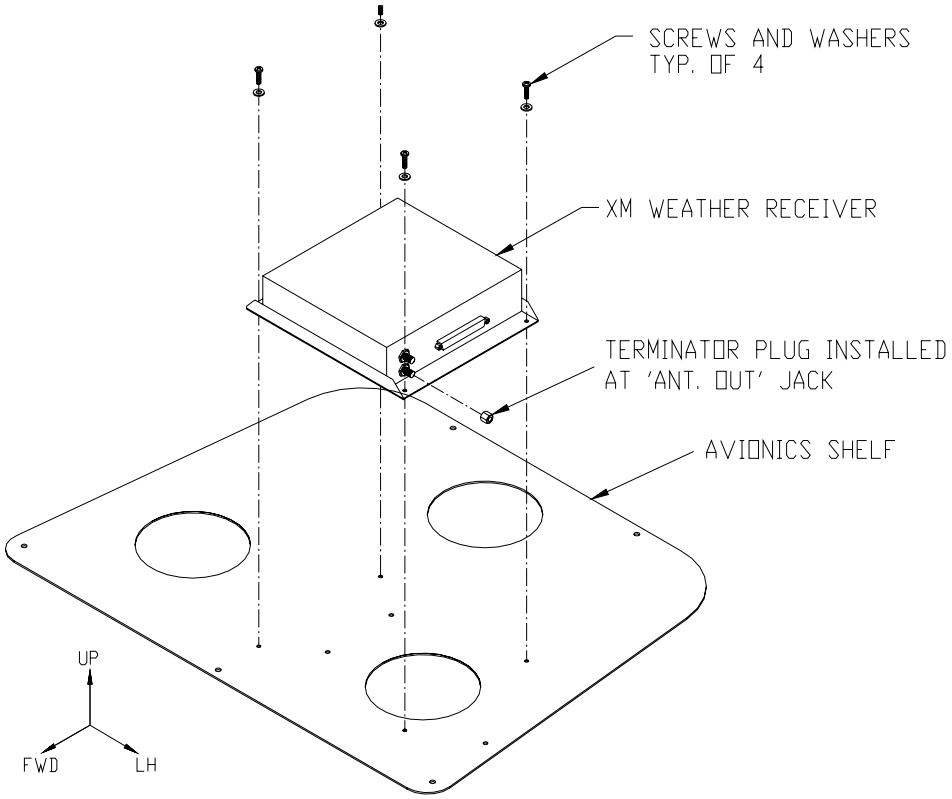


Figure 34 - 86 XM Weather Receiver (S/N 42501 and on)

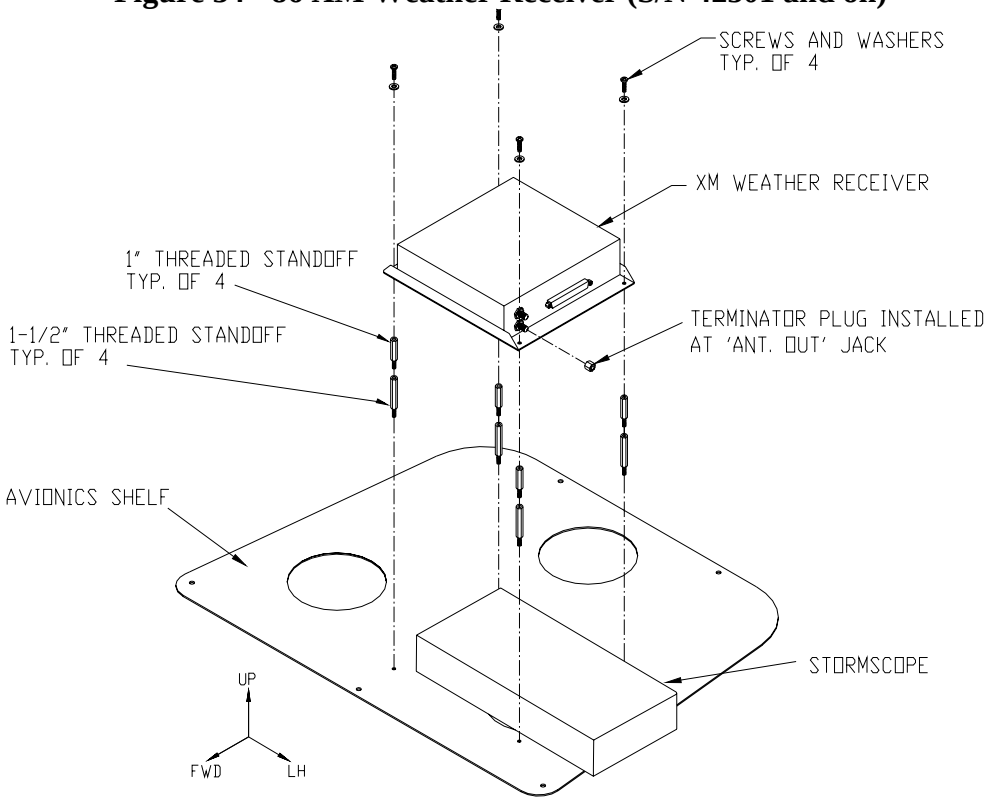


Figure 34 - 87 XM Weather Receiver if Stormscope Installed (S/N 42501 and on)

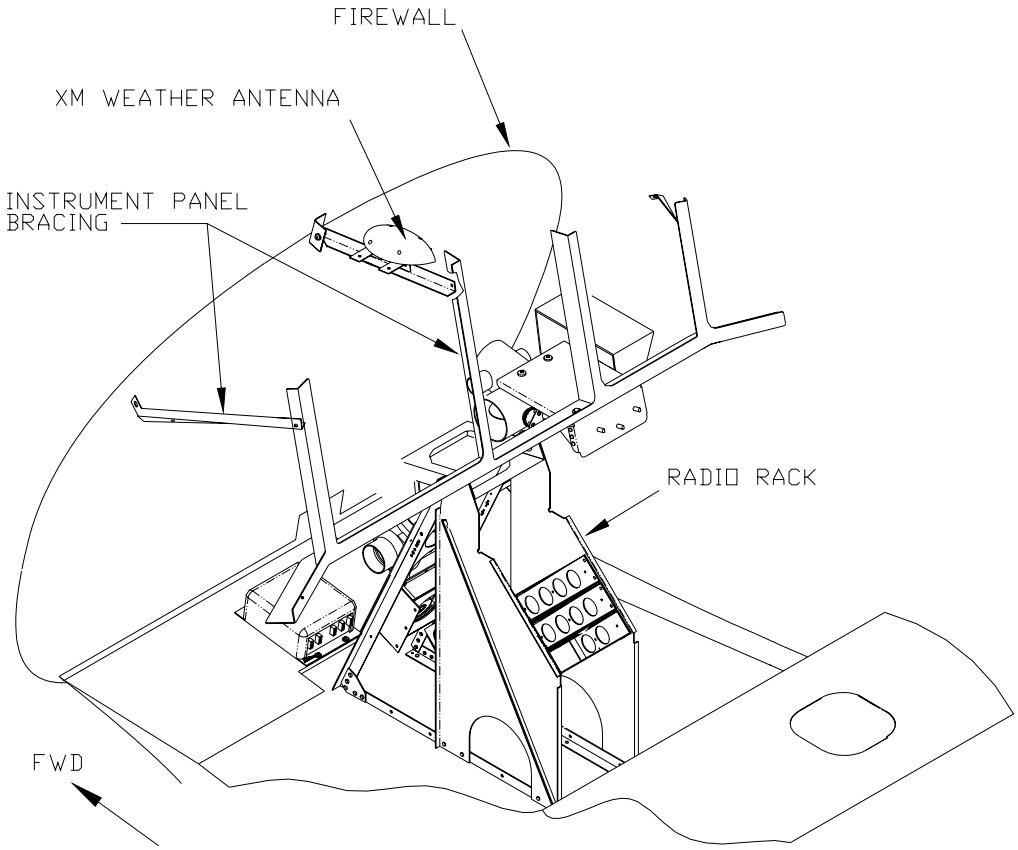


Figure 34 - 88 XM Weather Antenna Location

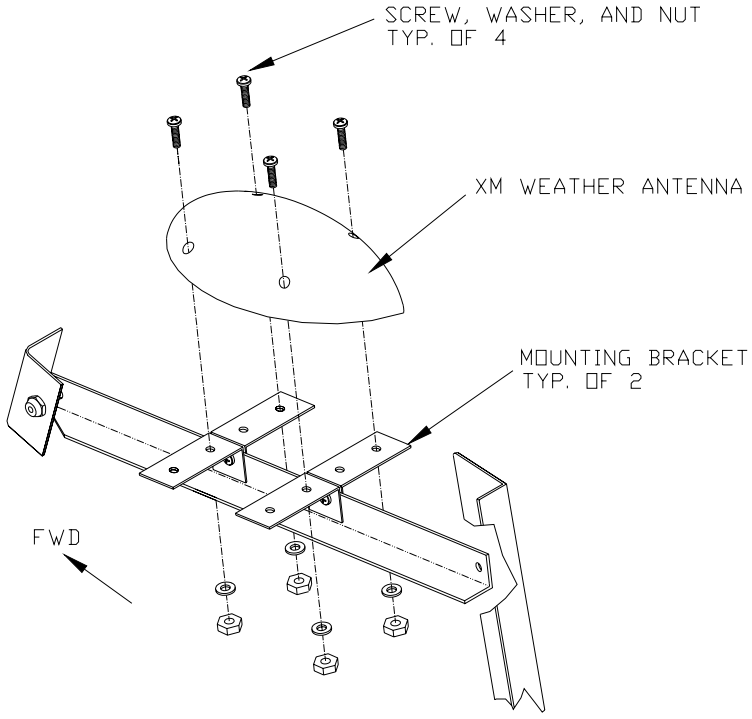


Figure 34 - 89 XM Weather Antenna

CHAPTER 35

OXYGEN

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Chapter 35

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35-1. GENERAL

The fixed oxygen system consists of three oxygen cylinders, high and low-pressure lines and fittings, a fill port, an overfill/overpressure burst valve, a regulator/valve assembly, an oxygen distribution manifold, a guarded emergency manual low-pressure oxygen shutoff, a logic and display panel, oxygen distribution devices, and flow meters. High-pressure lines connect the oxygen cylinders, fill port, overfill/overpressure burst valve and the regulator/valve assembly to the system. From the regulator/valve assembly, low-pressure oxygen lines are routed to the overhead console where a manual switch and distribution manifold are located. Attached to the low-pressure manifold and the regulator/valve assembly are pressure transducers that are connected electronically to the display/logic controller mounted in the instrument panel (Basic or Avidyne option) or to the MFD (Garmin G1000 option). See Figure 35 - 1.

a. **Oxygen Bottles –** There are three oxygen cylinders, each with 14 cu ft. capacity at 2000 psi. The oxygen cylinders used are manufactured to DOT-3AL/2015 with a 2015 psi working pressure, per 49 CFR §178.46. They are located in the right wing between the root rib and the WS 46 rib, and between the forward and aft spars. They are mounted to brackets attached to the lower wing skin. Stainless steel band clamps secure the cylinders to the brackets. The cylinders are fitted with brass discharge fittings that are connected by high-pressure stainless steel tubing and fittings.

b. **Tubing –** Stainless steel lines are used throughout the system. The tubing routes high-pressure oxygen from the fill port to the high-pressure manifold on the regulator/valve assembly and from there to the cylinders and the overfill/overpressure burst valve. All oxygen lines are marked with breathing oxygen identification tape, per MIL-STD-595B and MIL-STD-1247C, located within 3 inches of all fittings and at intervals along the lines not to exceed 48 inches. The ends of the tubing are flared at 37º. The system uses stainless steel, 37° JIC fittings. Copper fitting seals (Seco Seals) have been incorporated at every flared tube connection to help ensure leak free connections

c. **Flexible Hose –** Behind the forward overhead light panel a flexible hose connects the cabin distribution manifold to the low pressure side of the system. Flexible hose connection to tubing is accomplished using quick disconnect fittings.

d. **Fill Port –** The oxygen system fill port is located just inside the baggage door on the baggage hat rack. It consists of a schrader type valve, silver soldered to a stainless line which is routed to the high-pressure manifold. There is a placard cautioning “OXYGEN FILL PORT DO NOT EXCEED 2000 p.s.i.”. The valve includes a protective cap to screw over the valve when not in use.

e. **Overfill/Overpressure Burst Valve Assembly –** The system incorporates an overfill/overpressure burst valve assembly that functions to discharge the system in the event it is overpressured. The valve is located in the center of the lower wing skin just under the rear footwell. Stainless steel tubing connects the assembly to the high-pressure manifold located on the root rib. A threaded cartridge bursts when oxygen pressure exceeds 2800 psi. The burst valve exhausts through a hole in the lower wing skin. Under normal conditions there is a pressed-in cap plug that closes the hole in the wing skin and protects the system from dust and dirt. In the event of an overpressure condition, the cap will be blown out of the hole from the internal pressure and act as an additional indication of the system overpressure protection being activated during preflight.

f. **Oxygen Regulator/Valve –** The oxygen regulator/valve assembly includes the high pressure manifold, high pressure transducer, regulator, low-pressure valve, and valve actuating solenoid. The regulator/valve assembly is located on the inboard side of the root rib and is attached to the root rib with bolts, washers, and locknuts. The high-pressure manifold part of the assembly has three external ports for the fill line, the cylinder line and the overpressure burst valve line. The pressure regulator provides 22 to 33 psig to the cabin distribution manifold. The remote high-pressure control is an electrically operated on-off valve that is actuated by a latching solenoid connected to the aircraft essential electrical bus through the logic and display assembly. There is one low-pressure outlet port on the assembly. The assembly also contains a high-pressure transducer that sends a signal to the logic and display unit to show cylinder pressure. The cylinder pressure directly relates to the quantity of oxygen in the system.

g. **Cabin Distribution Manifold –** The cabin distribution manifold is located in the overhead console, just behind the pilot's head. It has four quick-disconnect ports.

h. **Manual Low-Pressure Oxygen Shutoff Valve –** The manual oxygen shutoff valve is located next to the distribution manifold. Its function is to turn off the flow of low pressure oxygen as necessary in case of a failure of the regulator/valve assembly. There is a guard around the switch to prevent inadvertent contact with the shutoff switch.

i. **Logic and Display Unit (Basic or Avidyne Option Only) –** The logic and display unit consists of two assemblies: the display located on the lower right of the instrument panel and the remote unit located behind the right kidney panel. The remote unit contains a circuit board, a display board assembly, and a dust cover. The display consists of a three position switch (ON, OFF, and DISP) that controls the main high-pressure regulator, an LED lit oxygen quantity indicator, and an LED lit low-pressure indicator, and a fault light. The DISP position on the switch turns on the low-pressure and oxygen quantity indicators without energizing the regulator. The ON position turns power on to the display, and energizes the solenoid for a short duration to open the low-pressure valve and provide low-pressure oxygen to the cabin distribution manifold. The OFF position will disengage the solenoid and there will be no display. The logic controller also includes a fault indication for a short to ground, or an open wire for the solenoid wiring only. The system has built-in current protection in case the solenoid wires short each other.

j. **Oxygen Flow Controls –** Four manually operated oxygen flow controls can be connected to the oxygen distribution manifold. The flow controls provide the means to distribute the appropriate amount of oxygen for the pressure altitude of flight and indicate the presence of flowing oxygen to the pilot or passenger(s). The flow controls can be either an A4 Flowmeter and Oxygen Conserving Cannulas or an A4 Flowmeter and Mask (Standard and Microphone).

k. **Grounding and Bonding –** All lines are electrically bonded across fittings, and to the oxygen system components to prevent the build up of static electricity. The whole system is electrically bonded to the ground plane of the aircraft for both static electricity and lightning protection.

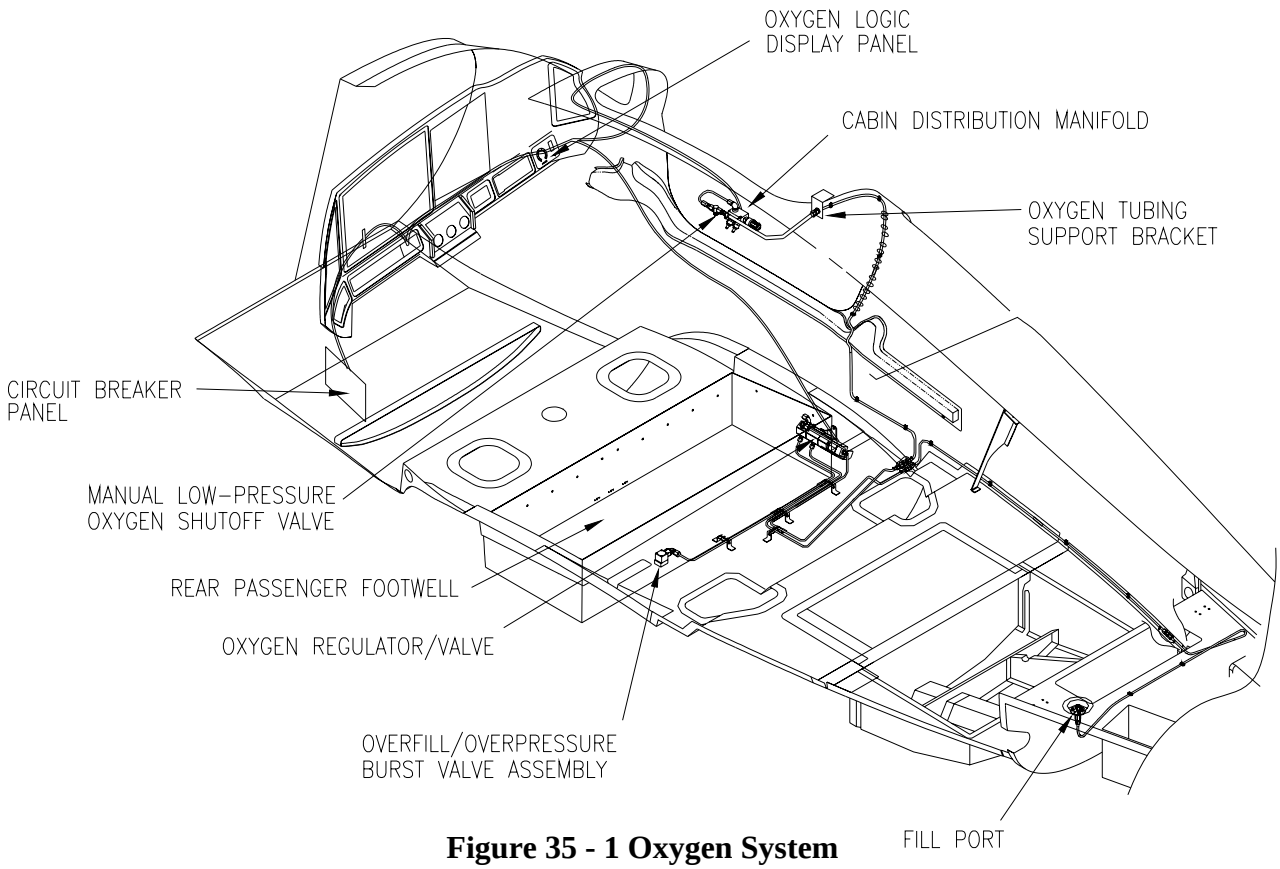

Figure 35 - 1 Oxygen System

35-2. SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM

In servicing and maintaining oxygen equipment, the greatest potential hazard is fire and explosion. Given conducive conditions, gaseous oxygen will cause combustibles like oil and grease to ignite and burn vigorously or explode. Other materials that do not burn in air will burn vigorously in an oxygen enriched environment. Oxygen equipment must be kept free of all types of oil and grease. Refer to this chapter and FAA Advisory Circular 43.13-1B/1 or subsequent prior to servicing the oxygen system.

a. **Cleanliness of Tools and Equipment –** Servicing tools and equipment must be cleaned regularly with anhydrous ethyl alcohol, isopropyl alcohol, or other suitable oxygen safe cleaning agents. After being cleaned and dried, tools and equipment must be protected with caps, plugs, or covers as applicable and stored in a clean storage area.

b. **Cleanliness of Personnel –** All clothing must be free of oil, grease, fuel, hydraulic fluid, etc. Personnel must be clean without any lotions or lip balms. Thorough washing of hands and face should be adopted as standard practice before handling of any oxygen equipment or servicing tools.

c. **Grounding –** Ground the system and servicing equipment before connecting to the fill port.

d. **Electric Power –** Disconnect all unnecessary electrical power from the aircraft. When charging the oxygen system keep operation of electrical switches to the minimum required. Do not connect or disconnect the ground power supply.

e. **Ignition Sources –** Avoid making sparks. Keep fire and burning cigarettes away from the vicinity of the aircraft during oxygen servicing.

f. **Fueling –** Do not service or perform maintenance on the oxygen system when fueling, or servicing with other flammable fluid, of the aircraft is in progress.
g. **Oxygen System Charging –** Do not charge the system too quickly. Rapid charging can generate heat and thus create a hazard to safety.
h. **Oxygen System Pressure –** The oxygen system is designed to be filled to a maximum pressure of 2000 psi. Verify the pressure of oxygen filling equipment prior to servicing the aircraft to prevent over-pressurizing the system.

35-3. OXYGEN SYSTEM SERVICING

Servicing of the oxygen system is as indicated in Chapter 12.

WARNING

Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing or maintenance of the oxygen system.

35-4. OXYGEN SYSTEM PURGING

The oxygen system should be purged anytime the system's pressure drops below 50 psig or the lines are left open. Removal of offensive odors in the system may also be accomplished by purging of the system. Use only MIL-O-27210 aviator's breathing oxygen for system purging.

WARNING

Do not use medical or industrial quality oxygen. They may contain excessive moisture that could freeze and cause malfunction of the oxygen system during flight.

a. Open all aircraft doors to provide air circulation.
b. Remove all unnecessary electrical power to the aircraft. Energize only the left bus of the aircraft electrical power (Basic or Avidyne option) or the avionics and the right bus (Garmin G1000 option).
c. Locate the oxygen fill port in the baggage bulkhead.
d. Ground the oxygen system by connecting a grounding cable from the oxygen fill port to the oxygen recharging cart, ensuring that the cart is properly grounded.
e. Remove the fill port protective cap and attach the hose from the oxygen recharging cart to the fill port.
f. Set the recharging cart to deliver 50 psig of oxygen to the system.
g. Turn on the regulator/valve using the switch on the logic and display panel (Basic or Avidyne option) or the softkey on the MFD (Garmin G1000 option).
h. Open one of the ports in the cabin distribution manifold by using an oxygen mask line or equivalent and ensure that oxygen is flowing.
i. Purge the system for one hour.
j. If purging was performed to remove offensive odors, and odors still linger after one hour, purge the system for an additional hour. If offensive odors still persist, replace the supply cylinders, and repurge the system.
k. Turn off the regulator/valve using the switch on the logic and display panel (Basic or Avidyne option) or the softkey on the MFD (Garmin G1000 option).
l. Turn off the recharging cart.

m. Remove the oxygen mask line or equivalent from the cabin distribution manifold.
n. After successful completion of system purging, service the system as described in Chapter 12.

WARNING

Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing or maintenance of the oxygen system.

35-5. OXYGEN SYSTEM DEPRESSURIZATION AND EVACUATION

The oxygen system must be depressurized and the oxygen removed for any work done to the system that involves removal of component(s).

a. **Using an Oxygen Cart Capable of Oxygen Recovery**
   1. Open all aircraft doors to allow for air circulation.
   2. Remove all unnecessary electrical power to the aircraft. Energize only the left bus of the aircraft electrical power (Basic or Avidyne option) or the avionics and the right bus (Garmin G1000 option).
   3. Locate the oxygen fill port in the baggage bulkhead.
   4. Ground the oxygen system by connecting a grounding cable from the oxygen fill port to the oxygen cart, ensuring that the cart is properly grounded.
   5. Remove the fill port protective cap and attach the hose from the oxygen cart to the fill port.
   6. Evacuate the oxygen system.
   7. Turn off the oxygen cart.

b. **Using the Aircraft Cabin Distribution Manifold**
   1. Open all aircraft doors to allow for air circulation.
   2. Remove all unnecessary electrical power to the aircraft. Energize only the left bus of the aircraft electrical power (Basic or Avidyne option) or the avionics and the right bus (Garmin G1000 option).
   3. Turn on the regulator/valve using the switch on the logic and display panel (Basic or Avidyne option) or the softkey on the MFD (Garmin G1000 option).
   4. Open one or more of the ports in the cabin distribution manifold by using an oxygen mask line or equivalent.
   5. Wait until all oxygen in the system has discharged.
   6. Remove the oxygen mask line or equivalent from the cabin distribution manifold.
   7. Turn off the regulator/valve from the logic and display panel (Basic or Avidyne option) or the softkey on the MFD (Garmin G1000 option).

WARNING

Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing or maintenance of the oxygen system.

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35-6. OXYGEN CYLINDER – See Figure 35 - 2.

a. **Removal of Oxygen Cylinder(s)**

1. Remove the oxygen regulator/valve as described in paragraph 35-10.
2. Cap or plug the open ends of tubing and fittings to protect against contamination.
3. Disconnect the grounding strap from the cylinder distribution manifolds.
4. Disconnect tubing from the center cylinder distribution manifold. Discard Seco seals.
5. Cap or plug the open ends of tubing and fittings to protect against contamination.
6. Loosen the ring clamp that secures the center cylinder to the cylinder support cradle in the wing.
7. Slide the center cylinder out through the ring clamp and through the opening in the footwell. Retain the ring clamp.
8. Remove the cylinder.
9. Disconnect tubing from the left cylinder distribution manifold and cap or plug the open ends of tubing and fittings to protect against contamination. Discard Seco Seals.
10. Loosen the ring clamp that secures the cylinder to the cylinder support cradle in the wing.
11. Slide the cylinder forward until its base clears the rear cradle. The cylinder may now be maneuvered into the center cylinder support cradle and out through the aircraft footwell. Retain the ring clamp.
12. Repeat steps 9 through 11 for the right cylinder.

b. **Installation of Oxygen Cylinder(s)**

1. Insert the right cylinder, base first, through the opening in the aircraft footwell and maneuver it into the right cradle. Ensure the base of the cylinder seats into the rear of the cylinder cradle.
2. Slide the ring clamp over the top of the cylinder and under the lip of the forward cylinder cradle.
3. Insert the left cylinder, base first, through the opening in the aircraft footwell and maneuver it into the left cradle. Ensure the base of the cylinder seats into the rear of the cylinder cradle.
4. Slide the ring clamp over the top of the cylinder and under the lip of the forward cylinder cradle.
5. Insert the center cylinder, base first, through the opening in the aircraft footwell and maneuver it into the center cradle. Ensure the base of the cylinder seats into the rear of the cylinder cradle.
6. Connect the tubing between the left cylinder distribution manifold and the center cylinder distribution manifold. Use new Seco seals.
7. Connect tubing between the right cylinder distribution manifold and the center cylinder distribution manifold. Use new Seco seals.
8. Install the oxygen regulator/valve as described in paragraph 35-10.
9. Adjust the position of the cylinders within the cylinder cradle until the lower lip of the forward cradle is just aft of the sloped top of the cylinder.
10. Tighten the ring clamps and torque 25 to 28 in-lbs.
11. Connect the tubing between the center cylinder distribution manifold and the oxygen regulator/valve.
12. Reconnect the grounding strap to the cylinder distribution manifolds.
13. Charge and test the system for leakage per paragraph 35-14.

14. If leakage is found, discharge the oxygen and reinstall the tubing and fittings until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary.
15. Test the system for flow rate per paragraph 35-15.
16. Purge the system as described in paragraph 35-4.
17. Replace the rear seat footwell and carpet. See Chapter 25 for detailed instructions.
18. Service the system as described in Chapter 12.

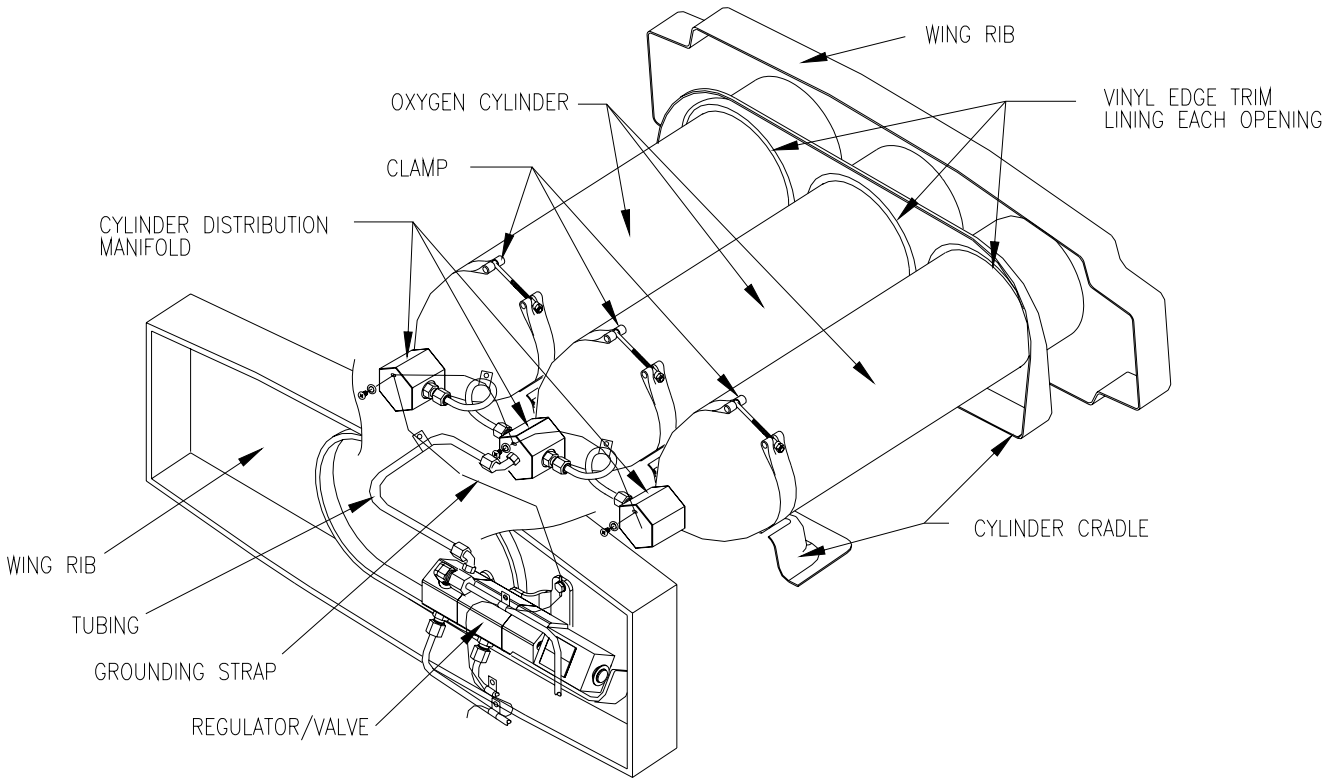

Figure 35 - 2 Oxygen Cylinders

c. **Servicing and Maintenance –** The oxygen cylinders must be inspected whenever malfunction occurs, and hydrostatically tested every 5 years.
   1. Concurrent with hydrostatic testing of the cylinders, all lines and connections should be carefully examined for leaks and abrasions, scratches, or any other damage due to vibration or looseness. Cylinder cradles and ring clamps should be examined for security and to ascertain that they do not damage the cylinder in any way. Check clamps for tightness and retorque as necessary.
   2. The date of hydrostatic testing must be metal stamped on the shoulder of each cylinder. A cylinder should never be refilled if it is near the expiration point of the most recent date stamp. At the time of hydrostatic testing, all cylinders should be examined for evidence of scratches, abrasions, bulges, or mishandling per Compressed Gas Association Pamphlet, “Standards for Visual Inspection of Compressed Gas Cylinders (CGA Pamphlet C-6)”. If cylinder damage is suspected, properly dispose of the cylinder and replace with new.

WARNING

Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing or maintenance of the oxygen system.

35-7. TUBING

a. **General –** All internal surfaces of tubing, hose assemblies, and fittings must be kept clean, and free of dust and petroleum contaminants. External and internal threads must be protected from contamination and damage prior to use in assembly. Regulators, gauges, and valves should have all open ports closed with oxygen clean and non-shredding polyethylene plugs or caps. Caps and plugs should be green or of a color to distinguish them from caps or plugs used on oily equipment. Where a cap or plug cannot be used, the opening should be sealed in a manner that insures protection.

1. Masking or self-sealing tapes, paper, pressure sensitive materials, hot dip coatings, rubber, soft plastic or adhesives should not be used. High density plastic closure materials such as Teflon or polyethylene are acceptable.
2. All items are to be shipped in clean plastic bags with nomenclature tags in or securely attached to the package and shipping container. Nomenclature tags must include title, part number, other information as required and include the following or similar statement: "CLEANED FOR OXYGEN SERVICE. DO NOT ALLOW PETROLEUM CONTAMINANTS OF ANY KIND TO BE USED OR STORED ON OR ABOUT THIS CONTAINER." Nomenclature tags must be attached to the package or shipping container in such a way as to allow identification of the item without breaking the container or tag.
3. No component of the oxygen system is to be internally exposed for more than 10 minutes. If exposed for more than 10 minutes the component must be cleaned.
4. Any component of the oxygen system that requires cleaning, should have contamination blown out using canned or dry filtered air and be cleaned with oxygen system cleaning compound conforming to MIL-C-8638, Heavy Duty Degreaser C by Micro Care, trichloroethylene, acetone, or hexane. These solvents must be fresh; not having been used for any other purpose. Blow the tubing and fittings clean and dry with a steam of nitrogen (Heavy Duty Cleaner C does not require this step). Visually inspect immediately for cleanliness, and cover with clean protective plugs, caps and bags.

NOTE

All oxygen tubing flare fittings are to be installed with Christo-Lube MCG111 (MIL-PRF-27617) lubricant. All pipe thread joints, on tapered pipe threads only and not on flare fittings, must be wrapped with teflon tape conforming to MIL-T-27730A.

b. **O-Ring Installation –** Use only O-rings that conform to MS 28778 which specifies 90 durometer nitrile per MIL-P-5510.

1. Prior to installation, visually inspect the O-ring for cuts, nicks, and excessive flashing. Most leaks are caused by damaged O-rings. No lubrication is allowed on the O-rings in the oxygen system. Protect O-rings from damage from threads during assembly with a sleeve temporarily installed over the threads.

2. Ensure the O-ring port is free of contamination prior to assembly. If necessary, remove contaminants with oxygen system cleaning compound conforming to MIL-C-8638. On tapered pipe thread joints only, wrap the threads with teflon tape conforming to MIL-T-27730A. When installing O-ring sealed fittings, turn the fitting into the straight thread boss until the metal back-up washer contacts the face of the boss, forcing the O-ring into the port. For elbows and tees, back off the fitting one turn and position the fitting to proper angular direction. Hold the fitting in position while tightening the locknut.
3. Torque steel to steel O-ring port fittings 95 to 140 in.-lbs. (7.9 to 11.7 ft.-lbs.) dry torque value.
4. Torque steel, aluminum, or brass to any combination of steel, aluminum, or brass O-ring port fittings 70 to 105 in.-lbs. (5.8 to 8.8 ft.-lbs.) dry torque value.
5. When attaching stainless steel to stainless steel or stainless steel to aluminum, use Christo-Lube MCG111 (MIL-PRF-27617) lubricant on the fitting threads. To prevent contamination, do not apply the lubricant to the tapered seat. Use the lower torque value indicated above when lubricant is used.

c. **Removal of Tubing**
1. Discharge the oxygen system per paragraph 35-5.
2. Remove the rear seats, the center overhead light panel (COLP), aft overhead side panel right (AOSPR), middle lower side panel right (MLSPR), baggage lower side panel right (BLSPR), carpeting on floor and hat rack, and rear seat footwell as required. See Chapter 25 for detailed instructions.
3. Remove clamps and grounding strap from the portion of tubing to be removed. Retain hardware. See Figure 35 - 3.
4. Cut tie wraps securing tubing.
5. Unscrew the flare nut at the end of the tubing and slide it away from the flared end of the fitting.
6. Disconnect the tubing and remove the Seco seal. Discard the Seco seal.
7. Temporarily cap or plug the end of the fitting so that contamination will not enter the oxygen system.
8. Repeat steps 3 through 7 at the other end of the tubing.
9. Remove the tubing.

b. **Installation of Tubing**
1. Apply Christo-Lube MCG111 (MIL-PRF-27617) lubricant to the male fitting threads. Do not apply compound to the first thread of the fitting or to the tapered seat of the tubing.
2. Install the Seco seal (dry, no lubricant permitted) onto the male portion of fitting.
3. Thread the flare nut onto the male fitting several turns with fingers until connection is snug. If the nut cannot be tightened with fingers, disassemble and correct the problem to prevent damage to the Seco seal.
4. Torque the flare nut to the value in Figure 35 - 4.
5. Wait 15 seconds for the Seco seal to compress and deform.
6. Do not loosen the flare nut. Torque the flare nut again to the value in Figure 35 - 4.
7. Charge and test the system for leakage per paragraph 35-14.
8. If leakage is found, discharge the oxygen and reinstall the tubing until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by

overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary. The Seco seal must be replaced if the joint is disassembled.

9. Fittings equipped with O-rings are installed as indicated in paragraph 35-7.b. Remove old O-rings and replace with new.
10. Test the system for flow rate per paragraph 35-15.
11. Secure the tubing to tie blocks with tie wrap.
12. Reinstall the grounding strap across all affected fittings. See Figure 35 - 3.
13. Mark all tubing with breathing oxygen label tape, per MIL-STD-595B and MIL-STD-1247C, located within 3 inches of all fittings and at intervals along the lines not to exceed 48 inches.
14. Purge the system as described in paragraph 35-4.
15. Reinstall the rear seat footwell, carpeting on the floor and the hat rack, and the interior panels. See Chapter 25 for detailed instructions.
16. Service the system as described in Chapter 12.

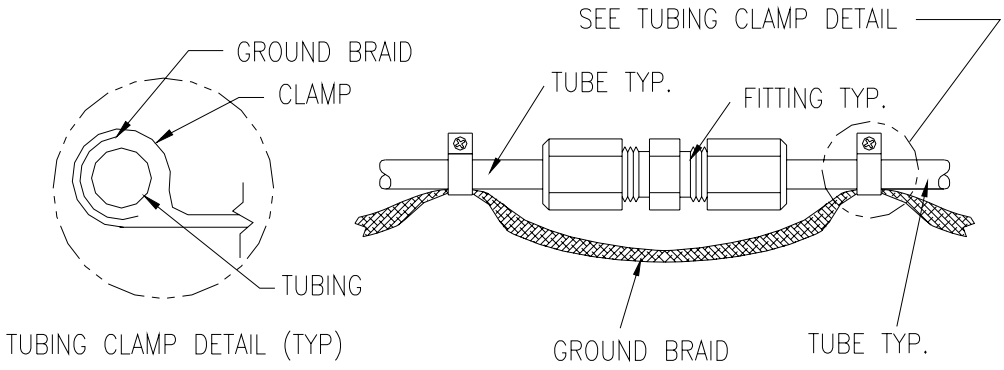

Figure 35 - 3 Typical Grounding Strap Installation

Tube Flare Nut Torque (with Seco seals)					
Size	Tube O.D.	Material and Torque Specification			
		Tube Material	Fitting Material	Threads PER MIL-S-8879	*Inch Pounds
- 2	0.125	SST	Steel	.3125-24UNJF-3B	35-50
		Aluminum	Steel	.3125-24UNJF-3B	22-30
- 3	0.188	SST	Steel	.375-24UNJF-3B	75-100
		Aluminum	Steel	.375-24UNJF-3B	30-45
- 4	0.250	SST	Steel	.4375-20UNJF-3B	115-150
		Aluminum	Steel	.4375-20UNJF-3B	40-60

* Wet torque values shown

Figure 35 - 4 Flare Nut Torque with Seco Seals

c. **Servicing and Maintenance –** The oxygen system tubing must be inspected every 5 years or whenever malfunction occurs.

WARNING

Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing or maintenance of the oxygen system.

35-8. FILL PORT – See Figure 35 - 5.

a. **Removal of Fill Port**

1. Discharge the oxygen system per paragraph 35-5.
2. Remove the hat rack carpet. See Chapter 25 for detailed instructions.
3. Remove the baggage bulkhead access panel by unscrewing the attachment screws.
4. Disconnect the tubing.
5. Temporarily seal the end of the fitting so that contamination will not enter the oxygen system.
6. Remove the three pan head screws, nuts, and washers that secure the fill port to the baggage bulkhead.
7. Remove the fill port, screw on cap, and bezel.

b. **Installation of Fill Port**

1. Insert the fill port into the baggage bulkhead.
2. Place the bezel over the fill port and align all screw holes with the holes in the baggage bulkhead.
3. Install three pan head screws, washers, and nuts ensuring that the screw on cap chain and the grounding strap are attached to the fill port. Tighten the screws snug.
4. Attach tubing.
5. Charge and test the system for leakage as described in paragraph 35-14.
6. If leakage is found, discharge the oxygen and reattach the tubing until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary.
7. Purge the system as described in paragraph 35-4.
8. Reinstall the baggage bulkhead access panel.
9. Reinstall the hat rack carpet. See Chapter 25 for detailed instructions.
10. Service the system as described in Chapter 12.

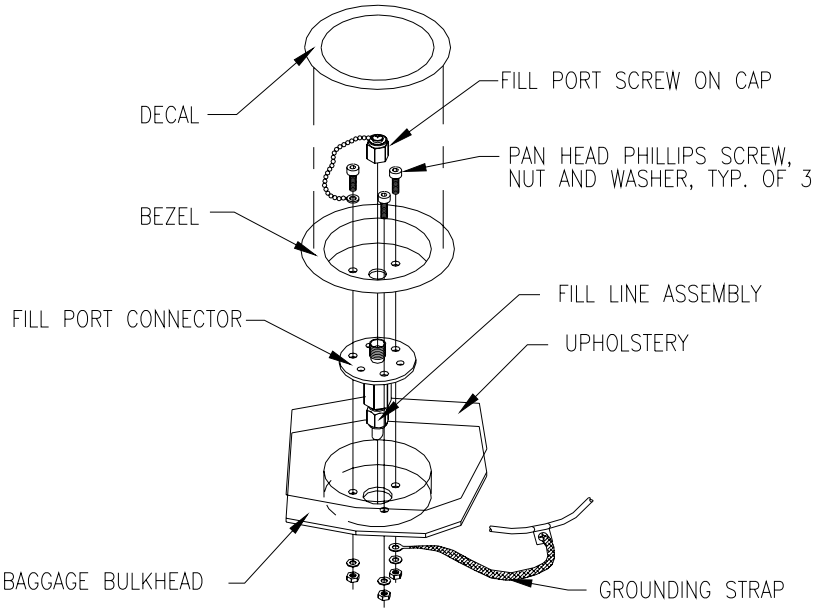

Figure 35 - 5 Fill Port

c. **Servicing and Maintenance –** The fill port must be inspected every 5 years or whenever malfunction occurs.

WARNING

Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing or maintenance of the oxygen system.

35-9. OVERFILL/OVERPRESSURE BURST VALVE ASSEMBLY – See Figure 35 - 6.

a. **Removal and Installation of the Burst Cartridge –** Replacement of the burst cartridge is only required when the cartridge has burst due to an overfill or overpressure of the oxygen system.
   1. Locate the exit orifice of the burst valve assembly in the underside of the wing.
   2. Using a hex wrench inserted into the orifice, unscrew the burst cartridge from the burst valve assembly.
   3. Installation is the reverse of removal.
   4. Install a green plug to cap the burst valve at the opening in the underside of the wing.
   5. Purge the system per paragraph 35-4.
   6. Service the system per Chapter 12.

b. **Removal of Overfill/Overpressure Burst Valve Assembly**
   1. Discharge the oxygen system per paragraph 35-5.
   2. Remove the rear seat carpet and footwell. See Chapter 25 for detailed instructions.
   3. Disconnect tubing and remove the Seco seals. Discard the Seco seals.
   4. Temporarily cap or plug the end of the tubing so that contamination will not enter the oxygen system.
   5. Remove the grounding strap from the burst valve. Retain hardware.

6. If present, remove the green plug that caps the burst valve at the underside of the wing.
7. From the underside of the wing, unscrew the four nylon screws that hold the burst valve to the fuselage.
8. Remove the valve.
9. Temporarily cap or plug the end of the fitting or tubing so that contamination will not enter the oxygen system.

c. **Installation of Overfill/Overpressure Burst Valve Assembly**

1. Align all screw holes in the burst valve with the holes in the fuselage.
2. With an assistant, install the four nylon screws and torque to 5 in-lbs.
3. Attach tubing.
4. Charge and test the system for leakage per paragraph 35-14.
5. If leakage is found, discharge the oxygen and reinstall the tubing until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary. The Seco seal must be replaced if the joint is disassembled.
6. Test the system for flow rate per paragraph 35-15.
7. Reinstall the grounding strap to the burst valve.
8. Purge the system per paragraph 35-4.
9. Install the green plug to cap the burst valve at the opening in the underside of the wing.
10. Reinstall the rear seat footwell and carpet. See Chapter 25 for detailed instructions.
11. Service the system per Chapter 12.

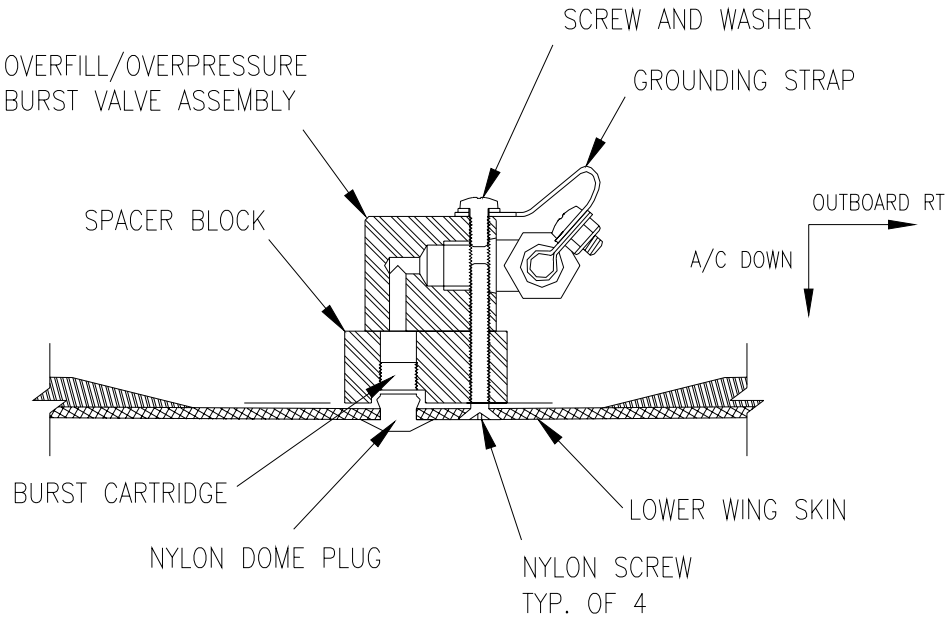

Figure 35 - 6 Overfill/Overpressure Burst Valve Assembly

d. **Servicing and Maintenance –** The overfill/overpressure burst valve must be inspected every 5 years or whenever malfunction occurs.

WARNING

Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing or maintenance of the oxygen system.

35-10. OXYGEN REGULATOR/VALVE – See Figure 35 - 7.

a. **Removal of Regulator/Valve**

1. Discharge the oxygen system per paragraph 35-5.
2. Remove the rear seat carpet and footwell. See Chapter 25 for detailed instructions.
3. Remove the clamp and grounding strap from the tubing as required. Retain hardware.
4. Disconnect the electrical connector.
5. Disconnect all tubing from the regulator/valve.
6. Discard the Seco seals.
7. Cap or plug the open ends of tubing and the regulator/valve inlets and outlets to protect against contamination.
8. Remove the two hex head cap screws, washers, and nuts holding the regulator/valve to the wing rib.
9. Remove the regulator/valve.

b. **Installation of Regulator/Valve**

1. Align screw holes in the regulator/valve with the holes in the wing rib.
2. Install the two hex head cap screws, washers, and nuts ensuring that the grounding strap is attached to the upper screw. Tighten 30 to 36 in-lbs.
3. Attach fittings and tubing.
4. Install the clamps and grounding strap.
5. Fittings equipped with O-rings are installed as indicated in paragraph 35-7.b. Remove the old O-ring and replace with new.
6. Charge and test the system for leakage per paragraph 35-14.
7. If leakage is found, discharge the oxygen, and reinstall the tubing until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary. The Seco seal must be replaced if the joint is disassembled.
8. Test the system for flow rate per paragraph 35-15.
9. Purge the system per paragraph 35-4.
10. Replace the rear seat footwell and carpet. See Chapter 25 for detailed instructions.
11. Service the system as described in Chapter 12.

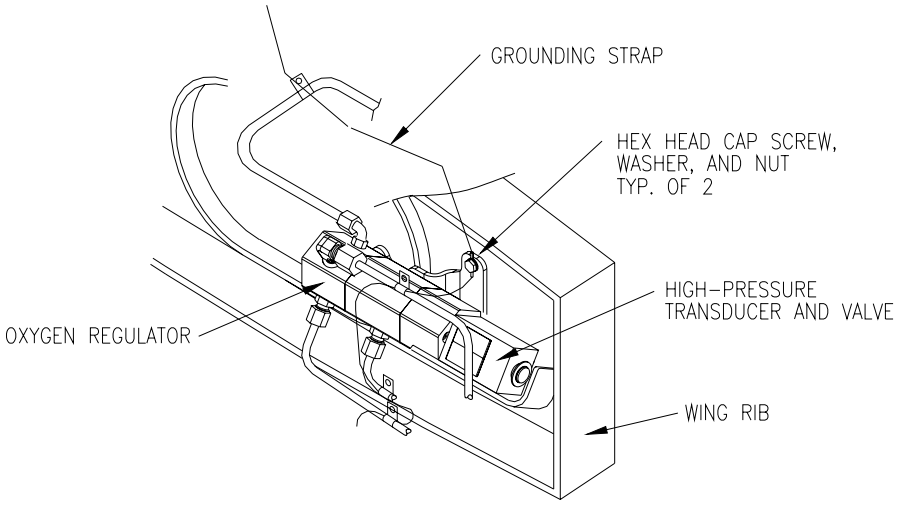

Figure 35 - 7 Regulator/Valve Assembly

c. **Servicing and Maintenance –** The regulator/valve must be inspected whenever malfunction occurs, and overhauled every 5 years. During overhaul replace o-rings and verify the regulator pressure setting.

WARNING

Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing or maintenance of the oxygen system.

35-11. MANUAL LOW-PRESSURE OXYGEN SHUTOFF VALVE – See Figure 35 - 8 Figure 35 - 9, and Figure 35 - 10.

a. **Removal of the Manual Low Pressure Oxygen Shutoff Valve**

1. Turn off the low-pressure oxygen supply to the cabin distribution manifold using the switch on the logic and display panel (Basic or Avidyne option) or the softkey on the MFD (Garmin G1000 option).
2. Remove the forward center overhead light panel per Chapter 25.
3. Remove the clamps and grounding strap as required. Retain hardware.
4. Disconnect the flexible oxygen hose at the quick disconnect bulkhead connector. See Figure 35 - 10.
5. Remove the nuts and switch cover attaching the shutoff valve to the overhead light panel.
6. Disconnect the tubing and remove the Seco seals. Discard the Seco seals.
7. Temporarily cap or plug any open fittings or tubing so that contamination will not enter the oxygen system or the valve.
8. Remove the shutoff valve.

b. **Installation of the Manual Low-Pressure Oxygen Shut-Off Valve**

1. Install the valve with the toggle facing forward, and port nearest mounting threads on the forward side. It may be necessary to remove the toggle pivot pin and rotate the toggle 180º to have the valve in the open position with the toggle facing forward. When the valve toggle is placed in the neutral position (straight up) and released, it should default to the closed (aft) position.

2. Place the switch cover over the shut-off valve toggle switch, and attach the assembly to the overhead light panel using the nuts. Apply Loctite 242 to the nuts before installation.
3. Attach the tubing and the flexible oxygen hose to the shutoff valve.
4. Connect the flexible oxygen hose to the quick disconnect bulkhead connector. See Figure 35 - 10.
5. Install the clamps and grounding strap.
6. Charge and test the system for leakage as described in paragraph 35-14.
7. If leakage is found, reinstall the tubing until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary. The Seco seal must be replaced if the joint is disassembled.
8. Test the system for flow rate per paragraph 35-15.
9. Reinstall the forward center overhead light panel per Chapter 25.
10. Service the system as described in Chapter 12.

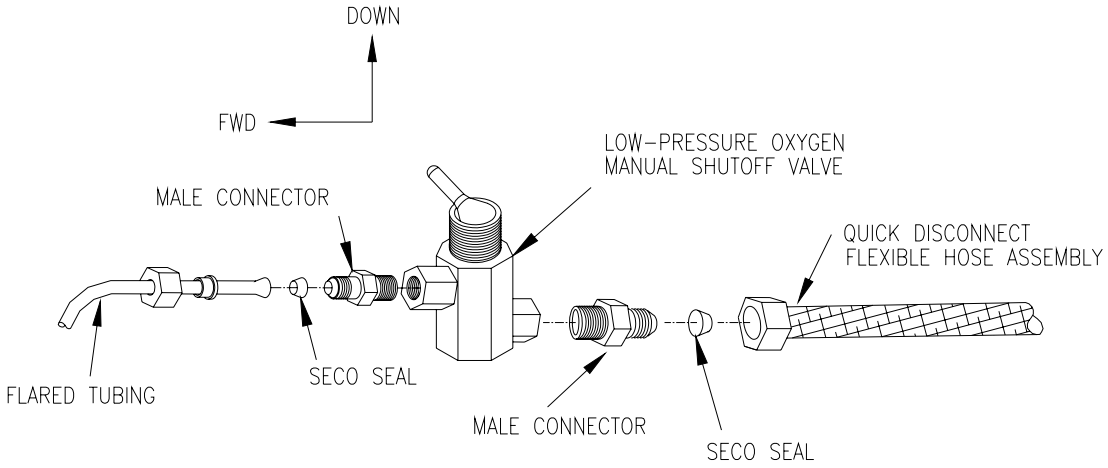

Figure 35 - 8 Manual Low-Pressure Oxygen Shutoff Valve

c. **Servicing and Maintenance –** The shutoff valve must be inspected every 5 years or whenever malfunction occurs.

WARNING

Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing or maintenance of the oxygen system.

35-12. CABIN DISTRIBUTION MANIFOLD – See Figure 35 - 9 and Figure 35 - 10.

a. **Removal of Cabin Distribution Manifold**

1. Turn off the low-pressure oxygen supply to the cabin distribution manifold using the switch on the logic and display panel (Basic or Avidyne option) or the softkey on the MFD (Garmin G1000 option).

2. Remove the forward center overhead light panel per Chapter 25.
3. Disconnect the electrical connector from the manifold.
4. Remove the clamps and grounding strap as required. Retain hardware.
5. Disconnect the flexible oxygen hose at the quick disconnect bulkhead connector. See Figure 35 - 10.
6. Disconnect the tubing from the manifold and remove the Seco seals. Discard the Seco seals.
7. Temporarily cap or plug all open tubing and fittings so that contamination will not enter the oxygen system or the manifold.
8. Remove the four pan head screws affixing the cover plate and the cabin distribution manifold support bracket to the overhead light panel.
9. Remove the four cap screws and washers holding the manifold to the manifold support bracket.
10. Remove the manifold.

b. **Installation of the Cabin Distribution Manifold**

1. Align the screw holes in the manifold with the holes in the manifold support bracket and install the four cap screws and washers. Torque the screws 20 to 25 in.-lbs.
2. install the four pan head screws affixing the cover plate and the cabin distribution manifold support bracket to the overhead light panel. Tighten the screws snug.
3. Attach the tubing.
4. Connect the flexible oxygen hose to the quick disconnect bulkhead connector. See Figure 35 - 10.
5. Connect the electrical connector.
6. Install the clamps and grounding strap.
7. Charge and test the system for leakage as described in paragraph 35-14.
8. If leakage is found, reinstall the tubing fittings until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary. The Seco seal must be replaced if the joint is disassembled.
9. Fittings equipped with O-rings are installed as indicated in paragraph 35-7.b. Remove the old O-rings and replace with new.
10. Test the system for flow rate per paragraph 35-15.
11. Reinstall the forward center overhead light panel per Chapter 25.
12. Service the system as described in Chapter 12.

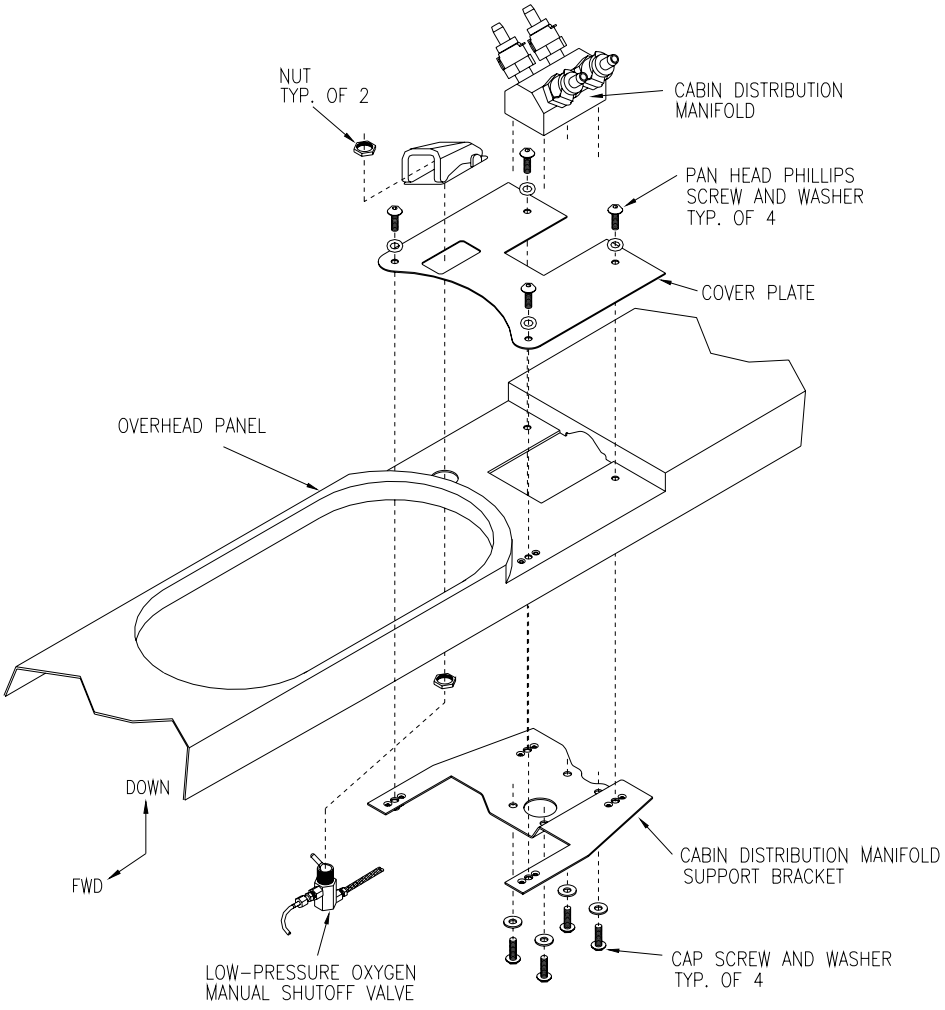

Figure 35 - 9 Cabin Distribution Manifold

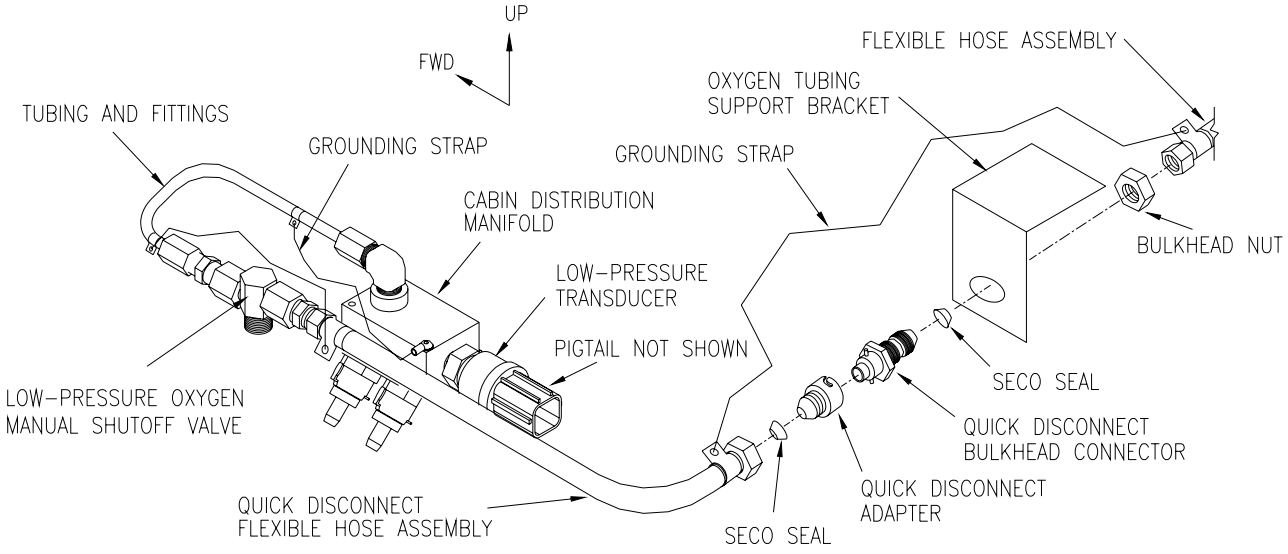

Figure 35 - 10 Cabin Distribution Manifold Assembly

c. **Servicing and Maintenance –** The cabin manifold must be inspected every 5 years or whenever malfunction occurs.

WARNING

Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing or maintenance of the oxygen system.

35-13. LOGIC AND DISPLAY UNIT (BASIC OR AVIDYNE OPTION ONLY)– See Figure 35 - 11.

a. **Removal of the Logic and Display Unit**
   1. Remove the kidney panel as described in Chapter 25. This will give access to the Logic and Display Unit.
   2. Disconnect the 25 pin connector from the logic and display unit.
   3. The logic and display unit display panel has four threaded 6-32 studs that protrude through mating holes into the backside of the instrument panel. Four self-locking 6-32 nuts with washers, accessible from the backside of the instrument panel, screw onto these studs to secure the display panel to the instrument panel. Removal of the nuts and washers allows the display panel to slide out for access to, and disconnection of, the bayonet plugs and ribbon cable connector. Make note or label each plug for reconnection to its proper location.
   4. Remove the three pan head screws and washers securing the Logic and Display Unit to the cross member at the back of the instrument panel and remove the unit.

b. **Installation of Logic and Display Unit –** Installation is the reverse of removal.

c. **Servicing and Maintenance –** The oxygen system logic and display unit requires no service or maintenance but must be inspected annually, or whenever malfunction occurs.

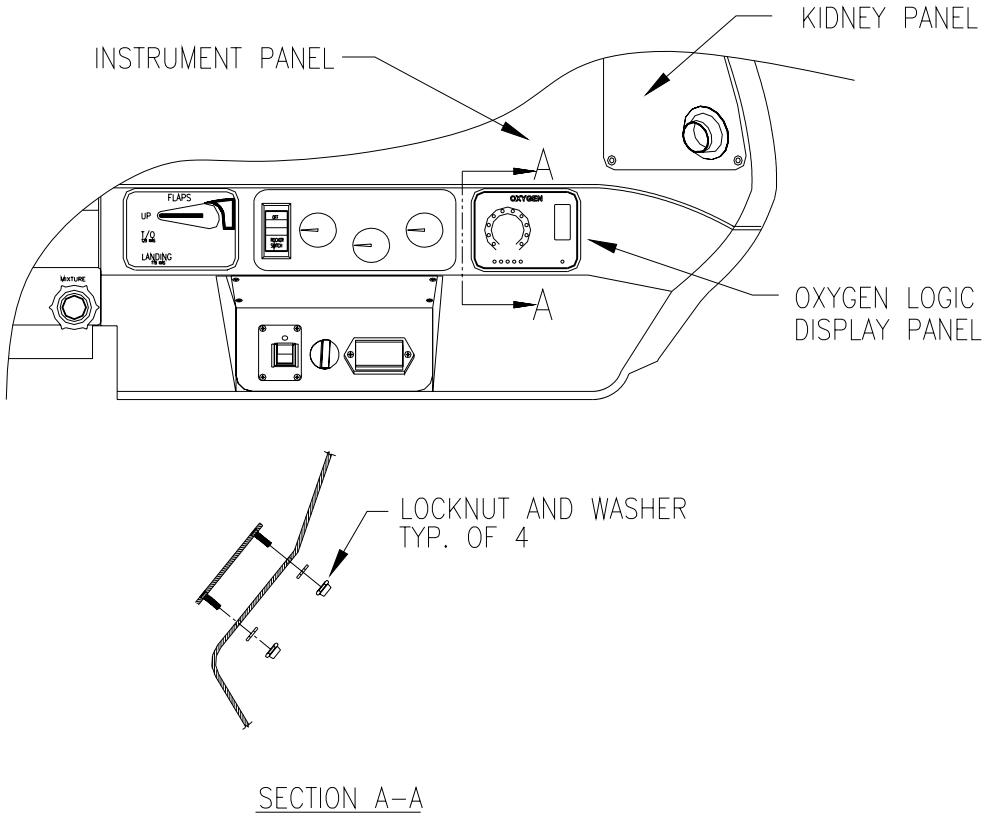

Figure 35 - 11 Logic Display Panel

35-14. LEAKAGE TESTING

Any removal and replacement of components or tubing in the oxygen system requires that leakage testing be performed. Apply oxygen safe leak detection compound conforming to MIL-L-25567 to the affected fittings and connections. Observe the tested area for a minimum of 30 seconds. Any formation of bubbles or deformation of the leak detection compound constitutes failure of the test.

a. Verify that the solenoid valve operates properly when the oxygen control panel is electrically switched on and off.
b. If complete evacuation of oxygen from the system was required, perform the following tests:
    1. Charge the system to 50 psig with oxygen and verify there are no leaks.
    2. Increase the system pressure to 400 psig and verify there are no leaks.
    3. Increase the system pressure to 1,000 psig and verify there are no leaks.
    4. Increase the system pressure to 2,000 psig and verify there are no leaks.
c. Any repair work to the low-pressure portion of the oxygen system from the regulator/valve to the cabin distribution manifold may not require complete evacuation of oxygen from the system. In this case test the system at its current pressure and verify there are no leaks.
d. Remove all leak test compound from fittings and connections after successful completion of the leak test.

35-15. FLOW TESTING

a. Connect an oxygen flow meter to one of the overhead cabin manifold ports.
b. Toggle the oxygen logic and display panel switch to ON.
c. Immediately check the oxygen flow and verify that it is a minimum of 0.43 cu. ft. per minute. Flow of less than 0.43 cu. ft. per minute is indication of a constriction in the system or a malfunction of one or more system components.
d. Immediately switch the oxygen logic and display panel switch to OFF.
e. Disconnect the oxygen flow meter.

35-16. DISCONNECTION AND RECONNECTION FOR WING REMOVAL– See Figure 35 - 12.

a. **Disconnection**
   1. Discharge the oxygen system as indicated in 35-5.
   2. Remove the clamps and grounding strap from the tubing at the wing to fuselage connection. Retain hardware.
   3. Cut the tie wrap securing the tubing to the fuselage stand-off.
   4. Disconnect the tubing at the wing to fuselage connection, and remove the Seco seals. Discard the Seco seals.
   5. Temporarily cap or plug the tubing and fittings so that contamination will not enter the oxygen system.
   6. Proceed with wing removal per Chapter 57.

b. **Reconnection**
   1. Connect the tubing at the wing to fuselage connection.
   2. Charge and test the system for leakage as described in paragraph 35-14.
   3. If leakage is found, discharge the oxygen and reinstall the tubing until the leakage is corrected. Do not attempt to correct any leakage or misalignment of a joint by overtorquing. Instead, disassemble and check for nicks, burrs, dirt, etc. Reassemble using new parts if necessary. The Seco seal must be replaced if the joint is disassembled.
   4. Secure the tubing to fuselage stand-off with a tie wrap.
   5. Reinstall the grounding strap across the fittings at the wing to fuselage connection.
   6. Proceed with wing installation per Chapter 57.
   7. Service the oxygen system as described in Chapter 12.

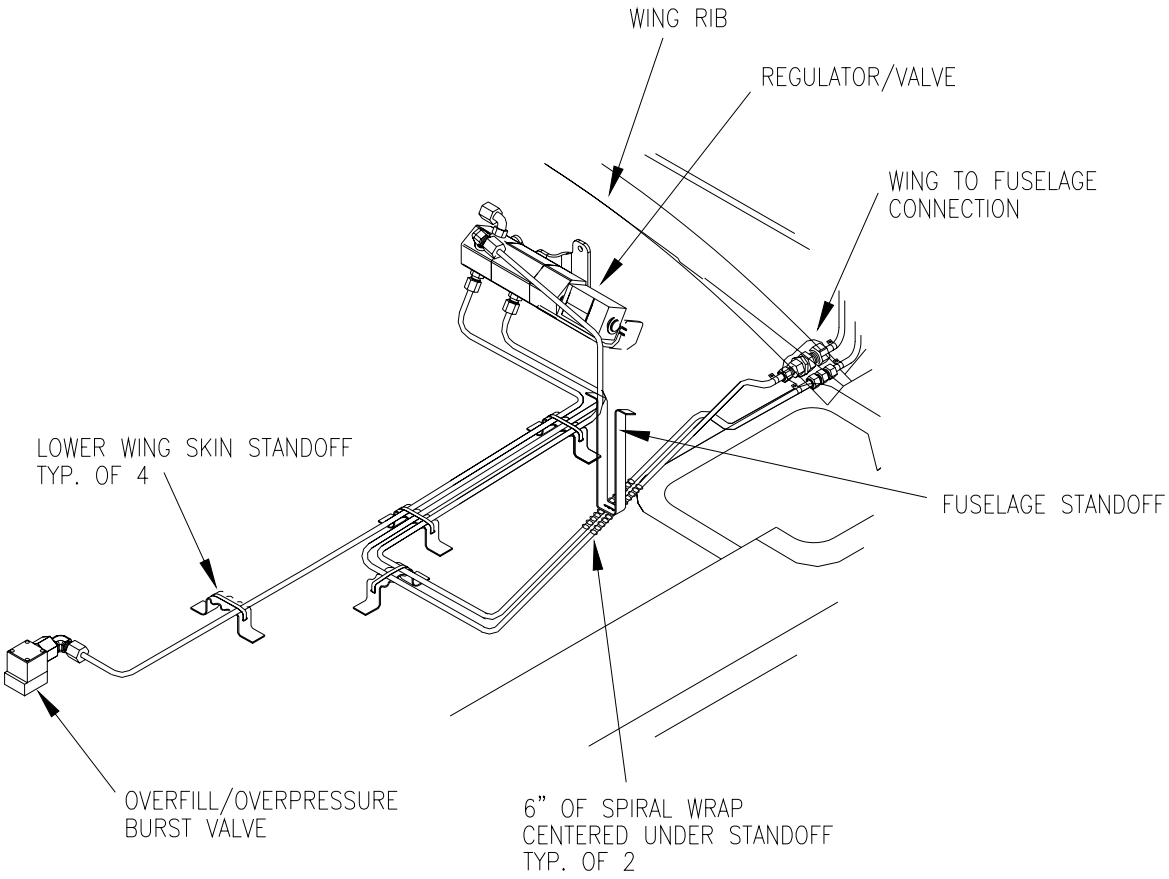

Figure 35 - 12 Wing to Fuselage Connection

WARNING

Refer to SAFETY GUIDELINES FOR SERVICING THE OXYGEN SYSTEM before attempting any servicing or maintenance of the oxygen system.

35-17. OXYGEN SYSTEM TROUBLESHOOTING

Condition	Probable Cause	Remedy
Excessive leakage of system pressure.	Fill Port leaking.	Replace the fill port
	Overfill/overpressure burst valve blown.	Replace the overfill/overpressure burst cartridge.
	Leak in lines.	Check tubing, fittings, and connections and repair or replace as necessary. See paragraph 35-17. a. below.
No low pressure reading at the logic and display panel (Basic or Avidyne option) or on the MFD (Garmin G1000 option)	System not charged.	Replenish the oxygen supply.
	Defective logic and display unit or panel.	Replace the logic and display unit or panel.
	Defective low pressure transducer in cabin distribution manifold.	Replace the low pressure transducer.
No system oxygen quantity reading at logic and display panel (Basic or Avidyne option) or on the MFD (Garmin G1000 option)	System not charged.	Replenish the oxygen supply.
	Defective logic and display unit or panel.	Replace the logic and display unit or panel.
	Defective high pressure transducer and switch in the regulator/valve.	Replace the regulator/valve.
Oxygen does not flow from the masks when the system is turned on.	System not charged.	Replenish oxygen supply.
	Oxygen mask defective.	Overhaul or replace mask.
	Defective low-pressure shutoff valve	Replace the low-pressure shutoff valve.
	Defective high pressure transducer and switch in the regulator/valve.	Replace the regulator/valve.

a. **Oxygen System Leaks –** With the oxygen system turned off, leakage is most likely to occur in the high pressure side of the system.

1. Remove the hat rack carpet and cushion, the aft baggage bulkhead access panel, the baggage lower side panel right (BLSPR), and the middle lower side panel right (MLSPR). See Chapter 25 for specific instructions.

2. Perform leakage testing per paragraph 35-14 on the fill port, tubing, fittings, and connections accessible in this area of the aircraft.
3. Repair any leaks and perform leakage testing. If no leaks are found in this area or the system is still losing pressure after leaks are repaired, the remainder of the high pressure side of the system must be exposed and tested as indicated below.
4. Remove the flap and aileron push rod access panel on the underside of the plane, and the aft cabin footwell to gain access to the overfill/overpressure burst valve and the oxygen regulator/valve.
5. Perform leakage testing per paragraph 35-14 on the overfill/overpressure burst valve, oxygen regulator/valve, oxygen cylinders, tubing, fittings, and connections
6. Repair any leaks and perform leakage testing.
7. Upon successful completion of leakage testing replace all access panels, the footwell, interior panels, hat rack and carpeting.
8. Service the oxygen system as described in Chapter 12.

35-18. ANNUAL INSPECTION WITH GARMIN G1000 SYSTEM – If the Garmin G1000 system is installed perform the following steps in addition to regular annual inspection of the oxygen system.

1. Turn on both batteries, and the avionics to validate the Oxygen system.
2. If the oxygen system is charged beyond 250 PSI no oxygen quantity annunciation should be displayed. Note: the oxygen system must have some charge in order to be verified by these procedures.
3. Plug an oxygen mask with a flow indicator into the distribution manifold. On the MFD, full screen EIS page, press the “oxygen” soft key, and turn the oxygen switch at the distribution manifold to ON.
4. The G1000 should issue and advisory for oxygen on (while on the ground), Check the oxygen outlet pressure is in the green band, and the mask is flowing oxygen.
5. Press the “oxygen” soft key on the G1000, the oxygen outlet pressure should decay to zero as the system runs out of air, and the mask should stop flowing.
6. Turn the oxygen switch at the distribution manifold to OFF, and then press the “oxygen” soft key on the G1000 to turn the oxygen on again.
7. Verify the oxygen outlet stays at zero, no oxygen is flowing to the mask and the G1000 annunciates oxygen pressure – no flow detected but the oxygen system is turned on.
8. Turn the oxygen switch at the distribution manifold to ON and verify the outlet pressure increases and the oxygen pressure annunciation on the G1000 clears.
9. Press the “oxygen” soft key, turning the oxygen system off, clearing all oxygen annunciations.

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CHAPTER 37

VACUUM

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37-1. GENERAL

a. The Cessna 350 does not have a vacuum system. All flight instruments are electrically powered.

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CHAPTER 39

ELECTRICAL PANELS

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39-1. GENERAL

Because there are a limited number of electrical panels in the airplane, and the topic is interrelated with equipment and furnishings, this subject is covered in Chapter 25, subsection 25-11-00.

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CHAPTER 51

STANDARD PRACTICES STRUCTURES

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51-1. GENERAL

This chapter covers non-structural repairs that may be performed in the field. Discussion and procedures include:

a. Repairable Aircraft Parts – A list of the field repairable parts as well as allowable repairs.
b. Damage Assessment – Damage area classification and determination of allowable repairs.
c. Repair Methods – Repair methods to be used after damage assessment.
d. Definitions – Material and procedure terminology definitions.

For structural repairs refer to the latest revision of the Cessna Field Structural Repair Manual, RX512100.

51-2. DEFINITIONS

a. **Controlled Surface –** For parts covered in this chapter, controlled surface is the exterior surface of the part which is painted or upholstered. During its initial cure at fabrication it sits against the mold surface and subsequently after cure will appear more smooth than the other surface.
b. **Uncontrolled Surface–** For parts covered in this chapter, uncontrolled surface is the interior surface of the part which is not painted or upholstered. During its initial cure at fabrication it is not in contact with the mold surface and subsequently after cure will not appear smooth. For the interior parts mounting hardware may exist on this surface.
c. **GTM –** Grain Transfer Mat. This mat allows the repair to have the same texture as leather or ultra-leather. It is to be made at least 24 hours in advance of the repair using Flexproof Compound and Catalyst.
d. **Chill Bar –** A clean, room temperature piece of metal which will allow a heated surface to be compressed and cooled.

51-3. REPAIRABLE AIRCRAFT PARTS

Part Name	Applicable Repairs
Wheel Pants	Paint, Laminate
Nose Fairing	Paint, Laminate
Leg Fairings	Paint, Laminate
Speed Buns	Paint, Laminate, Bonding
Wing Fillets	Paint, Laminate, Bonding
Aileron Fences	Paint, Laminate, Bonding
Stall Strips	Paint, Laminate, Bonding
Exterior Access Panels	Paint, Laminate
Interior Access Panels	Upholstery, Laminate
AC Ducting	Laminate, Bonding
Glare Shields	Laminate, Bonding, Upholstery
Horizontal Stabilizer Gap Seals	Paint, Laminate, Bonding
Elevator Fairings	Paint, Laminate, Bonding
Interior Panels	Upholstery, Laminate
Cabin Door Windows	Transparency

Part Name	Applicable Repairs
Passenger Windows	Transparency
Windshield	Transparency
Landing/Taxi Light Cover	Transparency
Position Light Covers	Paint, Transparency

51-4. DAMAGE ASSESSMENT

NOTE

During damage assessment it is possible to come across defects unrelated to the suspect damage being investigated. While these areas may or may not have been repaired they have been evaluated by Cessna Quality and Engineering personnel. These areas are to be classified as pre-dispositioned and are not to be considered during an assessment of damage. If you have any concerns please contact the Customer Service Department of Cessna for a history of dispositions on the part.

a. **Paint Damage** – Classified as damage to a painted surface which has removed paint but not damaged the composite part beneath it.

NOTE

Paint damage can be a sign of other types of damage. For example a crack in the paint in an area where two components are bonded together could indicate bond failure. In these cases it is advised to remove paint and primer to investigate.

b. **Mesh Damage –** To determine whether foil needs to be repaired, first evaluate which "Zone" of the aircraft the damage is in. See Figure 51 - 8. Antenna mounting pads and "covered areas" require separate review. A covered area is an area not exposed to direct lightning attachment. Foiled areas inside a cove, under an access panel, and under a static wick mounting pad, for example, are considered covered areas. Major direction is the greater dimension of a fore-to-aft or left-to-right measurement of the assembly's foil coverage. Minor direction is the lesser dimension of a fore-to-aft or left-to-right measurement of the assembly's foil coverage.

1. **Damage Not Requiring Repair**
   a) Zone 1 – The loss of expanded metal foil not greater than two inches average diameter, not to exceed 4 sq. in. per defect on a surface area. Combined losses not to exceed 10% of the minor direction or two inches, whichever is greater.
   b) Zone 2 or 3 – The loss of expanded metal foil not greater than three inches average diameter, not to exceed six sq. in. per defect on a surface area. Combined losses not to exceed 10% of the minor direction or three inches, whichever is greater.
   c) Antenna Mounting Pad – If greater than 50% of the antenna mounting pad area has visible foil, is somewhat flat, and there is electrical foil connection to both sides of the fuselage then no repair is necessary.
   d) Covered Area – If greater than 50% of the covered area has visible foil, then no repair is necessary
2. **Damage Requiring Repair**
   a) Zones 1, 2, and 3 – Any damage greater than that defined above as damage that does not require repair.

b) Antenna Mounting Pad
   1) If greater than 25%, but less than 50%, of the antenna mounting pad area has visible foil, and there is an electrical connection to both sides of the fuselage, then foil repair will be limited to the antenna mounting area.
   2) If less than 25% of the antenna mounting pad area has visible foil, or there is no electrical connection to both sides of the fuselage, a foil repair with 1 inch overlaps to existing foil (past the mounting pad area) will be required.

c) Covered Area
   1) If greater than 25%, but less than 50%, of the covered area has visible foil, then the foil repair can be limited to the covered area.
   2) If less than 25% of the covered area has visible foil, a full foil repair with 1 inch overlaps to existing foil (past the covered area) will be required.

c. **Laminate Damage** – Classified as damage to, or penetration of the fibers of the composite part. The two types of damage repairable through methods in this chapter are as follow:

1. **Surface Cuts and Abrasions –** Cuts, scratches, and abrasions that extend through the surface resin but not into the first layer of fabric shall be acceptable.
2. **Delamination of Plies –** This is defined as the separation of the plies in the composite part. Visually it will appear as a whitened or cloudy area within the composite part though it is important to note that the resin in the composite part may also appear white. Delamination between several different layers of plies is possible and will be denoted by contrasts in the whitened areas. If you suspect an area may be delaminated but cannot confirm this visually the following method known as a Tap Test may be used.
   a) Using a coin (quarter or larger), small metallic disk or a small tapping hammer lightly tap the surface repeatedly near the delamination. Tap lightly but hard enough to hear the impact against the composite part. Being in a quiet environment will help.
   b) While performing this, move the tapping to the area in which you suspect may be delaminated.
   c) Compare the difference in tone between the suspect and surrounding areas. If the tone in the suspect area does not ring like the tone in the surrounding area then it is likely that the area is delaminated.
   d) By using the preceding steps you will be able to determine how large the delaminated area is.
3. **Delamination of Plies around Attachment Holes** – This has the same definition of Delamination of plies above but occurs specifically around attachment holes. This is limited to the elevator fairings and interior/exterior access panels of the part.
4. **Fracture of Plies –** This is defined as damage either occurring to or penetrating the fibers of the composite part. It will appear as a break in the plies where the fibers of the composite part have been compromised. It is common to see the reinforcement fibers of the ply during a fracture.

NOTE

With impact damage to the laminate, it is common to see a fractured area surrounded by an area where delamination of the plies has occurred. In these cases, it is recommended to use the tap test method to locate the areas surrounding the fracture which are delaminated.

d. **Repair Defects –** Any of the following discrepancies which may occur due to improper repair techniques:
   1. **Tackiness**: Tackiness is defined as a soft or fluid area of resin that has not fully cured. The entire laminate shall be cured and free of tackiness. This is caused by an inadequate cure of the repair.
   2. **Wrinkles:** A wrinkle is defined as an imperfection in a laminate that has the appearance of a fold or crease in one or more layers of reinforcing fibers. There are two types of wrinkling:
      a) **Protruding wrinkles:** Wrinkling which appears as a protrusion of the reinforcement fibers from the surface of the part. This is caused by inadequate compaction of the plies during lay-up.
      b) **Intruding wrinkles:** Wrinkling where the reinforcement fibers fold over themselves but do not protrude from the surface of the part. This is caused by folding a ply over itself during lay-up.

e. **Pinching:** Pinching is classified as a deformation of the plies occurring within a part radius. Pinching appears as an area where the plies do not sit flush and appear to pinch at the part radius. The phenomena of plies doing this can also be confused with an area of excess resin at the radius.
   1. Pinching of fibers is shown in Figure 51 - 1. Typically pinching occurs in corners and the usual indications are an area of resin build up. Pinching is acceptable provided it is within the limits defined under “Pinching” in Figure 51 - 3 and Figure 51 - 6. Pinching of fibers is caused by inadequate compaction of the plies during lay-up.

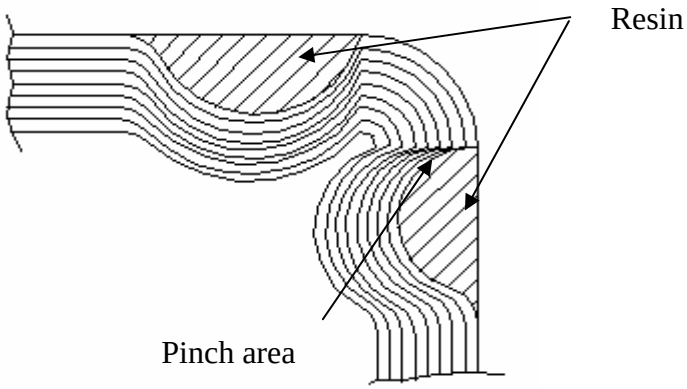


Figure 51 - 1 Fiber Pinching

   2. Pinching by resin without the involvement of fibers is allowed. Excess resin may be removed to determine if fabric is part of pinch. If the pinch is removed and no fibers are exposed, then this is not considered a defect. See Figure 51 - 2.

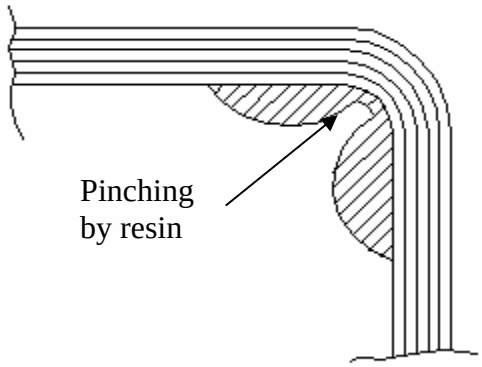


Figure 51 - 2 Resin Pinching

f. **Pits and Pinholes –** Pits are defined as small regular or irregular craters in the surface of a laminate, usually of a width the same order of magnitude as its depth. Pinholes are defined as small cavities that penetrate the surface of a cured part. Pits and pinholes in the resin that do not penetrate the reinforcement fabric shall be acceptable. If there is penetration of the reinforcement fabric, the defect is considered a fracture.

g. **Surface Resin Crazing –** Cracks in the resin that do not penetrate the reinforcement fiber are acceptable.

h. **Resin-Starved Areas –** Any condition that results in unencapsulation of reinforcing fibers shall be acceptable within the size as defined under "Delamination" in Figure 51 - 3 and Figure 51 - 6. This is caused by an inadequate amount of resin being applied during repair.

i. **Resin-Rich Areas –** Resin richness shall be acceptable provided no loose particles or jagged edges exist.

j. **Internal Defects –** Delaminations, air which is trapped between plies, interlaminar voids, bridging conditions, and foreign material within the laminate are acceptable provided that defective area is within the limit size as defined under "Delamination" in Figure 51 - 3 and Figure 51 - 6. Delaminations that result in a raised blister on the part surface are not acceptable. These internal defects typically occur due to poor compaction during lay-up.

k. **Upholstery Damage** – Damage to an upholstered surface which has cut or torn the upholstery but not damaged the composite part beneath it.

l. **Transparency Damage –** Damage to a transparent surface which causes a scratch, a pit or any other visible deformation. Transparent surfaces include the windshield, windows and the landing/taxi light cover.

m. **Bond Failure –** Damage which exists when two components begin to separate at the bonded surface.

51-5. FIELD REPAIRABLE DAMAGE

a. **Paint Damage:** Any damage to a painted surface which has not damaged the composite part beneath it shall be acceptable. The only considerations for repair will be for cosmetic purposes.

b. **Laminate Damage and Repair Defects**

1. **Exterior Parts –** Parts that are directly exposed to airflow or experience air loads. They are wheel pants, wing fillet, wing and vertical stabilizer access panels, flight

control surface access panels, and vortices generators. Allowable defects for these parts are given in Figure 51 - 3.

Defect	Allowables
Fracture	Fractures are not allowed.
Pinching	Pinching shall be acceptable provided the total length of the pinch does not exceed 5% of the length of the part.
Wrinkle	1. Protruding and intruding wrinkling on the uncontrolled surface is acceptable provided the total defect areas do not exceed 5% of the total area of the part. 2. Protruding and intruding wrinkle on the controlled surfaces are not acceptable.
Resin-Starve Areas	Resin-starve areas shall be acceptable provided the defect areas do not exceed 4.0 in^2. Resin-Starve at areas shown in figure 3 and figure 4 are not allowed.
Delamination around Attachment Holes	Delamination around attachment holes shall be acceptable provided the diameter of the delamination does not exceed the diameter of the head on the attachment bolt/screw.
Delamination in all other locations	Delamination shall be acceptable provided the total defect areas do not exceed 4.0 in^2. Delamination at areas shown in figure 3 and figure 4 are not allowed.
Porosity and Bridging	Porosity (Champagne bubbles) is acceptable. Bridging is acceptable provided no voids between plies exist at the bridged area.
Interlaminar voids and foreign material	Voids and foreign material areas shall be acceptable provided the defect areas do not exceed 4.0 $in^{2.}$

Figure 51 - 3 Maximum Allowable Defects (Exterior Parts)

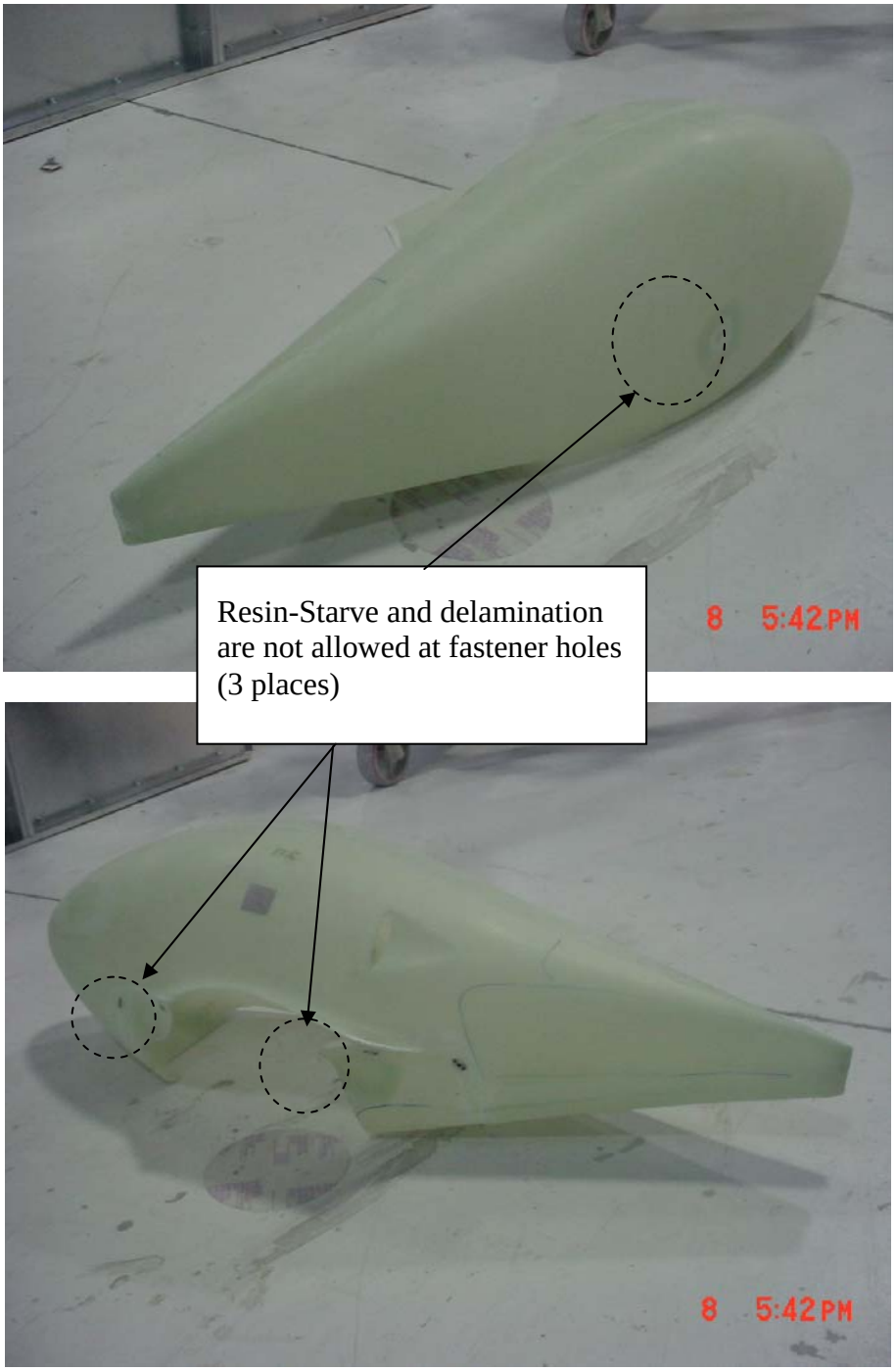


Figure 51 - 4 Main Gear Wheel Pant

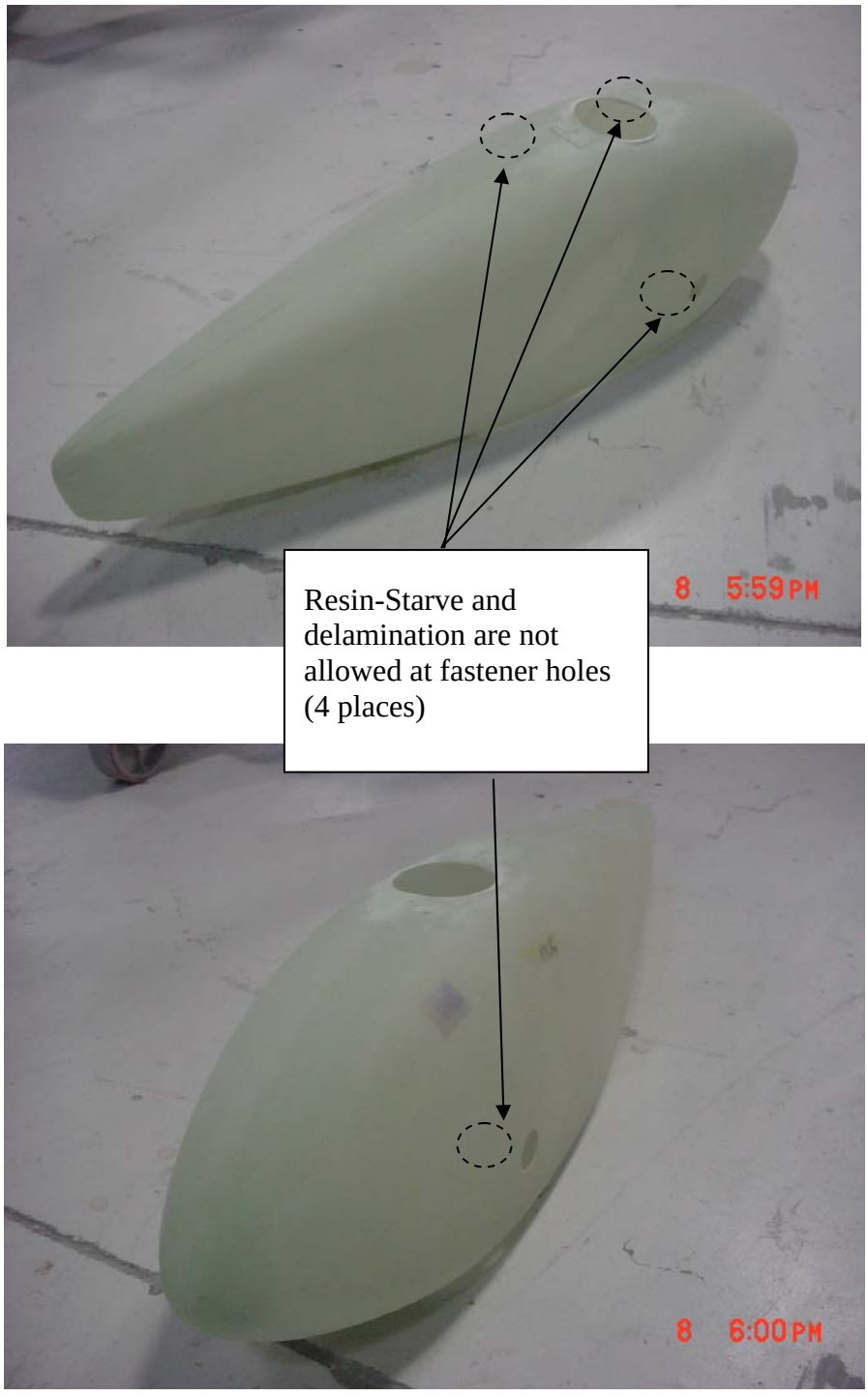


Figure 51 - 5 Nose Wheel Pant

2. **Interior Parts –** Parts that are not exposed to the airflow. They are interior paneling and their support brackets, cabin air vent ducts, access panels, and glare shield. Allowable defects for these parts are given in Figure 51 - 6.

Defect	Allowables
Fracture	Fractures are not allowed.
Pinching	Pinching is allowed except on interior paneling support brackets.
Wrinkle	1. Protruding and intruding wrinkling on the uncontrolled surface is acceptable. 2. Intruding wrinkle on the controlled surface shall be acceptable provided the total defect areas do not exceed 10% of the total surface area of the part. Protruding wrinkle on controlled surface is not allowed.
Resin-Starve Areas	Resin-starve areas shall be acceptable provided the defect areas do not exceed 5% of the total area of the parts
Delamination around Attachment Holes	Delamination around attachment holes shall be acceptable provided the diameter of the delamination does not exceed the diameter of the head on the attachment bolt/screw.
Delamination in all other locations	Delamination shall be acceptable provided the total defect areas do not exceed 5% of the total area of the parts
Porosity and Bridging	Porosity (champagne bubbles) and bridging are acceptable
Interlaminar voids and foreign material	Defect due voids or foreign material shall be acceptable provided the defect areas do not exceed 10% of the total area of the parts

Figure 51 - 6 Maximum Allowable Defects (Interior Parts)

a) **Upholstery Damage** – Damage to an upholstered surface which has cut or torn the upholstery but not damaged the composite part beneath it is allowable. The only considerations for repair will be for cosmetic purposes.

b) **Transparency Damage** – Damage to a transparent surface which causes a scratch, a pit or any other visible deformation is not allowed.

c) **Bond Failure** – Bond failure which meets the following criteria is acceptable:
   1) Exterior Parts – Acceptable provided the defect area does not exceed 4.0 in^2
   2) Interior Parts – Acceptable provided the defect area does not exceed 5% of the bonding surface.

51-6. Paint Repair and Painting

The paint system used to paint components of certified aircraft by Cessna is described in this section. All approved paints and step by step instructions for applying the paint are included. Specifically addressed are the primer filler used to fill-in void areas and make the surface smooth, primer sealer used to prevent the filler from penetrating through the paint, and the topcoats, both single-stage and two-stage. Safety and quality considerations are also addressed. Required paint thickness and solar absorptive levels for specific zones on the aircraft are included.

a. **Materials and Equipment –** PPG produces the standard paint products, and Fire Research Laboratories produces the fire resistant paint products. Recommended equipment and paint gun air pressures are listed for both types of paint. To achieve optimum results, mixing ratios can vary (up to 1% maximum) to suit equipment.

b. **Additional Information –** Should more data be required than presented in this document, such as equipment, health and safety issues, air pressures, tinting, touch-up procedures, humidity and altitude considerations, etc., refer to the technical manual for the paint being used and instructions on product containers.

WARNING

1) **Read labels on product containers for latest information on safety, emergency first aid, exhaust ventilation requirements, proper protective equipment or clothing, proper first aid equipment and supplies, and fire extinguisher requirements.**
2) **Avoid inhalation and direct skin contact of products.**
3) **Keep paint products away from flame and sparks.**
4) **Use the appropriate respirators when products are sprayed and when cleaners are applied.**
5) **Do not remove or cover labels on product containers.**
6) **Refer to Material Safety Data Sheet (MSDS) for safety, health hazards, recommended protection, emergency first aid, fire extinguisher, etc.**

c. **General Quality Procedures** – The following practices should be incorporated for the highest quality product.
   1. Keep the painted article clean.
   2. Keep the atmosphere free from bugs and dust.
   3. For anodized aluminum omit sealing operation of the anodizing process and avoid contamination by painting as soon as possible after drying the anodized article.
   4. Sand completely and evenly.
   5. Keep room temperature above 60° F.
   6. Measure all components of the mixture, do not estimate.
   7. Blend mixture thoroughly.
   8. Do not allow more than five minutes before force drying.

d. **Approved Body Filler –** Body filler is used to fill in and shape large areas such as bonded joints.
   1. Any polyester or epoxy-based body filler may be used as long as the total thickness requirements are not exceeded and the filler is non-conductive.
   2. **Procedure** – Manufacturer's instructions for the application of body filler must be followed.

e. **Approved Primer Surfacer –** The primer surfacer is used to fill in small divots, holes, and cracks for a smooth surface.
   1. **Approved Primer Surfacers –** PPG D822 Corrosion Resistant Primer
   2. **Procedure**
      a) Bubbles and voids must be opened up, sanded out, filled, and block sanded smooth to surrounding surface.
      b) Clean entire joint and bond area with soap and water if necessary.
      c) Orbital sand with 240 grit paper. Hand sand all hard to reach areas using 240 grit dry paper.
      d) Use a filtered air blow gun to remove sanding dust from corners and cracks.
      e) Clean with D837 Spirit Wipe.
      f) Mask for primer application.
      g) Mix 3 part D822 with 1 part D823 by volume. Pot life of mixture is 1- 1.5 hours at 68° F. It is acceptable to thin up to 1/2 volume with D870, D872, or D873 to improve flow and smoothness.
      h) Set spray gun to 1.4-1.6 mm, 3-4 bars (45-55 psi) spray pressure, and spray viscosity of 22 secs ZAHN 2 at 68° F (temperature may be changed based on the hardener and thinner being used).
      i) Apply mixture into fill areas up to 6 mils maximum dry coat thickness.
      j) Allow 1-2 hours to dry at 68° F or 20 to 30 minutes at 140° F. Use a maximum of three coats.
      k) Sand after primer has dried, preferably within 12 hours of drying. If sanding after primer has dried more than 12 hours, clean surfaces with D837 Spirit Wipe prior to sanding. Orbital sand major areas with 320 grit or finer sandpaper and hand sand other areas with 400 grit sandpaper. The smoother the sanded finish, the smoother the top coat application.

f. **Approved Primer Sealer –** The primer sealer is used as a wet-on-wet sealer to prevent fillers from penetrating through the paint.
   1. **Approved Primer Sealers**
      a) PPG D825 2K Tintable Surfacer Base
      b) PPG D822 Corrosion Resistant Primer
   2. **Procedure**
      a) Wash surface with soap and water if necessary.
      b) Wipe down with D837 Spirit Wipe.
      c) Mix per manufacturer's instructions. The following hardener and thinner combinations should be used for the corresponding temperatures:

Temperature	Hardener	Thinner
Up to 65° F	D803 Fast	D870 Fast Thinner
65 ° F - 77° F	D841 Medium	D871 Medium Thinner
77° F - 95° F	D861 Slow	D872 Slow Thinner
Over 95° F	D861 Slow	D873 Very Slow Thinner

d) Set spray gun to 1.4-1.6 mm, 4 bars (55 psi) spray pressure, and spray viscosity of 19-22 secs ZAHN 2 at 68° F.
e) Apply mixture up to 2.4 mils maximum dry coat thickness.
f) Allow 5-10 minutes to dry at 68° F between coats. Use a maximum of 2 coats.
g) If rework is necessary dry sand with 360 grit paper or wet sand with 400 grit paper.

NOTE

Pot life for the mixture is 1.5 hours at 68° F.

D825 may be tinted with DG colors if desired.

Coverage is approximately 396 ft^2 per gallon at 1 mil thickness.

g. **Topcoats –** The topcoat is the last layer in the painting process that contains the color and protects the part. There are three different types of topcoats: single-stage, two-stage, and three-stage. The advantage of using a single-stage is that it is much faster than the two-stage. The two stage, however, is easier to blend and more durable. The three-stage is used for special pearl coats. The needs of the application should be considered when choosing which method to use.

1. **Approved Topcoats**
   a) PPG DG single-Stage system
   b) PPG BC Two-Stage System
   c) PPG BC Three-Stage System
2. **Single-Stage System Procedure**
   a) Sand after primer has dried, preferably within 12 hours of drying. If sanding after primer has dried more than 12 hours, clean surfaces with D837 Spirit Wipe prior to sanding. Wet-sand with 400–500 grit or finer sandpaper or dry-sand with 360-400 grit or finer sandpaper. The smoother the sanded finish, the smoother the top coat application.
   b) Use a filtered air blow gun to remove sanding dust from corners and cracks. Wipe with D837 Spirit Wipe and tack cloth.
   c) For two-tone paint application, allow 30 minutes at 68° F before applying masking tape.
   d) For the first coat, mix per manufacturer's instructions. The following hardener and thinner combinations should be used for the corresponding temperatures:

Temperature	Hardener	Thinner
Up to 65° F	D841 Medium	D870 Fast Thinner
65 ° F - 77° F	D841 Medium	D871 Medium Thinner
77° F - 95° F	D861 Slow	D872 Slow Thinner
Over 95° F	D861 Slow	D873 Very Slow Thinner

NOTE

Pot life for the mixture is 1-3 hours at 68° F.

e) Set spray gun to 1.4-1.6 mm and 3-4 bars (45-55 psi) spray pressure and spray viscosity of 17-20 secs ZAHN 2 at 68° F (temperature may be changed based on the hardener and thinner being used).
f) Apply mixture up to 2.5 mils maximum dry coat thickness. The first coat should be medium wet.
g) Allow 10 minutes to dry at 68° F between coats.
h) For the second coat, mix the same as the first coat. The second coat should be full wet.
i) If DG color needs to be flatted, wet sand with 1200 grit paper or finer.

NOTE

For maximum total paint thickness (including primer filler, primer sealer, and topcoats) see section 51-6.j. Colors must be chosen to meet the solar absorptive levels of section 51-6.k.

Coverage is approximately 647 ft^2 per gallon at 1 mil thickness.

3. **Two-Stage System Procedure**
   a) Sand after primer has dried, preferably within 12 hours of drying. If sanding after primer has dried more than 12 hours, clean surfaces with D837 Spirit Wipe prior to sanding. Wet-sand with 500–600 grit or finer sandpaper or dry-sand with 400-500 grit or finer sandpaper. The smoother the sanded finish, the smoother the top coat application.
   b) Use a filtered air blow gun to remove sanding dust from corners and cracks. Wipe with D837 Spirit Wipe and tack cloth.
   c) For the base coat (first stage), mix per manufacturer's instructions. The following thinner should be used for the corresponding temperatures:

Temperature	Thinner
Up to 65° F	D870 Fast Thinner
65 ° F - 77° F	D871 Medium Thinner
Over 77° F	D872 Slow Thinner

NOTE

There is no pot life for the mixture. Stir well before using.

d) Set spray gun to 1.3-1.5 mm and 3-4 bars (45-55 psi) spray pressure and spray viscosity of 14-20 secs ZAHN 2 at 68° F
e) Apply mixture up to 1.0 mils maximum dry coat thickness
f) Allow 5 to 10 minutes to dry at 68° F between coats (not to exceed 24 hours). Number of coats is 2 medium wet. Allow 30 minutes at 68° F before applying masking tape

NOTE

Coverage is approximately 243 ft^2 per gallon at 1 mil thickness.

g) For the clear coat (second stage), mix per manufacturer's instructions. The following hardener and thinner combinations should be used for the corresponding temperatures:

Temperature	Hardener	Thinner
Up to 65° F	D803 Fast	D870 Fast Thinner
65 ° F - 77° F	D841 Medium	D871 Medium Thinner
Over 77° F	D861 Slow	D872 Slow Thinner

NOTE

The pot life for the mixture is 6 hours at 68° F.

h) Set spray gun to 1.5-1.7 mm and 4 bars (55 psi) spray pressure and spray viscosity of 16-17 secs ZAHN 2 at 68° F

i) Apply mixture up to 1.0 mils maximum dry coat thickness. Allow 10-15 minutes to dry at 68° F between coats. Number of coats is 1 light and 1 full wet.

NOTE

Coverage is approximately 321 ft^2 per gallon at 2 mil thickness

For maximum total paint thickness (including primer filler, primer sealer, and topcoats) see section 51-6.j. Colors must be chosen to meet the solar absorptive levels of section 51-6.k.

h. **Fire Resistant Paint –** Ocean Fire Retardants #477 (2 part) Epoxy is used for fire protection of aluminum and composites. When #477 Epoxy is exposed to flame, it expands up to form a very dense, insulating, spongy foam crust, which retards heat penetration to the under surface. The following items are applicable to fire resistant paints.

1. Application over epoxy primer is preferred but not required for best adhesion. Prime per Section 51-6.e.
2. Ensure surface is dry and free of oil or grease.
3. Mix 8 parts #477A with 1 part #477B by volume. Allow 45 to 60 minutes to activate 1 gallon and 30 to 40 minutes to activate 5 gallons at room temperature (70° F).
4. Reduce mixture after activation with Xylene or Solvent Naphtha at a ratio of 1/2 pint per gallon.
5. Application method can be by brush, roll, or spray. Airless spray application is the application of choice. Airless spray equipment is preferred with recommended pressure of 2650 psi with a 0.015 to 0.017 in. tip size. Clean equipment with Xylene immediately after application.
6. With air-less spray, avoid wet film build-ups greater than 10 mils per pass. Rolling produces a pattern that should be brushed out.
7. Dry film recommended build up is 10 to 12 mils and typically requires two coats minimum, which is a coverage rate of 165 ft2 per gallon. Wet film build up is 23 to 27 mils total. Shelf life is 6 months. Store in a well ventilated area at 40° to 120°F. Do not store or use in areas with open source of ignition. Flash point is 105°F. VOC maximum is 1.5 lbs. per gallon.

8. Cure times at 70° F are, dry to recoat – 60 minutes, dry to touch – 4 hours, dry to handle – 24 hours, through cure – 7 days, and maximum cure and hardness takes 2 to 3 weeks. Higher temperatures accelerate speed of cure. Clean up with Xylene.

i. **Paints for Special Components**

1. **Windows and Windshields –** The window and windshield sashes may be painted black for aesthetic purposes. The painted area will be along the perimeter and will cover 1/2 in. ± 1/8 in. from the exterior paint. The paint on the exterior of the window must cover the black paint so that no more than 1/4 in. extends past the exterior paint when looking from the outside through the window. The paint and paint procedures used must be in accordance with section 51-6.g. A specific paint thickness is not required.
2. **Landing and Taxi Light Cover –** The landing and taxi light cover must be painted black per Figure 51 - 7 (hatched areas indicate paint). The paint and paint procedures used must be in accordance with section 51-6.g. of this specification. A specific paint thickness is not required.

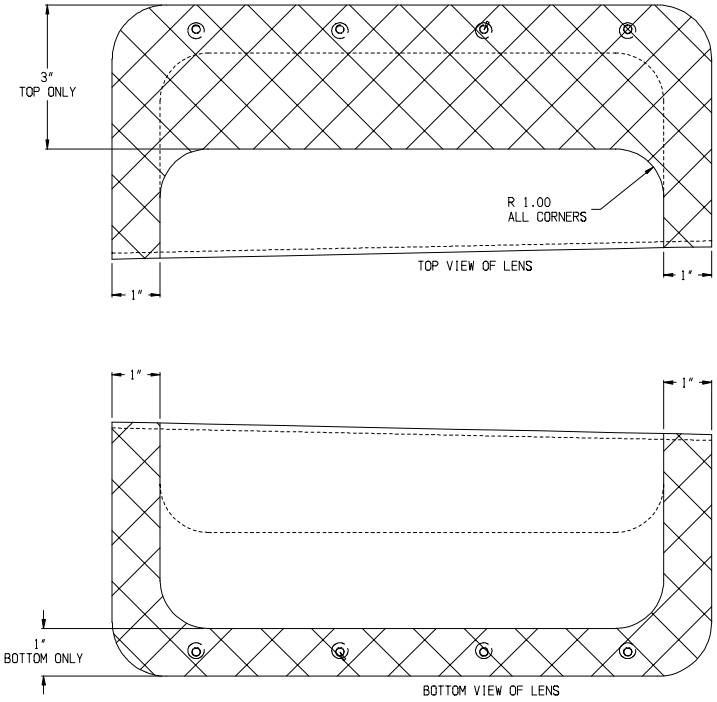


Figure 51 - 7 Landing and Taxi Light Paint Areas

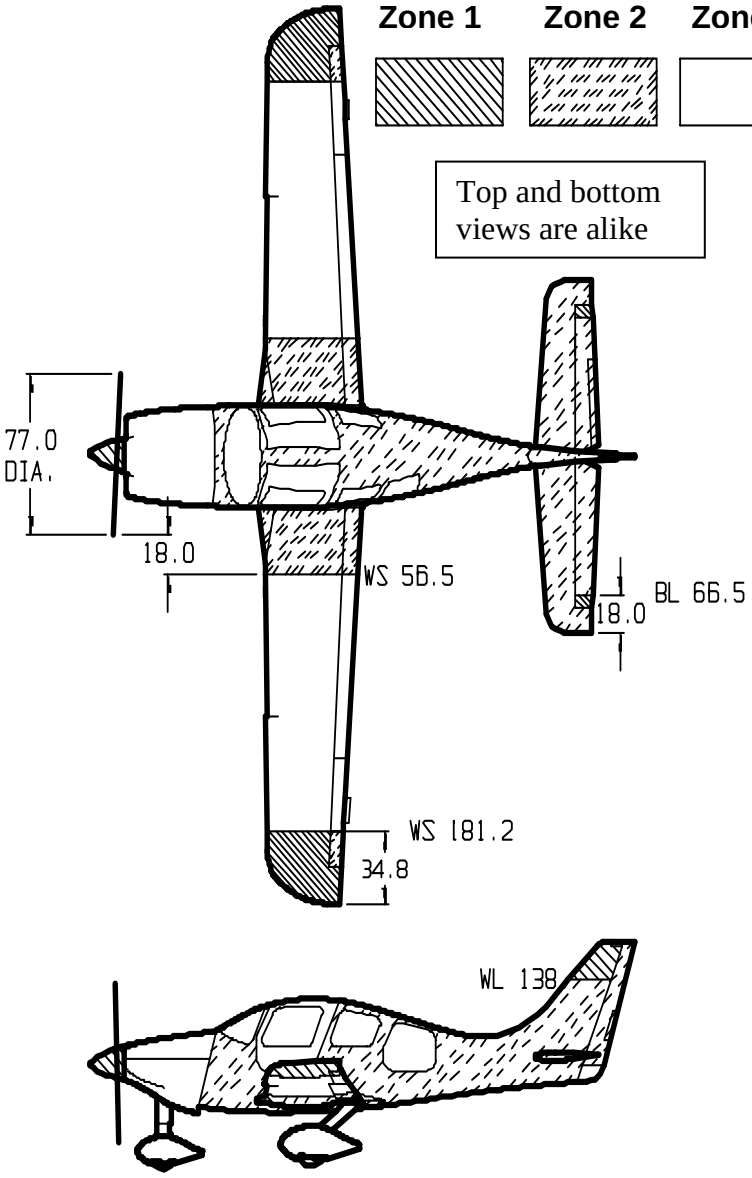
3. **Interior Panels –** All interior panels that are required to be painted must be painted in accordance with this section.
   a) **Approved Paints**
      1) VQT DuPont Vinyl Color VQT Custom Match Tan
      2) PPG Exterior White
      3) PPG DG9300 Black
      4) PPG Custom Paint Platinum Silver (#771 Med. Alum: 355.6, #772 Coarse Alum: 492.6, #743 Yellow: 494.9, #753 White: 496.1, and #759 Flattener: 503.2.

b) **DuPont Paint Procedure**

1) Bubbles and voids must be opened up, sanded out, filled, and block sanded smooth to surrounding surface.
2) Clean entire joint and bond area with soap and water if necessary.
3) Orbital sand with 240 grit paper. Hand sand all hard to reach areas using 240 grit dry paper.
4) Use a filtered air blow gun to remove sanding dust from corners and cracks.
5) Clean with D837 Spirit Wipe.
6) Mask for primer application.
7) Shake paint well before using.
8) Set spray gun as follows: for a traditional gun, 25-40 psi spray pressure, for a HVLP gun, 6-8 psi spray pressure. Set spray viscosity of 18-22 secs ZAHN 2.
9) Spray until covered evenly.
10) Allow 5 minutes to dry between coats. Apply 2-3 coats. Allow 30 minutes to dry.
11) A specific paint thickness is not required.

c) **PPG Paint Procedure –** Follow the procedures in section 51-6.g. for PPG paint. A specific paint thickness is not required.

j. **Paint Thickness Zones –** Paint thickness must be controlled so the aircraft has proper conductivity to prevent damage from lightning strikes. Figure 51 - 8 shows the maximum paint thickness for different zones on the aircraft.



Paint Zone 1: Thickness not to exceed 10 mils (0.010 inch) except for repair or seams for joined parts requiring body filler for a smooth surface. This area cannot exceed 4” in the minor direction – major direction unlimited. Discontinuities in thickness are allowed up to 4” in the minor direction with 4” minimum between discontinuities. Combined discontinuities not to exceed 20% of the zone area.

Paint Zone 2: Thickness not to exceed 15 mils (0.015 inch) except for repair or seams for joined parts requiring body filler for a smooth surface. This area cannot exceed 5” in the minor direction – major direction unlimited. Discontinuities in thickness are allowed up to 5” in the minor direction with 5” minimum between discontinuities. Combined discontinuities not to exceed 20% of the zone area.

Paint Zone 3: Thickness restrictions do not apply.

Figure 51 - 8 Paint Thickness Zones

k. **Paint Color Zones –** Paint color zones control exterior paint colors based on solar absorptive levels. Solar absorptive values indicated are for the zones shown in Figure 51 - 9 and Figure 51 - 10. If colors different than originally on the aircraft and not listed in Figure 51 - 11, are to be used in Zone A or B, the absorbtivity must be determined using ASTM E903; only paints that meet the absorbtivity levels may be used. Any paint color lighter than black, using an appropriate paint system specified in this chapter, may be used in Zone C without absorbtivity testing. Approved colors and associated zones are listed in Figure 51 - 11.

- Zone A – $\alpha s < 0.29$ White to Zinc Oxide
- Zone B – $\alpha s < 0.65$ Gray to Medium Green, Blue, or Tan
- Zone C – $\alpha s < 0.95$ Black

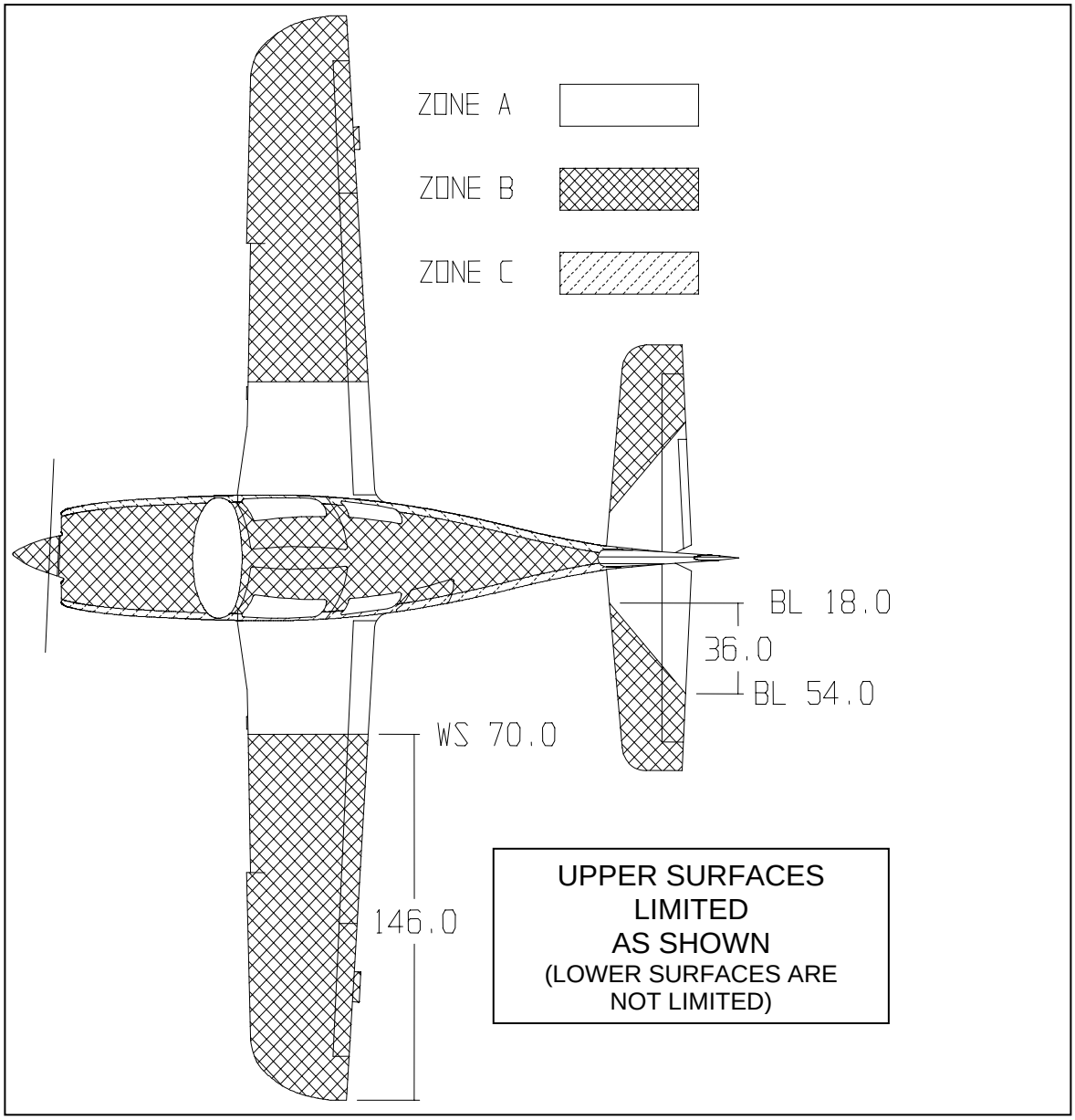
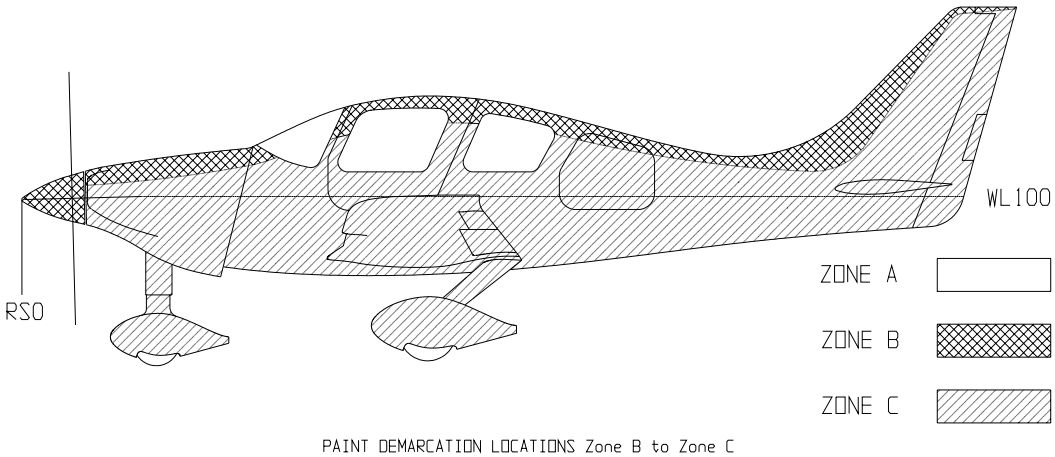


Figure 51 - 9 Paint Color Zones for UV and Temperature



PAINT DEMARCATION LOCATIONS Zone B to Zone C

WATERLINE STATION	WL103.5	WL104.2	WL105.2	WL106.4	WL107.9	WL109.6	WL117.7	WL119.0	WL122.4	WL122.1	WL119.4	WL117.1
REFERENCE STATION	RS20	RS30	RS40	RS50	RS60	RS70	RS95	RS100	RS130	RS140	RS160	RS170

WATERLINE STATION	WL114.9	WL113.0	WL111.4	WL110.1	WL109.0	WL108.2	WL107.7	WL110.0	WL120.6	WL135.2	WL149.7
REFERENCE STATION	RS180	RS190	RS200	RS210	RS220	RS230	RS240	RS250	RS260	RS270	RS280

Figure 51 - 10 Paint Color Zones for UV and Temperature (Side View)

Generic Color	Color Number	Color	Zone
Whites	BC91046A / BC5135	White Diamond Pearl coat over Cessna 300 Standard White (3-stage)	Zones A, B, C
	BC5134A/BC5135A	Bright White Diamond (3-stage)	Zones A, B, C
	BC91046A	Standard Cessna White	Zones A, B, C
	BC 3563	Pearle White (Non Metallic)	Zones A, B, C
Yellows	BC84903B/BC84904A	Giallo Orion (3-stage)	Zones B & C
	BC4030	Viper Yellow	Zones B & C
Greens	BC5492	Electric Green	Zone C
	MBC46979/BC5135	Get Green Pearl	Zone C
	BC45817B	Hunter Green or PACUSA Green	Zone C
	5534	2001 Honda Clover Green Pearl	Zone C
	402587	2001 Mercedes-Benz Prismatingrun Met	Zone C
	195027	2001 Volvo Sydney Bay Blue Mica	Zone C
	47395	2001 Subaru Spruce Pearl	Zone C
	402374	2001 Volkswagen Green Collection Pearl	Zone C
	5409	2001 General Motors Forest Green Met	Zone C
	904571		Zone C
	6213		Zone C
	B648695	Racing Green	Zone C
Blues	BC4372	Solar Blue Met	Zone C only
	BC5588	Island Blue Met	Zone C only
	BC18514***A	Light Poly Teal	Zones B & C
	BC5222C	Bright Purple	Zone C only
	BC5223C	Dark Mulberry	Zone C only
	BC4784	Glacier Blue or PACUSA Blue	Zone C only
	190397	2001 Mercedes-Benz Quartz Blue Met	Zone C only
	193775		Zone C only
	BC19598	Navy	Zone C only

Generic Color	Color Number	Color	Zone
Reds/Oranges	BC74712	Lancair callout: Mojave Red or PACUSA Red	Zone C only
	BC74492C	Lancair callout: Pontiac Red	Zone C only
	BC61930C	Prowler Orange	Zone C only
	BC5359	Cranberry Pearl	Zone C only
	902122		Zone C only
	BC61930	Flame Orange	Zone C only
	BC75250	Ferrari Red	Zone C only
	5222		Zone C only
Golds	BC25622A	Pure Gold Radiance	Zones B & C
	BC4138	PACUSA Gold	Zones B & C
	5584	Gold (brown)	Zones B & C
	5383	Gold	Zones B & C
	BC 4138	Royal Gold	Zones B & C
Silvers	BC4010A	Silver Poly	Zones B & C
	BC35482	Moon Stone	Zones B & C
	BC33896	Aluminum	Zones B & C
	BC4010	Starlight Silver (metallic)	Zone A, B and C
	Custom color*	Cessna Starlight Silver (nonmetallic)	Zone A, B and C
	Custom color**	Cessna Platinum Silver (metallic)	Zone A, B and C
Blacks/Dark Greys	5559	River Rock Green Mica	Zone C only
	BC9300	Black	Zone C only
	BC5292	Dark Slate Pearl coat (2-stage)	Zone C only
	BC17038	Black	Zone C only
	905384		Zone C only
	BC36447	Viper Steel Gray	Zone C only
Others	BC35915	Shale	Zones B & C
	BC 26634	Cream Beige (Non Metallic)	Zone A, B and C

* 92.9% D753, 4.68% D740, 2.02% D778, .42% D779

** 15.2% D771, 21.0% D772, 21.1% D743, 21.2% D753, 21.5% D759

Figure 51 - 11 Approved Paint Colors

1. **Zone C Paint in a Zone B Area (Wing Tip and Vertical Tail) –** On the wing tip and vertical tail, Zone C paint may be used in a Zone B area provided the surface painted with Zone C paint does not exceed 2% of the overall Zone B area.
2. **Zone C Paint in a Zone B Area (Cowl) –** On the Zone B surface of the cowl, Zone C paint can be used to cover the overall area.

51-7. LAMINATE REPAIR

a. **Approved Materials –** All materials approved for use under this chapter are listed in Figure 51 - 12 through Figure 51 - 16 along with sources of supply. Approved interior laminates are listed in Figure 51 - 15.

Material	Product Designation	Cure Cycle	Storage Requirements	Manufacturer
Polyester	Dion FR 7704-00	1. Mix with 1.2 to 2.5% catalyst by weight. 2. Room temperature cure until laminate is firm.	Store in a closed container at or below 75°F and away from heat sources and sunlight. Open shelf life is 3 months. Closed shelf life is 12 months from date of manufacture.	Reichhold
Epoxy	MGS 285	1. Mix laminating resin L285 with either 285, 286 or 287 Hardener at a ratio of 100:39 parts by weight, ±2. 2. Precure until the part is hard to the touch. The part shall remain undisturbed until the precure is complete. 3. Postcure adhesive per table A.2.	Store in closed containers between 70° and 85°F. Unopened shelf life is 24 months from the date of manufacture. Open shelf life is 6 months.	MGS
Epoxy	Jeffco 1307-LV Resin with 3102 Hardener	1. Mix ratio 100:22 by weight (4:1 by volume) 2. Use epoxy within 11-15 minute gel time 3. Allow to set at room temperature (77°F) for a minimum of 3 hours.[1] 4. Post cure adhesive per table A.2.	Store in a closed container at or above 50°F. Unopened shelf life is 2 years from the date of manufacture. Open shelf life is 1 year.	Jeffco

Figure 51 - 12 Approved Resin Materials

[1] Laminate may be further worked with after this point, but repair work is not finished until Step 4 is complete.

Adhesive	Cure Temperature (°F)	Cure Time (Hours)
MGS 285	77	66
	87	46
	97	34
	107	24
	117	18
	127	14
	137	10
	147	8.0
	157	6.0
	167	5.0
	177	4.0
	187	3.0
	197	2.0
	207	1.75
	217	1.5
	227	1.25
	237	1
	247	.75
	257	.60
	267	.50
	277	.40
	287	.30
	297	.25
Jeffco	77	168
	87	76
	97	40
	107	24
	117	16
	127	11
	137	8.0
	147	6.1
	157	4.8
	167	4.0
	177	3.3
	187	2.8
	197	2.4
	207	2.1
	217	1.9

Figure 51 - 13 Approved Adhesive Cure Cycles

Material	Product Designation	Finish	Storage Requirements	Manufacturer
Glass Fabric	7725	Compatible with resin system being used – polyester or epoxy[2]	Store away from direct sunlight. No shelf life.	Hexcel
Glass Fabric	7500		Store away from direct sunlight. No shelf life.	BGF
Glass Fabric	7781		Store away from direct sunlight. No shelf life.	BGF
Carbon Fabric	3k Plain Weave		Store away from direct sunlight. No shelf life.	BGF

Figure 51 - 14 Approved Cloth Materials

Laminate	Fire Resistance Level
A minimum of four plies of 7725 glass cloth at any orientation with Dion FR7704-00 resin with MEKP catalyst, per this specification, painted with approved white PPG exterior paint.	Self-Extinguishing, 60 Second Vertical Test
A minimum of six plies of 7725 glass cloth at any orientation with Dion FR7704-00 resin with MEKP catalyst, per this specification, painted with D822 Primer Undercoat, D823 Catalyst, Primer Under Coat, and VQT DuPont Vinyl Color VQT Custom Match Tan paints from SX512701.	Self-Extinguishing, 60 Second Vertical Test
A minimum of 20 plies of 7781 glass cloth at any orientation with Dion FR7704-00 resin with MEKP catalyst, per this specification, painted with D822 Primer Undercoat, D823 Catalyst, Primer Under Coat, and VQT DuPont Vinyl Color VQT Custom Match Tan paints from SX512701.	Self-Extinguishing, 60 Second Vertical Test
A minimum weight of 25.9 oz/yd^2 of any glass cloth from this specification at any orientation with Dion FR7704-00 resin with MEKP catalyst per this specification, painted with white PPG exterior paint or D822 Primer Undercoat, D823 Catalyst, Primer Under Coat, and VQT DuPont Vinyl Color VQT Custom Match Tan paint from SX512701.	Flame Resistant

Figure 51 - 15 Approved Interior Laminates

[2] Volan finish was found to be non-compatible with MGS 418. Silane finish is acceptable because of its compatibility with polyester and epoxy resins.

Purpose	Material
Solvent Wash	MEK Acetone Isopropyl Alcohol **NOTE:** Reclaimed solvents are not acceptable.
Solvent Wipe	Lint and solvent free cheese cloth or equivalent cloth that is free of oil and soap
Abrasion of Composite Surface	Scotch-Brite™ pad 80 or 120 grit sand paper or equivalent abrasive

Figure 51 - 16 Approved Surface Preparation Materials

Defects		
Defect Type being Repaired	Number of Repair Plies	Repair Area
Fracture Delamination	Same as the number of plies affected during the removal of the defect. (See step 1 below)	1 – 2 Repair Plies: Defect and area surrounding by 0.5” 3 Repair Plies and more: Defect and area surrounding by 1.0”
Pinching Wrinkle Porosity and Bridging Interlaminar voids and foreign material	2 Plies	Defect and area surrounding by 0.5”
Resin Starved Area	1 Ply	

Figure 51 - 17 Repair Guidelines

b. **Laminate Repair Method**

1. Using sand paper or by bead blasting remove the entire defect area. If the damage area being repaired is a fracture through all plies it is permissible to leave enough of the damaged area on the part to provide an adequate contour for repair.
2. If defect type being repaired is a fracture or delamination it is necessary to record the number of plies which are removed by methods described in the following steps:
   a) Visually it can be determined by looking for the number of ‘rings’ which occur between the area where the defect was removed and surrounding areas. This ‘ring’ will appear as a translucent line where no orientations are visible. By lightly whetting the surface with alcohol and allowing it to dry, the rings can be seen by watching how the alcohol dries on one ply at a time.
   b) It can also be done by comparing the depth of the area where the defect was removed with surrounding areas. Ply thickness on average is 0.010”.
   c) By using a combination of these two methods the number of plies removed can be determined. If in doubt it is advised to be conservative with your determination.

3. Using 80 grit or finer sand paper (with an sander or sanding block) remove all paint and primer which surrounds the defect area by the value determined in table 9 under 'Repair Area'. Sand in a circular motion to prevent any damage to the underlying laminate.

WARNING

It is not permissible to use any chemical paint strippers as this may damage the composite part.

4. In defect area and that surrounding as defined in table 9 under 'Repair Area' perform the following steps which constitute surface preparation:
   a) Wash area with approved solvent and wiping cloth.
   b) Abrade surface of area with approved abrasive.
   c) Remove abrading dust using a vacuum.
   d) Wipe surface of area with an approved wiping cloth (do not reuse any wiping cloths) to remove residual dust.
   e) Wash area with a wiping cloth (do not reuse any wiping cloths) saturated with an approved solvent. Thoroughly wet the area with approved solvent.
   f) Immediately wipe area with wiping cloth (do not reuse any wiping cloths) using a single stroke motion while rolling the wipe away from the surface. After each wipe turn the drying cloth to a clean surface.

NOTE

Steps 4e & 4f must be performed within 2 hours of the application of the repair plies to the damaged area (done at Step 8 below). If more than 2 hours elapse between steps 4e/4f and step 8, steps 3e and 3f must be performed again.

5. Select an approved resin system and mix per directions in Figure 51 - 12. Follow all safety guidelines as provided by the supplier or noted in an MSDS sheet.
6. Select an approved fabric and cut a number of plies corresponding to column 'Number of Repair Plies' from Figure 51 - 17. Only use glass fabric when the substrate is glass and only use carbon fabric when the substrate is carbon. The size of the cloth must be adequate to cover the discrepant area in addition to the area defined by column 'Repair Area' from Figure 51 - 17.
7. Saturate the cloth with mixed resin. To prevent delamination and internal defects it is recommended that every ply of cloth is saturated with resin separately and then stacked on to any other plies that will be used in the repair. As plies are stacked they are to be compacted individually to remove any trapped air. This can be done with multiple tools, including a gloved hand, a squeegee or a tongue depressor.
8. Apply stack of plies to surface prepped area. Compact the repair plies against the original plies. Using the same method in step 7 for compaction, smooth out these areas to remove any trapped air. Add additional resin to the repair as necessary to prevent any dryness.
9. Cure the repair per column 'Cure Cycle' of Figure 51 - 12.
10. Restore primer and paint as necessary using methods outlined in those areas.

51-8. UPHOLSTERY REPAIR

a. **Approved Materials**

Purpose	Material	Supplier
Filler	LFP Compound P01LFP	Flexproof
Primer	Leather Primer **P04GRA08**	Flexproof
Texture Mats Compound	R Compound **T04RCM**	Flexproof
Texture Mats Catalyst	R Catalyst **T04CT03**	Flexproof
Degreaser	Acryliclean DX-330	PPG
Leather	Leather - **LL-3821**	Douglas Interiors
Ultra Leather	Buff Colored Ultra Leather – 3609	Scalise Marine
Ultra Leather	Papyrus Colored Ultra Leather - 1256	Scalise Marine

Figure 51 - 18 Approved Leather Repair Materials

b. **Grain Transfer Mat Creation**

1. Pour the Texture Mat Compound on a clean surface until the puddle is about 2-1/2 inches in diameter.
2. Pour a few drops of an approved interior paint into the middle of the Compound.
3. Pour a few drops of the Texture Mat Catalyst into the paint in the middle of the Compound.
4. Mix Catalyst and Paint with the compound. Even distribution of the paint in the mix will signify an even equal distribution of the Catalyst and Compound.
5. Take a scrap piece of leather or ultra leather (dependent on what type of upholstery repair you will be performing) and clean with degreaser.
6. Using masking tape, tape an area off measuring 2.5”x4.0”.
7. Spread the mixture evenly onto your taped off area until it is between 1/32 and 1/16 of an inch thick.
8. Peel off the tape from the leather/ultra leather.
9. Let the compound sit for at least 24 hours prior to removal from the leather/ultra leather.

c. **Repair Method**

1. Clean damaged and surrounding area by 1.0” with degreaser.
2. Using a q-tip saturate the discrepancy and area 1/8” surrounding damage with primer for leather. Primer must be dry (use heat gun as necessary) prior to proceeding to next step.
3. Clean GTM with alcohol and allow to dry before proceeding.
4. Fill discrepant area with filler.
5. Heat GTM to 200 degrees with heat gun (GTM will begin to smoke when this temperature is reached). Apply GTM to discrepant area with pressure for 15 seconds. Gently peel away GTM from area of repair.
6. Using a chill bar press down on the area of repair to flatten and cool it.
7. Repeat steps 5 and 6 as necessary until area of repair is smooth to touch and you can see the outline of the grain in the repair area.

8. Apply primer with a q-tip to all areas where filler was applied (this area should have a duller color than the leather).
9. Paint over area where primer was applied. Apply one coat at a time, drying as necessary with heat gun. If paint gets too hot it will boil.
10. Clean GTM with alcohol and allow it to dry before proceeding.
11. Heat GTM to allow it to imprint and flatten to paint. Do not heat GTM as excessively as in step 5. Apply heat gun for no more than 15 seconds at a time. If too much heat is applied to GTM the paint will boil or adhere to the GTM.
12. Apply GTM to painted area with pressure from the Chill Bar.
13. If any holes which allow for attachment of the interior panels were repaired and the area of the hole has been compromised it is recommended a soldering iron/gun is used to restore the full area of the hole.
14. Using an approved interior paint, paint the repair to match the color of the surrounding area.

51-9. TRANSPARENCY REPAIR

a. **Approved Materials**
   1. Novus No. 1 Plastic Clean & Shine (For general cleaning)
   2. Novus No. 2 Fine Scratch Remover (For fine scratches)
   3. Novus No. 3 Heavy Scratch Remover (For heavy scratches)
   4. Summer Laboratories 210® Plastic Cleaner and Polish (For general cleaning)
   5. I00% cotton flannel polishing cloths
   6. 3MTM Adhesive remover Citrus Base 6040-6042 (for general cleaning)

b. **General Cleaning Procedures –** Never do anything that will scratch the surface of the acrylic plastic. The following points for cleaning and caring for the transparencies will ensure no further damage occurs to the transparency during repair.
   1. First, when cleaning any transparency, remove rings and watches as they can cause deep scratches. In this vein, long sleeve shirts must be turned up a few rolls to hide exposed buttons.
   2. If the transparency does not have built-up dirt and bugs on the surfaces, but rather has finger prints and/or dust, Novus No. 1 can be applied directly per product specific application instructions. Summer Laboratories 210® can also be used for this purpose per product specific application instructions.
   3. It is necessary to removing bugs and/or heavy accumulations of dirt, avoid touching the surface. If possible, remove most of the dirt by flushing the windows and windshield with water and a mild dish soap mixture. Allow the accumulation of dirt and/or bugs to soak for a few minutes. If rubbing is required, a bare hand is best. When all the debris on the surface of the window is loosened, apply a second water flush, dry with a 100% cotton cloth, and apply Novus No. 1 per product specific application instructions.
   4. The above steps and products may not remove all substances, since they are intended primarily for dirt and smudges. To remove difficult items such as tape residue, oil, and grease, the safest solvents are 100% mineral spirits or kerosene. Some alcohols are safe, such as isopropyl alcohol.
   5. The best polishing cloth is the softest cotton available. One hundred percent cotton flannel is ideal and available in yard goods stores. Never use any type of paper product or synthetic material. In particular, never use shop rags or shop towels. Be

sure the polishing cloth is clean and dry. Reserve polishing cloths should be stored in a plastic bag or airtight container to limit dirt accumulation. Do not reuse soiled polishing cloths until they are properly laundered.

6. Small scratches, the type that can be seen but cannot be felt with a fingernail, should be treated with Novus 2. First, clean the transparency per instructions in Step Nos. 2, 3 and 4 (if necessary), and then follow product specific application instructions for using Novus No. 2.
7. If after using Novus No. 2, scratches are still visible, apply Novus No. 3 per product specific application instructions.
8. Scratches of greater magnitude (those that can be felt with a fingernail) require the use of high abrasives and removal of some of the window's surface around the greatest depth of the scratch. This procedure requires considerable expertise and the scratch may not be correctable. Repair of transparencies with a scratch depth of approximately .005 of an inch or greater (the depth at which you will be able to feel it with your finger nail) must be referred to your Cessna Aircraft Service Station.

CAUTION

Do not use anything containing ammonia, aromatic solvents like methyl ethyl ketone, acetone, lacquer thinner, paint stripper, gasoline, benzene, alcohol, anti-ice fluid, hydraulic fluid, fire extinguisher solutions, or window cleaner on the acrylic transparency surfaces. The use of these substances may cause the surface to craze.

c. **Specific Application Instructions**

1. **Novus No. 1**
   a) If the surface is extremely dirty, apply Novus No. 1 liberally and wipe using long, sweeping strokes. Do not use pressure at this time or large dirt particles may scratch the plastic. If there are heavy accumulations of bugs and built-up dirt, follow Step No. 3 of the General Cleaning Procedures.
   b) Shake well before using.
   c) Re-apply Novus No. 1. Polish using short, circular strokes with a clean portion of the cloth.
   d) When surface is thoroughly clean and uniformly covered, buff to a slippery glaze with a clean portion of the cloth. Surfaces buffed to a high glaze are more resistant to dust and future scratching.
2. **Novus No. 2**
   a) Observe condition of the surface to be treated. The deeper the scratches, the greater the pressure required to remove them. If the surface is dirty, clean with Novus No. 1 before applying Novus No. 2.
   b) Apply Novus No. 2 liberally. Using a clean, soft cloth, polish with a firm back-and-forth motion at right angles to the scratches. Keep the cloth saturated with polish at all times.
   c) Shake well before using.
   d) When the worst scratches have been polished out, reapply Novus No. 2 uniformly in a circular motion to the entire surface using short, circular strokes and light pressure. Allow to dry to a light haze.

e) Using a clean portion of the cloth, buff the surface to a slippery glaze using firm, short strokes. This procedure is imperative in achieving the best results.
f) For heavy scratches, multiple applications of Novus No. 2 or application of Novus No. 3 may be required. Follow with Novus No. 1 for best results.

3. **Novus No. 3**
   a) Remove surface dirt and dust with Novus No. 1.
   b) Shake well before using.
   c) Apply Novus No. 3 liberally. Using a clean, soft cloth, polish with a firm back-and-forth motion at right angles to the scratches. Keep the cloth saturated with polish at all times.
   d) Continue polishing until only fine scratches remain. Using cloth, remove all remaining polish, this is very important!
   e) Using a clean cloth, apply Novus No. 2 in a circular motion uniformly to entire surface. Polish using light pressure until Novus No. 2 dries to a light haze.
   f) Using a clean portion of the cloth, buff the surface clean. Follow with Novus No. 1 for best results.
4. **Summer Laboratories 210 (Plastic Cleaner and Polish)**
   a) This cleaner is dispensed from an aerosol can and ideal for light and quick cleaning jobs.
   b) Shake the can well before using.
   c) Spray a light coat of 210® Cleaner/Polish across the surface of the transparency.
   d) With a soft clean cloth, wipe and buff the surface until it is dry.
5. **3MTM Adhesive remover Citrus Base 6040-6042**
   a) This cleaner is dispensed from an aerosol can and ideal for light and quick cleaning jobs.
   b) Thoroughly coat adhesive residue / paint overspray.
   c) Wait at least 5 minutes for product completely penetrate and loosen adhesive or paint.
   d) Use a soft clean cloth to clean adhesive residue or paint.
   e) Repeat process if necessary.

51-10. BOND REPAIR

a. **Approved Materials –** All materials approved for use under this chapter are listed in Figure 51 - 19.

Material	Product Designation
Flock	525 Cotton Flock
Resin System	Hysol EA 9321 (Primer Part A and B)
Resin System	Loctite Product 330 with Activator 7387
Resin System	Jeffco 1307-LV Epoxy Resin with 3102 Hardener
Resin System	Plastic Welder
Resin System	Plastic Welder 2
Micro	K15 Glass Bubbles
Filler Mix	525 Cotton Flock with K15 Glass Bubbles and M5 Cab-o-Sil at a ratio of 1.0 Flock : 1.0 Micro : 0.22 Cab-o-Sil by VOLUME (e.g. 9 fl oz : 9 fl oz : 2 fl oz)

Figure 51 - 19 Approved Materials for Bond Repair

1. **Hysol EA 9321**
   a) Storage Requirements: Hysol Primer Part A has a shelf life of 1 year from date of manufacture when stored at 40°F or below or 3 months when stored at 77°F or below. Hysol Primer Part B has a shelf life of 1 year from date of manufacture when stored at 77°F or below. This shelf life is applicable to both unopened and opened containers, as long as they are kept tightly closed after opening.
   b) Curing Agent Content: 100:50 wt%
   c) Pot Life: 40 minutes
   d) Mixing Instructions: See attached product information sheets
   e) Filler Ratio: Up to 5% Flock by weight.
2. **Loctite 330**
   a) Storage Requirements: Loctite does not have a published shelf life.
   b) Curing Agent Content: This is a single component adhesive, but should be used with Activator 7387 as described on the attached product information sheets.
   c) Pot Life: 15 minutes
   d) Mixing Instructions: See attached product information sheets
3. **Jeffco 1307-LV with 3102.**
   a) Storage Requirements: Store in a closed container at or above 50°F. Unopened shelf life is 2 years from the date of manufacture. Open shelf life is 1 year.
   b) Curing Agent Content: 100:22-28 by weight (4:1 by volume)
   c) Pot Life: 11-15 minutes
   d) Mixing and Cure Instructions: Use epoxy within gel time (pot life). Allow to set at room temperature for a minimum of 3 hours.[3] After room temperature set, cure at room temperature (77°F) for 7 days or at 212°F for 2 hours. Use Figure 51 - 20 for other time and temperature options.
   e) Filler Ratio: Up to 40% by weight of: Micro, Micro/Flock mixture, or "Filler Mix."

Cure Temperature (°F)	Cure Time (Hours)
77	168
87	76
97	40
107	24
117	16
127	11
137	8.0
147	6.1
157	4.8
167	4.0
177	3.3
187	2.8
197	2.4
207	2.1
217	1.9

[3] Bonded assembly may be further worked with after this point, but repair work is not complete until cure cycle is complete.

Figure 51 - 20 Jeffco Cure Requirements

4. **Plastic Welder**
   a) Storage requirements: Devcon does not have a published shelf life.
   b) Pot life: 4-6 minutes
   c) Mixing instructions: This is a prepackaged adhesive. No mixing is required.
   d) Cure and handling instructions: The adhesive will have handling strength in 10-15 minutes and will fully cure in 45-60 minutes at room temperature.
5. **Plastic Welder 2**
   a) Storage requirements: Devcon does not have a published shelf life.
   b) Pot life: 15-20 minutes.
   c) Mixing instructions: This is a prepackaged adhesive. No mixing is required.
   d) Cure and handling instructions: The adhesive will have handling strength in 30-35 minutes and will fully cure in 120-180 minutes at room temperature.

Purpose	Material
Solvent Wash	MEK Acetone Isopropyl Alcohol **NOTE:** Reclaimed solvents are not acceptable.
Wipe	Lint and solvent free cheese cloth or equivalent cloth that is free of oil and soap
Abrasion	Scotch-Brite™ pad 80 or 120 grit sand paper or equivalent abrasive

Figure 51 - 21 Approved Surface Preparation Materials

b. **Bond Repair Method**
   1. Using sand paper or by bead blasting remove as much of the adhesive from the bonding surfaces of both parts in area of bond failure as possible without causing damage to the composite part.
   2. On surfaces which are to be bonded perform the following steps which constitute surface preparation:
      a) Wash area with approved solvent and wiping cloth.
      b) Abrade surface of area with approved abrasive.
      c) Remove abrading dust using a vacuum.
      d) Wipe surface of area with an approved wiping cloth (do not reuse any wiping cloths) to remove residual dust.
      e) Wash area with a wiping cloth (do not reuse any wiping cloths) saturated with an approved solvent. Thoroughly wet the area with approved solvent.
      f) Immediately wipe area with wiping cloth (do not reuse any wiping cloths) using a single stroke motion while rolling the wipe away from the surface. After each wipe turn the drying cloth to a clean surface.

NOTE

Steps 2e & 2f must be performed within 2 hours of the application of the adhesive to the areas to be rebonded (done at Step 7 below). If more than 2 hours elapse between steps 2e/2f and step 7, steps 2e and 2f must be performed again.

3. Select an approved adhesive and mix (if required) per directions in Table 10. Follow all safety guidelines as provided by the supplier or noted in an MSDS sheet.
4. Apply adhesive to bonding surfaces in area of bond failure.
5. In the case of complete bond failure where two parts are completely separated: Apply the adhesive in a pyramid formation on one bonding surface. This will assure an adequate spread of the adhesive over the bonding surfaces during close out.
6. In the case of bond failure where two parts are still connected by bond: Apply the adhesive to the area between the two bonding surfaces.
7. Apply pressure between the two parts in area of initial bond failure. Remove any excess resin which may have squeezed out of the bonding area.
8. While continuing to apply pressure cure per requirements of the adhesive used.
9. Restore primer and paint as necessary using methods outlined in those areas.

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51-11. EXPANDED METAL FOIL REPAIR

Expanded metal foil is used on Cessna aircraft for lightning strike protection, precipitation static conduction from the impact areas to the static wicks, antenna ground plane, and fuel cap ring grounding to the airframe during refueling. Both aluminum and copper foil are used on the aircraft.

a. **Approved Repair Materials**

Product	Supplier	Product ID
Bare Aluminum Mesh	Astroseal Products	AL016CXMC or AL016CX
Bare Copper Mesh	Astroseal Products	CU029CXMC or CU029CX
Microspheres	3M	K15 Glass Bubbles
Epoxy Resin	Jeffco Products	1307-LV Epoxy Resin with 3102 Hardener
Sealant Tape	Commercial	AT-200Y
Peel Ply	Precision Fabrics Group	Polyester Peel Ply, Style 56030 with 060 finish (natural – scoured and heatset)
Breather Cloth	Airtech Advanced Materials Group	Airweave N-4 60” or 3”
Perforated Film	Commercial	Release Film, 0.00075” thick, perforated, 0.015” holes on 0.25”
Vacuum Bagging Material	Commercial	Single or V-Fold Bagging Film
Methyl Ethyl Ketone (MEK)	Commercial	Technical Grade
Isopropanol	Commercial	99% Anhydrous
Acetone	Commercial	Commercial Grade

b. **Surface Preparation** (See Figure 51 - 22)
   1. Sand the repair area thoroughly at least 3 inches beyond damaged area using 80 grit sand paper on glass parts and 120 grit sand paper on carbon parts.
   2. Body work the affected areas and finish with 120 or 180 grit sand paper.
   3. Prepare surrounding area at least 3 inches around body work to expose bare foil, taking care not to sand through the foil in excess of the constraints defined in paragraph 51-4.b.1. This is to allow electrical contact for new foil to provide continuity over the repair area.
   4. Clean all areas to be re-foiled with an approved solvent and allow to dry.

c. **Material Preparation** (See Figure 51 - 22)
   1. Cut the foil so there is at least a 1 inch overlap onto the existing foil. Use copper to repair copper, and aluminum to repair aluminum. If there is an edge, bond seam, or if the repair abuts non-foiled area, the 1 inch overlap does not apply.
   2. Cut peel ply at least 1 inch larger than the foil.
   3. Cut perforated film the same size as peel ply.
   4. Cut breather cloth 1 inch smaller than peel ply.

d. **Material Application** (See Figure 51 - 22)

1. **Repair With Vacuum**

a) Lay sealant tape down at least four inches outside the bodywork area.
b) Re-clean areas to be repaired with an approved solvent.
c) Mix Jeffco resin and apply to surface using a brush or spreader, thoroughly wet the entire area to be repaired. Use epoxy within gel time (pot life).
d) Store Jeffco 1307-LV with 3102 hardener in a closed container at or above 50ºF. Unopened shelf life is 2 years from the date of manufacture. Open shelf life is 1 year.
e) Curing agent content: 100:22-28 by weight (4:1 by volume).
f) Pot Life: 11-15 minutes.
g) Apply foil to the area and work down smooth to surface making sure there is a one-inch minimum overlap to the existing bare exposed foil for electrical contact.
h) Wet out area again with resin mixture and apply peel ply over the expanded aluminum or expanded copper foil.
i) Wet out peel ply lightly and apply perforated film over peel ply.
j) Apply breather cloth over the perforated film.
k) Fold a small piece of breather cloth four layers thick for placement of the vacuum valve assembly in the center of repair area, and then close out repair area with bagging material. Cut a hole in the center of the bagging material for installation of the vacuum valve.
l) Apply vacuum.

2. **Repair Without Vacuum**

a) Re-clean areas to be repaired with an approved solvent.
b) Mix Jeffco resin and apply to surface using a brush or spreader, thoroughly wet the entire area to be repaired.
c) Store Jeffco 1307-LV with 3102 hardener in a closed container at or above 50ºF. Unopened shelf life is 2 years from the date of manufacture. Open shelf life is 1 year.
d) Curing agent content: 100:22-28 by weight (4:1 by volume).
e) Pot Life: 11-15 minutes.
f) Apply expanded aluminum or expanded copper foil to the area and work down smooth to surface making sure there is a 1 inch minimum overlap to existing bare exposed foil for electrical contact.
g) Wet out area again with resin mixture and apply peel ply over the expanded aluminum or expanded copper foil
h) Wet out peel ply lightly and apply perforated film over peel ply.
i) Apply breather cloth over the perforated film.
j) Cover area with bagging film and weigh down.
k) If the area cannot be weighed down because of location, firmly tape down the bagging material.

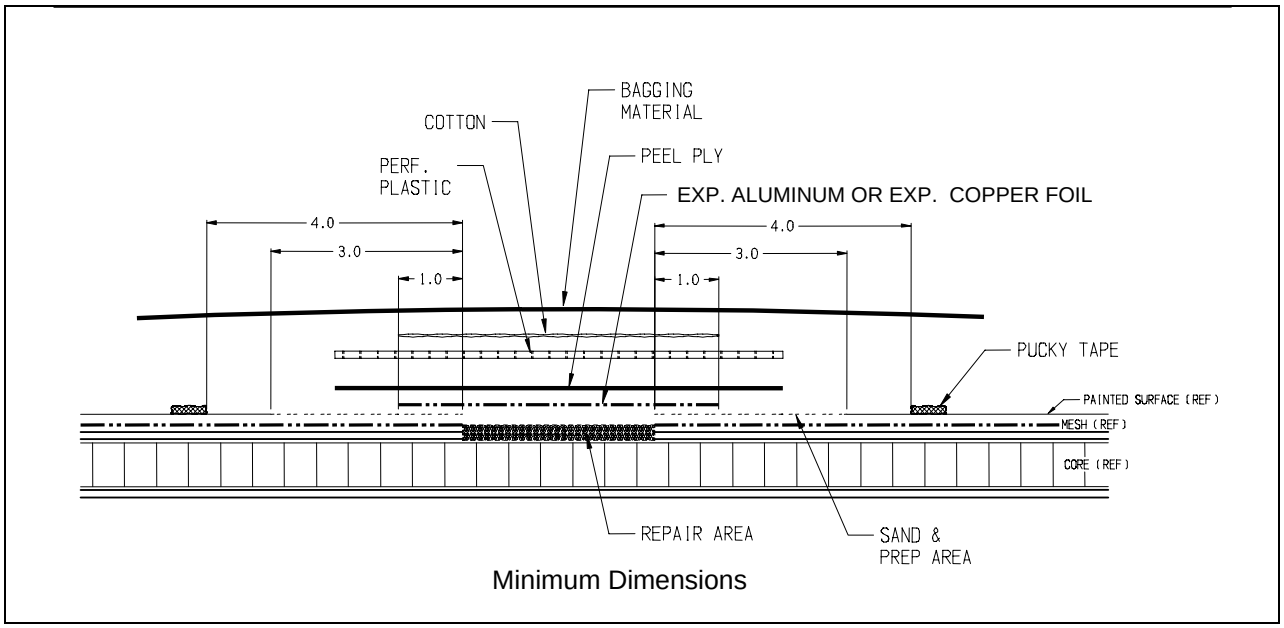

Figure 51 - 22 Repair Profile

3. **Curing**
   a) Allow resin to set at room temperature for a minimum of 3 hours.
   b) After room temperature set, cure at room temperature (77°F) for 7 days or at 212°F for 2 hours. Use the table below for other time and temperature options.

Cure Temperature (°F)	Cure Time
77	168
87	76
97	40
107	24
117	16
127	11
137	8.0
147	6.1
157	4.8
167	4.0
177	3.3
187	2.8
197	2.4
207	2.1
217	1.9

4. **Repair Completion**
   a) Remove bagging materials and tape.
   b) Clean area with an approved solvent.
   c) Visually inspect the repair for wet out and adhesion.
   d) Perform conductivity test, using a digital ohmmeter.
   e) For Zones 1, 2, and 3, abrade the expanded aluminum or expanded copper foil to allow good connection and check the repaired area to a local area of the ground

plane close to the repair. Resistance should measure less than 10 ohms. Higher readings require the repair to be redone.

f) For antenna and static wick mounting pads, abrade the expanded aluminum or expanded copper foil to allow good connect and check the repaired area to a local area of the ground plane close to the repair. Resistance should measure as less than 1 ohm. Higher readings require the repair to be redone.
g) Sand area with 120 grit sand paper.
h) If the area is not under an antenna or static wick, feather out with body filler and paint as indicated in paragraph 51-6.

CHAPTER 52

DOORS

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Chapter 52

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52-1. GENERAL DESCRIPTION OF CABIN AND BAGGAGE DOORS

a. **Cabin Doors** – Each cabin door is constructed of an inner and outer composite panel bonded together. The door skin is constructed of carbon cloth. The doubler or door inner panel is constructed of glass and carbon cloth. The doors are mounted in a gull wing configuration with the hinges located at the top of the aircraft. With the door unlatched, a slight nudge or pull will start the opening of the door, and a gas strut will assist to fully open the door.

b. **Latching Mechanism** – From the exterior, the latching mechanism on each cabin door is operated through movement of the exterior door handle. The handle is mounted on the side of the door in the bottom, aft position and has two ranges of movement. The handle is recessed into the door with adequate room for a handhold. A safety release on the handle must be disengaged before the door will open. The release is activated by pulling the handle away from the door. Moving the forward end of the handle from its normal middle position to the six o'clock position disengages the latching mechanism. To secure the door, return the handle to the middle position.

   In the full open position, the handle will be somewhat lifted in an internal mechanism detent. This prevents the latching pins from striking the side of the fuselage as the door is closed. After closing the door, lifting the handle will free it from the detent and allow it to rotate and engage the latching pins.

   From the interior, both latching mechanisms are engaged and disengaged through use of a handle near the bottom, aft position of the interior door. Again, the safety release is disengaged by pulling the handle away from the door. To activate the latching mechanism, move the door handle down from its near horizontal position until the slide bolts are fully engaged and the curved end of the handle is resting in the safety detent. There are placards on the interior doors labeled "Open" and "Closed" with direction arrows. When both doors are properly closed with the latching mechanism engaged and the baggage door is locked, the "Door Open" annunciator in the upper left position of the annunciator panel will be extinguished.

c. **Door Locks** – The right cabin door can only be locked from the inside using a finger/thumb latch. It is placarded to remain unlocked during flight. The left cabin door can only be key locked from the outside. A distinction is made here between the terms *latching mechanism* and the security *door locks*. The latching mechanism ensures that the doors will remain secured during flight. The door locks are primarily antitheft devices and restricts use of the latching mechanism.

d. **Door Hinges** – Each door is hinged at the top of the door with two hinges. Half of the hinge is attached to the cabin door. The other half of the hinge, the hinge block, is bolted into cutouts in the top of the fuselage. There are hinge pins attached to the hinge element, which are secured with a dowel. The hinge pins of each hinge are inserted into the hinge block, however, they are secured differently on each side. On the left side the pins are held in the hinge block by the emergency exit wires, and on the right side the pins are secured with roll pins. See Figure 52 - 1.

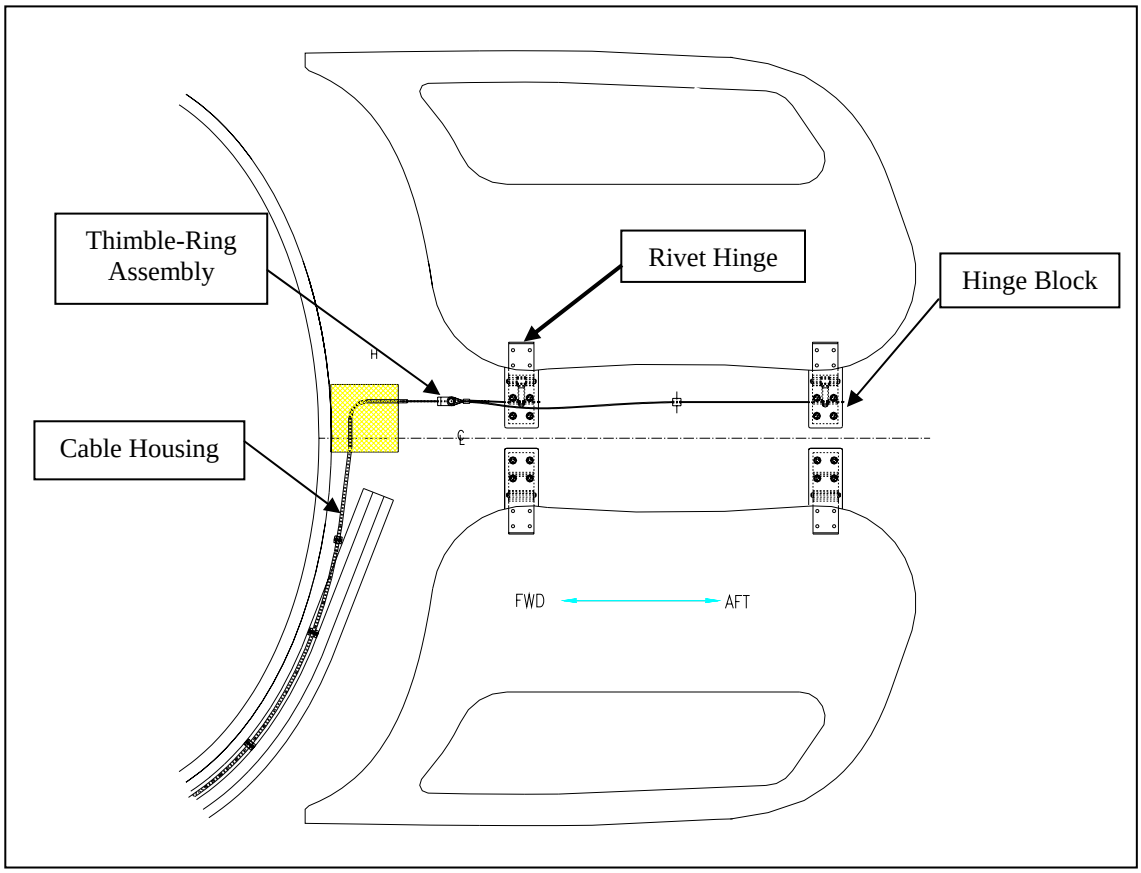

Figure 52 - 1 Door Hinges

e. **Emergency Exit Door Release** – There is an emergency door release located in the forward left wing saddle in the compartment that contains the fuel gascolator. The release cable is designed to release the hinges on the pilot's door during emergency egress from the airplane when it is in the inverted position, and the passengers are incapacitated. Detailed operational information is contained in the *Pilot's Operating Handbook and FAA Approved Airplane Flight Manual*.

The emergency door release cable is in a spiral metal housing that runs from the gascolator compartment, under the front seat box, and exits the seat box on the copilot's side. From this point, the cable runs forward to the right kick-panel, then up through the right rollover structure and across the front of the windshield to an area near the left, forward overhead light panel. See Figure 52 - 1. When the emergency cable strap in the gascolator compartment is activated, the steel wires that run through the hinge pins and hinge block are pulled out of the hinge block, which permits removal of the door, once the latching mechanism is disengaged. See Figure 52 - 2.

The forward part of the cable is connected to a thimble-ring, which is part of the thimble-ring assembly. The thimble-ring contains a stainless steel wire with a Nicopress swage collar. One of the steel wires runs through the forward hinge block of the left door, through a hole in the hinge pin, and exits slightly aft of the hinge block. The other

stainless steel wire runs through a guide block and then into the aft hinge block in the same manner as the forward hinge block. See Figure 52 - 3.

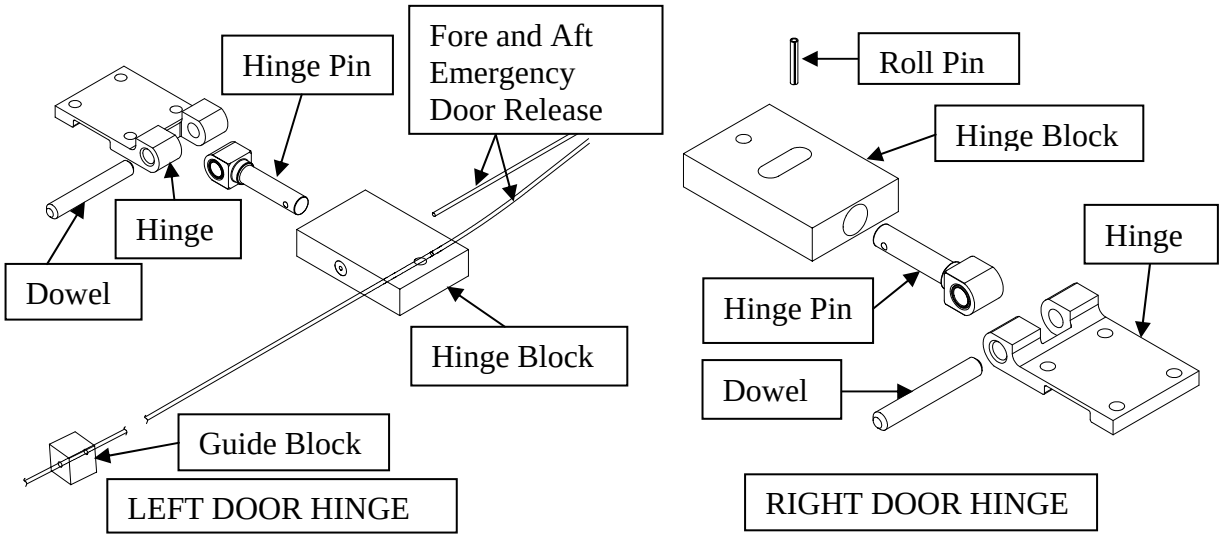

Figure 52 - 2 Door Hinge Assembly

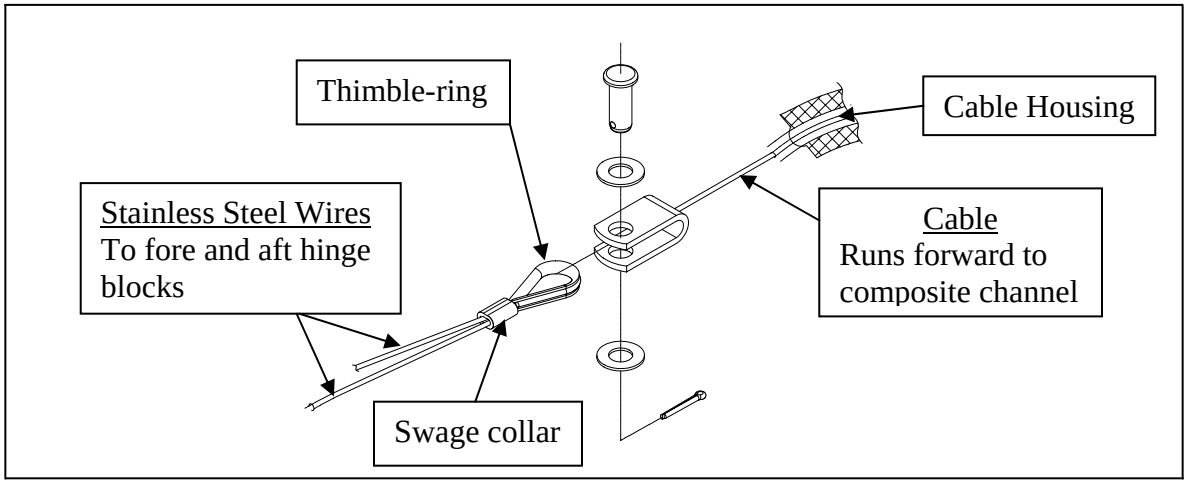

Figure 52 - 3 Emergency Door Release Cable

f. **Baggage Door** – The baggage door is piano hinged at the top and opens with a gas spring assist similar to the cabin door. The baggage door is key locked fore and aft by a common key, which is the same key that is used in the left cabin door lock. The forward lock of the baggage door retains the key so that it can only be removed when the lock is in the latched position. The aft baggage door lock, like the left cabin door lock, allows key removal in either the latched or unlatched positions.

To open the baggage door, insert the key into each lock and rotate 90º clockwise. Open the aft lock first. Once the aft lock is unlatched, remove the key, and open the forward lock. This design reduces the possibility of taking off with the baggage door open, provided the ignition and baggage door keys are on the same key ring. When the second

lock is unlatched, the gas strut will raise the door. The baggage door is part of the door annunciator system. If the baggage door is not properly closed and the forward latch secured, the red “Door Open” light in the annunciator panel will illuminate.

g. **Door Seal System** – The airplane is equipped with a pneumatic door seal system that limits air leakage and improves soundproofing. An inflatable gasket around each main door expands when the door seal system is turned on. An electric motor near the pilot’s rudder pedals operates the system, which maintains a differential pressure of 12 to 15 psi. The system is activated by a switch in the rocker switch panel labeled “Door Seal” and is protected by a five-amp circuit breaker. The cabin doors must be closed for the door seal system to operate. The latching mechanism of each door moves a plunger-type switch, which, assuming the baggage door and the other main door are properly latched, turns off the door open annunciator and enables the system.

   The cabin door latching mechanism also controls the dump valve. When either cabin door latching mechanism is moved more than a half inch towards the open position, the dump valve is engaged, the pump is turned off, and the pressure in the seals is dumped. This prevents inadvertent operation of the doors when they are sealed; however, setting the doors seal switch to the off position after landing is recommended.

   Normally, the door seal switch remains in the on position for the entire flight. If the system pressure drops below 12 psi, the air pump will cycle on until pressure is restored. If the pump runs continuously, it is an indication that a seal is damaged and incapable of holding pressure. In this situation, the door seal system should not be operated until repairs are made.

52-2. CABIN DOOR REMOVAL

a. Removal of the cabin door can be accomplished as follows.

1. Disconnect the pneumatic door seals using the following steps.

a) With the door in the full open position, use a Phillips screwdriver to loosen the Adel Clamp on the air supply hose near the air supply nipple in the overhead panel. (See Figure 52 - 4)

b) Disconnect the supply hose from the nipple in the overhead panel.

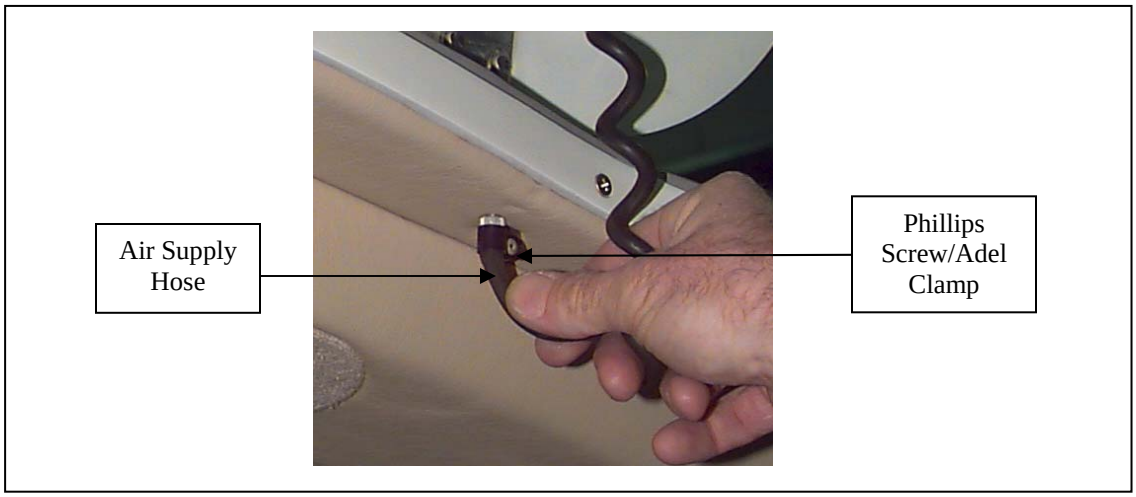

Figure 52 - 4 Pneumatic Door Seal Removal

2. Disconnect the gas strut from the door jam using the following steps.

a) Support the door in the full open position so that all weight is removed from the gas strut.

b) The gas strut is secured to the door by a retaining cap located in the end of the strut. Remove the strut retaining cap by inserting a small flat-blade screwdriver just under the base of the cap and prying up on it. Raise the cap in increments by moving the blade of the screwdriver around the circumference of the cap. (See Figure 52 - 5.)

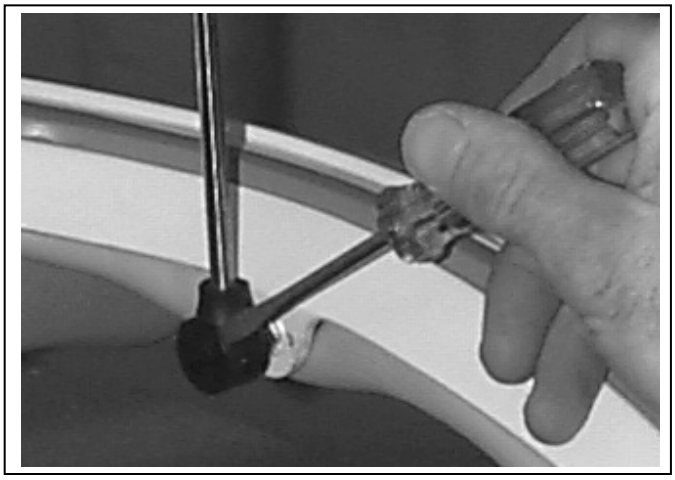

Figure 52 - 5 Gas Strut Removal

3. There are two methods for removing the doors. (1) Quick Removal–door removal without removing the interior hinge bolts, and (2) Standard Removal–removal of the interior hinge bolts. The Quick Removal method may only be used for removal of the left door. The right door must be removed with the Standard Removal technique. The technique used depends on the maintenance task.
4. Overhead Panel Description – There are basically five panels in the overhead portion of the main cabin area.
   a) **Center Overhead Light Panel (COLP)** – This panel is located in the center portion of the main cabin area and contains the electrical lights and speaker. Refer to Chapter 25 in this manual for removal of the COLP.
   b) **Forward Overhead Side-Panels (FOSP)** – The forward panels to the left and right of the COLP are referred to as the left and right overhead side-panels. Remove the left and right forward panels by removing the six forward screws in each door casing.
   c) **Aft Overhead Side-Panels (AOSP)** – These side panels are just aft of the FOSP and normally require only removal of the first two screws in the door casing that supports these panels. Normally, removing these two screws will provide sufficient access to the aft hinge bolts and emergency door release cable. Care must be taken to not damage these panels when removing/installing the hinge bolts and emergency door release stainless steel wire.

52-3. QUICK REMOVAL LEFT DOOR

a. Follow the steps below for quick removal of the left door.
   1. Remove the pneumatic air supply line and gas strut as described in steps 1 and 2 of paragraph 52-2.
   2. Remove the COLP, left FOSP, and the forward two screws of the left AOSP.
   3. Close the door, and engage the latching mechanism.
   4. Pre-load the door, i.e., a person on the outside of the door should push in and down.
   5. Grasp the thimble-ring, which is connected to the emergency door release wires, and pull forward toward the windshield until the stainless steel wires are removed from the hinge blocks. The thimble-ring is located forward of the door cutout in the area of the left overhead panel. See Figure 52 - 1. It may work better to grasp the thimble-ring with needle-nose pliers.
   6. From the outside, disengage the door locking mechanism, and carefully open the door while keeping pressure on the door hinges. When the door is about half open, pull it away from the cabin. It works best if two people perform this operation. While the door is not too heavy, its size and shape makes it somewhat awkward to handle with just one person.

52-4. INSTALLATION FOR LEFT DOOR (Quick Removal Method)

a. Use the following steps for installing the left door after it was removed using the Quick Removal Method.
   1. Insert the hinge pins into the hinge blocks. It works best if two people do this since the size and shape of the door makes it somewhat awkward to handle. Also, the door should be held in about the half open position while the hinge pins are inserted into the hinge blocks.

2. After the hinge pins are inserted in the hinge block, slowly rotate the door into the door opening, and secure the door latching mechanism.
3. Preload the door, i.e., a person on the outside of the door should push in and down.
4. Apply wax or mold release to the last 3 in. of the wire, then with the door preloaded, use needle-nose pliers to reinsert the fore and aft stainless steel wires through the hole in the composite cutout and into the hinge block. Ensure that the wire goes through the block and exits each hinge block on the aft side.
5. Reinstall all overhead panels that were removed or partially removed. Follow instructions in Chapter 25 for the COLP. Torque screws in the FOSP and AOSP 10 to 12 in.-lbs.
6. Reconnect the gas strut and. the pneumatic air supply line using the reverse of the procedures described in paragraph 52-2.

52-5. STANDARD REMOVAL OF THE LEFT DOOR

a. **General** - The "Standard Removal" of the left door involves removal of the hinge blocks. See Figure 52 - 2. The hinge blocks are secured by four bolts that are inserted into threaded holes in the bottom of the block. The bolts, which have a $^{3}/_{8}$ in. hex head, are accessed from within the cockpit. The following points must be considered when removing the hinge blocks.
   1. The locations of the hinge blocks are **not** interchangeable. Accordingly, each hinge block must be labeled with its location prior to removal, i.e., LF, LR, etc.
   2. The screws on the dynamic hinge need not be removed unless there is a compelling reason to do so, e.g., a cracked hinge or a door replacement. Contact the manufacturer for specific instructions for removing and installing the screws on the door hinge.

b. Use the following procedure for the Standard Removal of the left door
   1. Follow the instructions in paragraph 52-3 Quick Removal Left Door steps 1 through 6.
   2. Remove the lockwire from the hinge bolts.
   3. Using a $^{3}/_{8}$ in. socket wrench, back all the bolts about halfway out of the hinge block.
   4. Apply an upward pressure on the hinge bolts to break the silicone seal around the base of both the forward and aft hinge blocks.
   5. Remove all hinge bolts from the hinge block.
   6. Remove all excess silicon from the hinge block and cutout in the top of the fuselage.

52-6. STANDARD REMOVAL OF THE RIGHT DOOR

a. **General** - The "Standard Removal" of the right door involves removal of the hinge blocks. See Figure 52 - 2. The hinge blocks are secured by four bolts that are inserted into threaded holes in the bottom of the block. The bolts, which have a $^{3}/_{8}$ in. hex head, are accessed from within the cockpit. The following points must be considered when removing the hinge blocks.
   1. The locations of the hinge blocks are **not** interchangeable. Accordingly, each hinge block must be labeled with its location prior to removal, i.e., RF, RR, etc.
   2. The rivets on the rivet hinge should never be removed unless there is a compelling reason to do so, e.g., a cracked rivet hinge or a door replacement. Contact the manufacturer for specific instructions for removing and installing the rivets on the door hinge.

b. Use the following procedure for the Standard Removal of the right door
   1. Remove the pneumatic air supply line and gas strut as described in steps 1 and 2 of paragraph 52-2.
   2. Close the door and engage the latching mechanism.
   3. Remove the roll pin in the right door hinge blocks. See Figure 52 - 2.
   4. From the outside, disengage the door locking mechanism, and carefully open the door while keeping pressure on the door hinges. When the door is about half open, pull it away from the cabin. It works best if two people perform this operation. While the door is not too heavy, its size and shape makes it somewhat awkward to handle with just one person.

NOTE

It is permissible to remove the right door, using the Standard Door Removal, without disconnecting the roll pins. If this technique is used, the hinge block remains attached to the right door. While this eliminates the need to remove and possibly damage the roll pin, it does require removal of interior paneling and breaking the silicon seal. If the interior panels are removed, follow instructions in paragraph 52-7 to reinstall the hinge blocks.

   5. Remove the lockwire from the hinge bolts.
   6. Using a $^{3}/_{8}$ in. socket wrench, back all the bolts about halfway out of the hinge block.
   7. Apply an upward pressure on the hinge bolts to break the silicone seal around the base of both the fore and aft hinge blocks.
   8. Remove all hinge bolts from the hinge block.
   9. Remove all excess silicon from the hinge block and cutout in the top of the fuselage.

52-7. INSTALLATION OF HINGE BLOCKS AND DOORS

a. Use the following steps for installing the door hinge blocks.
   1. Place the hinge block over its appropriate position as previously marked.
   2. Install the No. 10 hinge block bolts (four per hinge) and torque 30 to 36 in.-lbs.
   3. Lockwire the four bolts on each hinge per MS33540.
   4. Place a small bead of RTV 5818 (or equivalent) silicone around the base of each hinge block. Do not allow RTV on hinge wires.
   5. Install the left door per instructions in paragraph 52-4 Installation for Left Door (Quick Removal Method) steps 1 through 6.
   6. Install the right door per the following instructions:
      a) Insert the hinge pins into the hinge blocks. It works best if two people do this because the size and shape of the door makes it somewhat awkward to handle. Also, the door should be held in about the half open position while the hinge pins are inserted into the hinge blocks.
      b) After the hinge pins are inserted in the hinge block, slowly rotate the door into the door opening, and secure the door latching mechanism.
      c) Preload the door, i.e., a person on the outside of the door should push in and down.
      d) With the door preloaded, installed the roll pin in the hinge block.

7. Reinstall all overhead panels that were removed or partially removed. Follow instructions in Chapter 25 for the COLP. Torque the screws in the FOSP and AOSP 10 to 12 in.-lbs.
8. Reconnect the gas strut and. the pneumatic air supply line using the reverse of the procedures described in paragraph 52-2.

52-8. BAGGAGE DOOR REMOVAL AND INSTALLATION

a. Use the following steps for baggage door removal. See Figure 52 - 6.
   1. Remove the left side push-pull tube channel cover in the baggage compartment by first removing the fore and aft screws on top of the channel cover and sliding it out of the floor clips.
   2. Remove the gas strut from the door in the same manner as described in paragraph 52-2.a.2
   3. Remove the covered cabin door surround inside the baggage compartment for access to the self-locking nuts (Nyloks). The surround is secured by two self-locking nuts with washers on the bottom and is inserted in clips on the top of the door opening. Remove the nuts and screws on the bottom of the surround, and slide down to remove the surround from the clips. If the surround does not easily slide down, the clips are rotated 180º and the surround must be slid up then out and down.
   4. Remove the baggage door piano hinge by removing the six, No. 8 x ½ in. screws with self-locking nuts (Nyloks) that hold it to the top of the doorjamb. Note: The two top clips which hold the cabin door surround are removed when the piano hinge is removed.

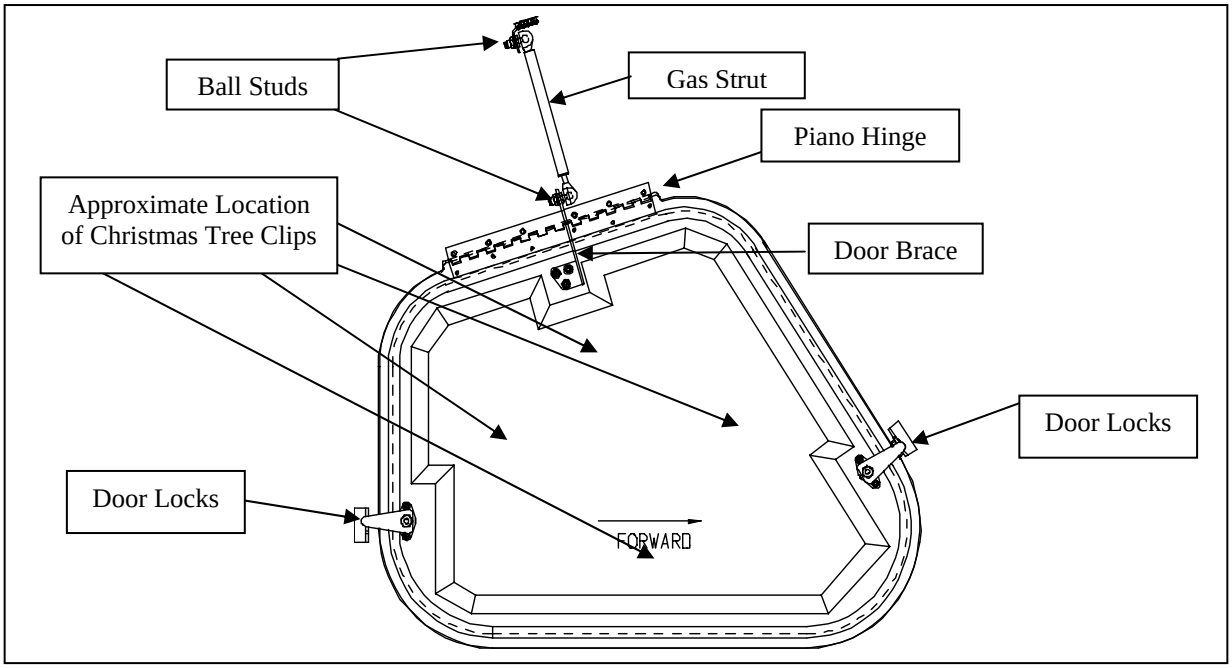

Figure 52 - 6 Baggage Door

b. To install the baggage door use the following steps.
   1. Using two people, attach the piano hinge and torque the six screws 20 to 25 in.-lbs. Note: Be sure to reattach the clips for the surround.

2. Reinstall the baggage door surround. First, insert the top portion into the clips and then attach the screws to the base. Torque screws 30 to 36 in.-lbs.
3. Reinstall the gas strut onto the ball stud.
4. Install the push-pull tube channel cover and torque screws 30 to 36 in.-lbs.

c. Other parts on the baggage door have the following specifications.
1. If the three screws with self-locking nuts on the door brace are removed and installed, they should be torqued 30 to 36 in.-lbs.
2. If the ball studs on the door brace or overhead bracket are removed and installed, they should be torqued 140 to 180 in.-lbs.

52-9. CABIN AND BAGGAGE DOOR LOCKS

a. **Left Door Lock** – To remove the left door lock, use the following steps.
1. Set the door latching mechanism to the open position, open the door completely, and stand on the skid strips of the left wing. Ensure that the door lock is in the unlocked position.
2. Remove the interior door handle by removing the aft cotter pin and clevis as shown in Figure 52 - 7. Note, there are two clevis pins in the door handle.
3. When the clevis is removed, slide the handle aft to remove it.
4. Remove the four Phillips screws on the interior door access panel, and remove the panel. Note, a spacer around the door mechanism will normally fall out as the panel is removed.
5. Mark the position of the drive tab on the interior of the door panel. This is a convenient reference when reinstalling the door lock.
6. Remove the two pan head Phillips screws that hold the door lock in place, and remove the door lock. See Figure 52 - 7.

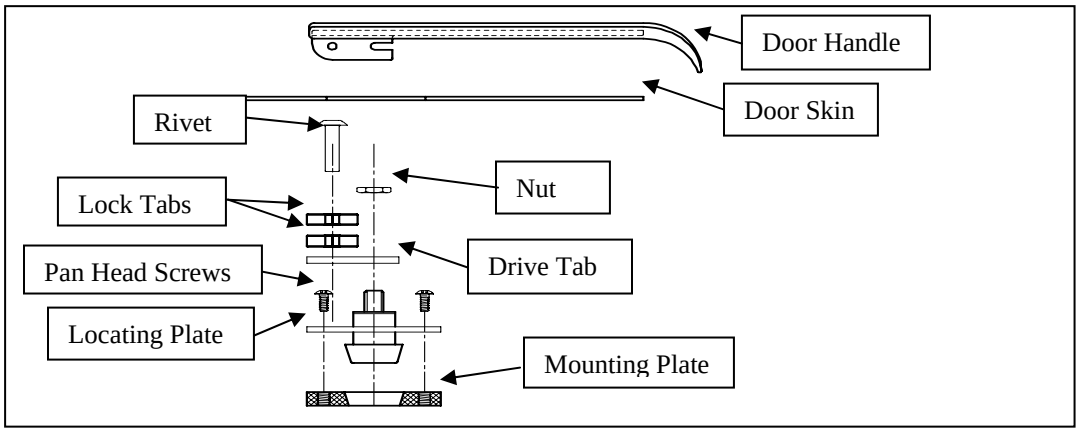

Figure 52 - 7 Left Door Lock Assembly

b. **Right Door Lock** – To remove the right door lock, use the following steps.
1. Place the thumb latch in the locked position, and remove the four Phillips screws holding the right door access panel in place. Remove the panel.
2. Before removing the right door lock from the access panel, mark the inside of the access panel with the location of the drive tab when in the locked position. This is a convenient reference when reinstalling the door lock. Note that when viewing the

panel from the exterior, the thumb latch is in the three o'clock position, and the drive tab (behind the panel) is in the one o'clock position. See Figure 52 - 8.

3. Remove the latch screw and lock washer. It is important to mark the relationship of the latch to the drive tab, the drive tab to lock, and lock to the access panel since the drive tab can be assembled upside down and rotated to any of four positions.
4. Remove the nut from the lock assembly to remove the lock.

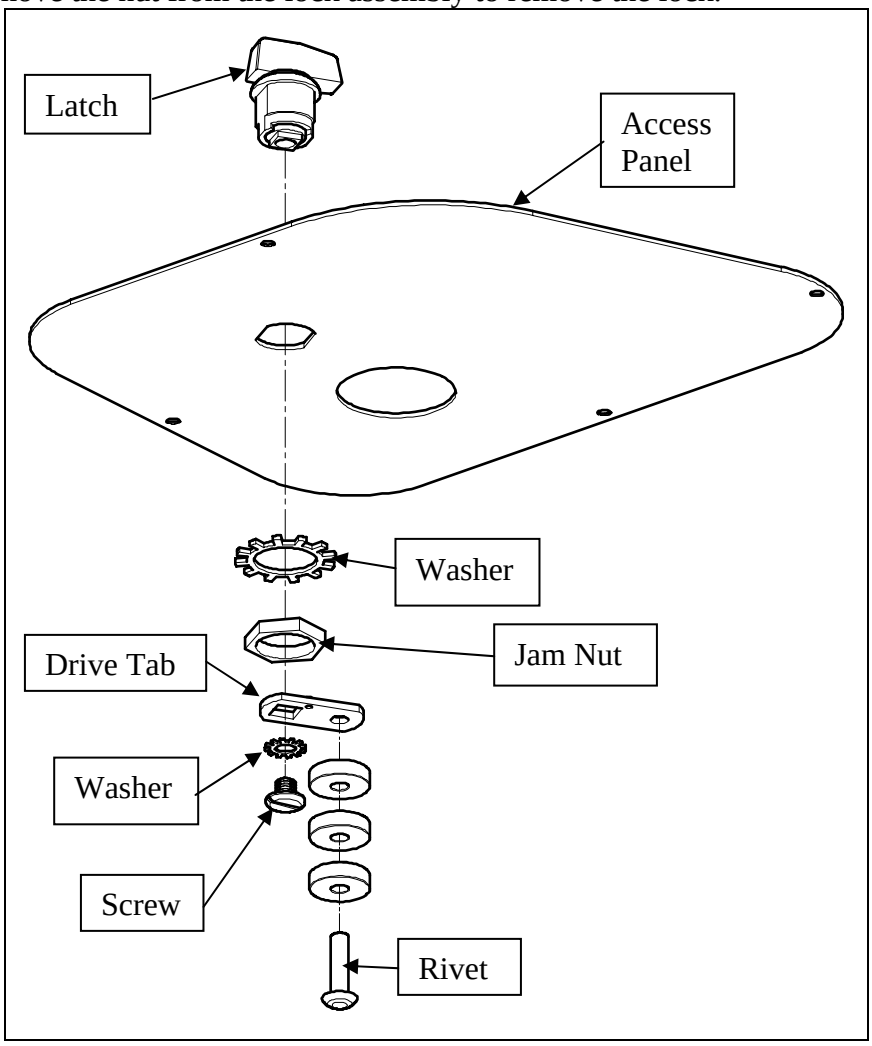

Figure 52 - 8 Right Door Lock Assembly

c. **Baggage Door Locks** – To remove the baggage door locks, use the following steps. See Figure 52 - 9.

1. Remove the upholstered panel on the interior of the door. The panel is held onto the baggage door with four Christmas Tree clips approximately arranged in the shape of a diamond. **Do not pull the upholstered panel by the edges since this can cause damage to the panel.**
2. It works best to insert a hand between the panel and the door with the palm facing the panel. Move palm towards the area of the bottom or right clamp, and force the panel away from the door.
3. Note the orientation of the door lock, and make a reference mark for reinstallation.

4. Remove the two No. 6 self-locking nuts (Nyloks), the two washers, locating plate, and the lock.
5. The self-locking nut holding the lock tab can be removed if the tab needs to be replaced.

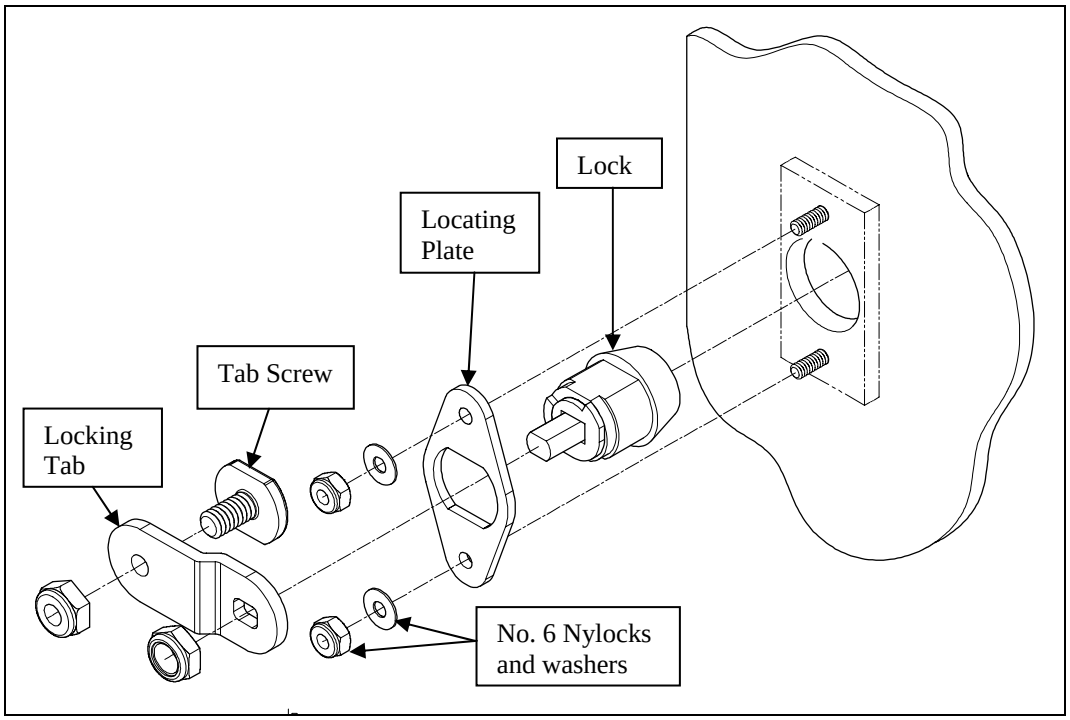

Figure 52 - 9 Baggage Door Lock

52-10. INSTALLING DOOR LOCKS

a. To install the locks, reverse the steps described in paragraph 52-9 for each particular door lock. However, the following items need to be considered.
   1. Ensure that the door locks are oriented in the proper direction since it is possible to install both the left and right locks backwards or upside down.
   2. Access panel screws should be tightened until they bottom out.
   3. The two screws holding the lock on the **left door** should be torqued 12 to 15 in.-lbs.
   4. The **screw** on the **right door** lock should be torqued 38 to 44 in.-lbs.
   5. The **nut** on the **right door** lock should be torqued 100 to 120 in.-lbs.
   6. The two No. 6 **baggage door** self-locking nuts should be torqued 6 to 8 in.-lbs.
   7. The self-locking baggage door nut that retains the lock tab should be torqued 36 to 42 in.-lbs.
   8. It is permissible to install one AN690-516 washer between the lock and the lock tab if required. Two threads must extend past the nut attaching the tab.

52-11. CABIN AND BAGGAGE DOOR ANNUNCIATOR SYSTEM

a. **Cabin Doors** – The cabin door annunciator switch is mounted on the forward door pin receptacle in the fuselage. The switch is retained with two opposing setscrews. The left or

right FOSP (See paragraph 52-1.a.4 for details) must be removed to gain access to the switch. To remove or adjust the switch, use the following steps. See Figure 52 - 10.

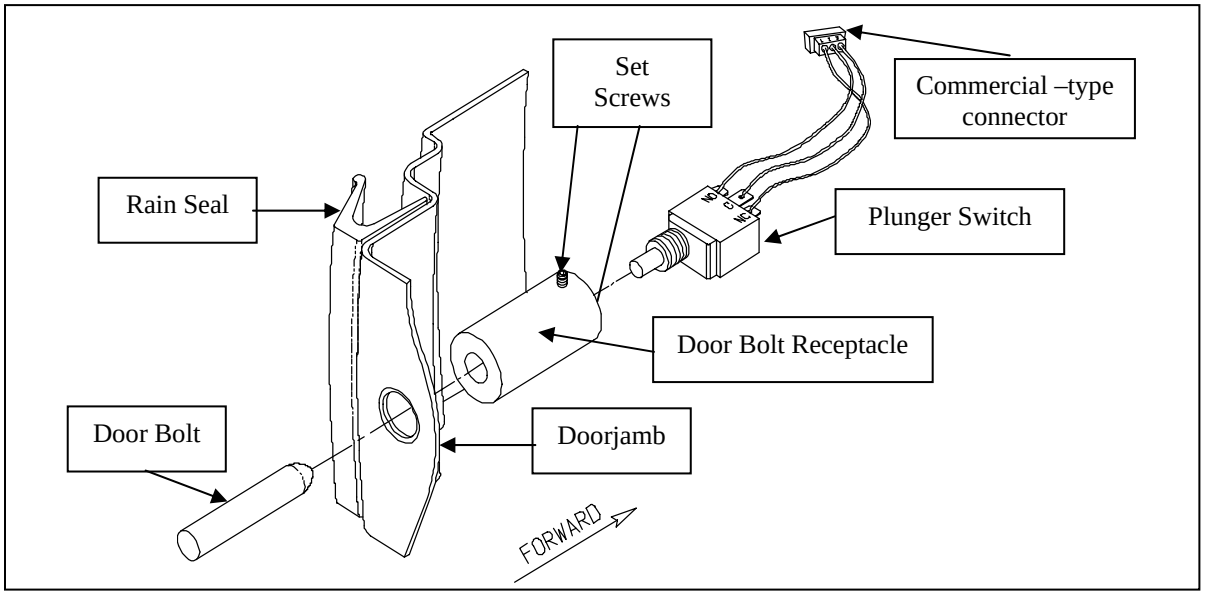

Figure 52 - 10 Cabin Door Annunciator Switch Assembly

1. **General** – With the right or left side panel removed (depending on which switch is accessed), loosen the set screws on the top and bottom of the door bolt receptacle.
2. **Installation or Adjustment of Door Annunciator Switch** – With the door closed and latched and the baggage door locked, move the switch aft towards the door bolt until it activates the plunger switch (turns off the Door Open annunciator), then move the switch an additional 0.025 in. aft. With the set screws removed, apply Form-A-Gasket #2 by Permatex to the threads, and reinsert and tighten the set screws.
3. **Switch Removal** – To remove the switch, loosen the two set screws, unplug the commercial-type connector, and remove the switch.

b. **Baggage Door** – The baggage door annunciator switch is part of the forward locking mechanism. When the lock tab is moved to the locked position, the tab moves the contact on the snap-action switch. This action removes the ground on the switch and turns off the door annunciator light when the left and right doors are closed with the ground removed from their switches. To remove or adjust the baggage door annunciator switch, use the following steps. See Figure 52 - 11.

1. **Switch Removal or Adjustment** – The baggage door annunciator switch is mounted to a stud plate that mounts through the switch bracket with two washers and self-locking nuts. To remove the switch, remove the two self-locking nuts and washers, and disconnect the commercial-type connector. To adjust the switch, close and lock the door, and slide the switch toward the lock tab until the switch is activated. Slide an additional 0.020 in. toward the lock tab before torquing the nuts.
2. **Switch Installation** – To install the switch, secure it to the stud plate that mounts through the switch bracket with two washers and self-locking nuts with Loctite. Torque nuts 2 to 3 in.-lbs.

3. **Test the System** – Close and lock the baggage door, and close the lock to the cabin doors while sitting in the cockpit. Turn on the master switch. If the Door Open annunciator light is not illuminated, the switch is properly adjusted. If the annunciator light remains on, adjust the switch as described in paragraph 1 above (moving the switch closer to the lock tab), and retest the system.

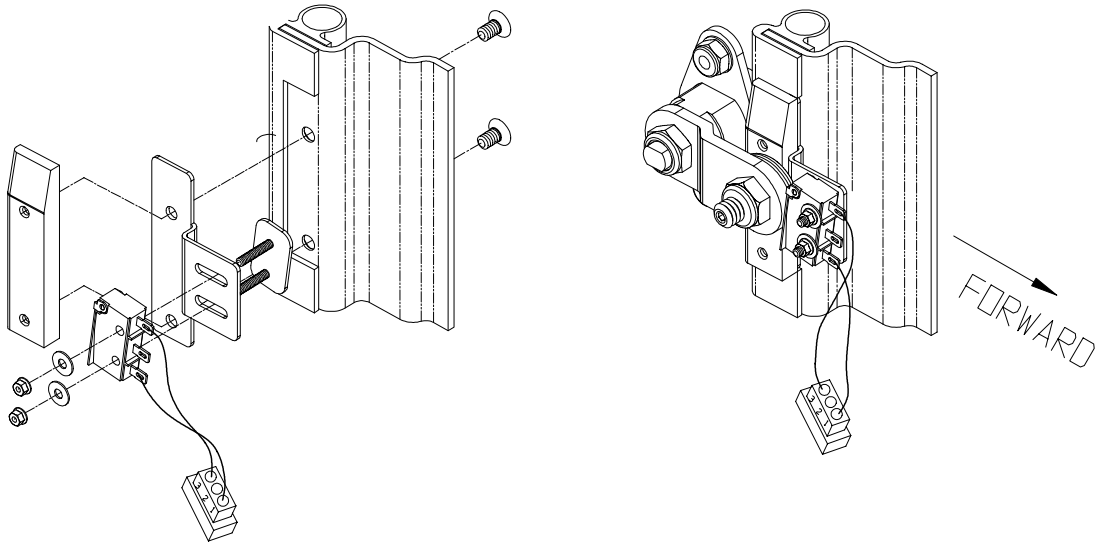

Figure 52 - 11 Baggage Door Switch Assembly

4. **Troubleshooting –** If during flight, the door open annunciator light activates intermittently, the following troubleshooting procedure should be followed.
   a) Ensure that the baggage door hinge is securely fastened to both the door and the fuselage. Check the hinge for excessive wear. Repair or replace the hinge as required.
   b) Check the locks for correct mounting and function. With the key inserted into the lock, the key should only turn approximately 90°. If there is under or over-rotation, replace the lock.
   c) Ensure the lock is properly secured to the door. The lock must be firmly held inside the tapered hole in the door by the locating plate. The protruding studs for attachment of the lock plate to the door should not be bent, causing misalignment of the lock. The lock plate should not be loose. Repair or replace as required.
   d) Ensure that the lock tab is secured to the lock. Confirm that the lock tab is fully seated onto the lock shaft and the shaft nut is tight. When closed and latched, there should be minimal (0.06 maximum when measured at the lock tab) inboard or outboard door movement. Make sure the rain seal is not loose and interfering with door closing. The lock tab may be bent inboard or outboard as required to decrease this door movement.
   e) Inspect the switch lever arm to see if it is bent excessively or broken. Also, examine the lever arm attachment/pivot points for wear or breakage.
   f) Finally, make sure the switch retaining screws are tight enough to prevent switch movement during activation of the switch. Repair or replace as required.

g) If, after performing the above listed preliminary checks along with any required repairs or parts replacements resulting from above inspections, the problem still exists, adjustments to the switch may be required.
h) Loosen the switch retaining screws, and move the switch as far as possible toward the lock. Although there is a very limited amount of movement available for this adjustment, it is usually sufficient to ensure switch activation. Tighten screws 2 to 4 in.-lbs.
i) There is also a very limited amount of adjustment available by loosening the shaft nut on the lock and moving the lock tab as far as possible toward the switch before tightening the shaft nut 36 to 42 in.-lbs.
j) A new, longer lock tab can also be fabricated and used to replace the existing lock tab in extreme cases.
k) Although bending the lever arm may result in proper switch activation, it is not the preferred or recommended method. The lever arm tends to be brittle and break when trying to bend it. In addition, without proper support of the lever arm, the arm pivot points are easily distorted or destroyed when trying to bend the lever arm.

52-12. REMOVAL OF THE CABIN DOOR LATCHING MECHANISMS

a. Removal of the cabin door mechanism is accomplished by using the following instructions. See Figure 52 - 12.
   1. Remove each door handle by removing the cotter pin of the short clevis pin, located through the ears of drum, and then removing the clevis pin. Each handle can be slipped out of the remaining clevis pin in the spindle.
   2. Remove four screws attaching the access panel in the cabin door. Remove the bushing in the access panel before removing the panel. The bushing may be glued to the access panel.
   3. Remove the cotter pin, washer, and clevis pin for each push tube at the connection of the drum ears. The latching pin assemblies can be removed from the door through the access panel.
   4. From the access panel, remove the hex head bolt, washers, nut (located under mounting plate), and spacer.
   5. Remove the truss head screw (located under the exterior handle), washer, nut (located on the access panel side), spacer and the pan head screw. This will allow removal of the remaining mechanism. The mechanism will need to be rotated to get the ears through the slot in the mounting block.
   6. Depressing the spring retainer will allow access to remove the snap ring. Remove the snap ring, spring retainer, spring, and spring housing.
   7. Identify the mating cam, drum, and spindle to the door (left or right) before disassembly. The spindle has two buttons which are opposite of each other and positioned to match the door. The remaining two holes match button locations for the other door.
   8. Remove the cam and spindle from the drum by removing the pin through the drum.
   9. Remove the cotter pins and clevis pins from the spindle.
   10. Some parts are common to both left and right cabin doors; however, it is not good practice to interchange door parts without close examination to size, shape, and length.

52-13. REASSEMBLING THE CABIN DOOR MECHANISMS

a. Use the following steps for reassembling the cabin door mechanism. See Figure 52 - 12. Use light oil or grease on moving parts during assembly, but keep fasteners dry.

1. Assemble the long clevis pins and new cotter pins to the spindle per MS33540.
2. Assemble the mating cam on the drum with the flat face adjacent to the face of the latch pin drive ears. Button positions on the cam must match the door to be assembled (left or right). Buttons may be bonded with Loctite 243 or equivalent.
3. Assemble the mating spindle with the drum, and align the slots of the spindle with the ears of the drum. Insert the pin through the cam, drum, and spindle.
4. Assemble the spring housing, spring, and spring retainer onto the pin. Depress the spring retainer, and assemble the retaining ring with the sharp edge away from the retainer and into the groove of the drum.
5. Add nylon washer to each end of the assembly by assembling through the access panel of the mating door. With the washer end of the drum toward the mounting plate, pass the drum ears through the slot in the mounting plate. Rotate the assembly to position the ears on the exterior of door at the aft location of the door.
6. Attach the cam to the mounting block with spacers, washers, bolt, screw, and nuts. Assemble the screw from the exterior of the door. Torque nut of screw and bolt 30 to 36 in.-lbs.
7. Reassemble the latch arms and pins with new cotter pins per MS33540. Assemble mating latch arms and pins of door through the access panel and through bushings in the door. Attach to the drum ears with the clevis pin and new cotter pins per MS33540.
8. Assemble the left door lock assembly with the mounting plate over the spindle, drive tab (marking shown) on lock, lock tab on drive tab, and nut torqued 36 to 42 in.-lbs.
9. Install pan head screw through door skin and mounting plate. Torque 12 to 15 in.-lbs.
10. Assemble the left door lock assembly to the mounting plate from the access panel with the plate over studs with two nuts and washers torqued 12 to 15 in.-lbs.
11. Assemble the right door lock assembly to the access panel with the thumb latch pointing aft on the panel side with countersink screw holes. Assemble with a star washer, and torque nut 100 to 120 in.-lbs. Align marks on the lock tab, drive tab, and lock assembly, and attach with lock washer and screw. Torque screw 36 to 42 in.-lbs.
12. Assemble the access panel bushing over the ears of the drum. After cycling latches several times with the access panel in place, bond the bushing to the access panel (countersunk screw holes showing) with Loctite 454, and attach the access panel to the door with flat head screws.

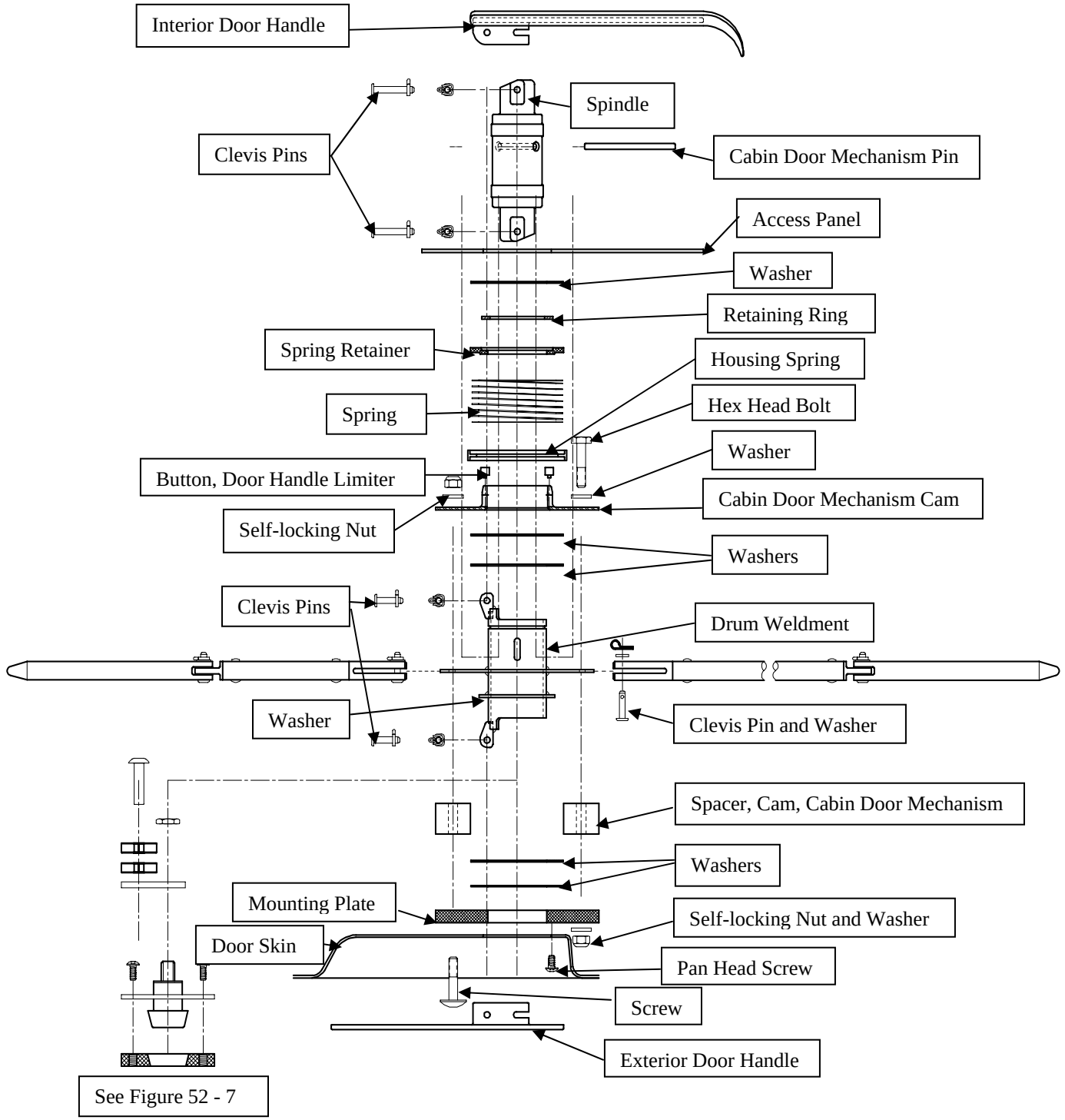

Figure 52 - 12 Cabin Door Locking Mechanism

52-14. REMOTE KEYLESS ENTRY SYSTEM

On 28V aircraft an optional remote keyless entry system allows unlocking of the pilot's door through use of a remote transmitter. The system consists of a transmitter, a remote keyless entry assembly, a receiver, relay, LED located in the left edge of the overhead console, and associated wiring. The system is activated by keying the door lock to "LOCK" and then to "AUTO". An LED illuminates indicating the remote keyless entry system is armed.

a. Remote Keyless Entry Assembly (See Figure 52 - 13)

1. Removal

a) Set the door latching mechanism to the open position, open the door completely, and stand on the skid strips of the left wing. Ensure the door lock is in the unlocked position.

b) Remove the interior door handle by removing the aft cotter pin and clevis. There are two clevis pins in the door handle.

c) When the clevis is removed, slide the handle aft to remove it.

d) Remove the four Phillips screws on the interior door access panel, and remove the panel. A spacer around the door mechanism will normally fall out as the panel is removed.

e) Mark the position of the drive tab on the interior of the door panel. This is a convenient reference when reinstalling the remote keyless entry assembly.

f) Disconnect the electrical connectors.

g) Remove the two flat head screws holding the remote keyless entry assembly in place, and remove the assembly.

h) If necessary the door lock may be removed at this time by removing the nut, drive tab, cam, and locknut holding the lock to the mounting bracket. See Figure 52 - 13.

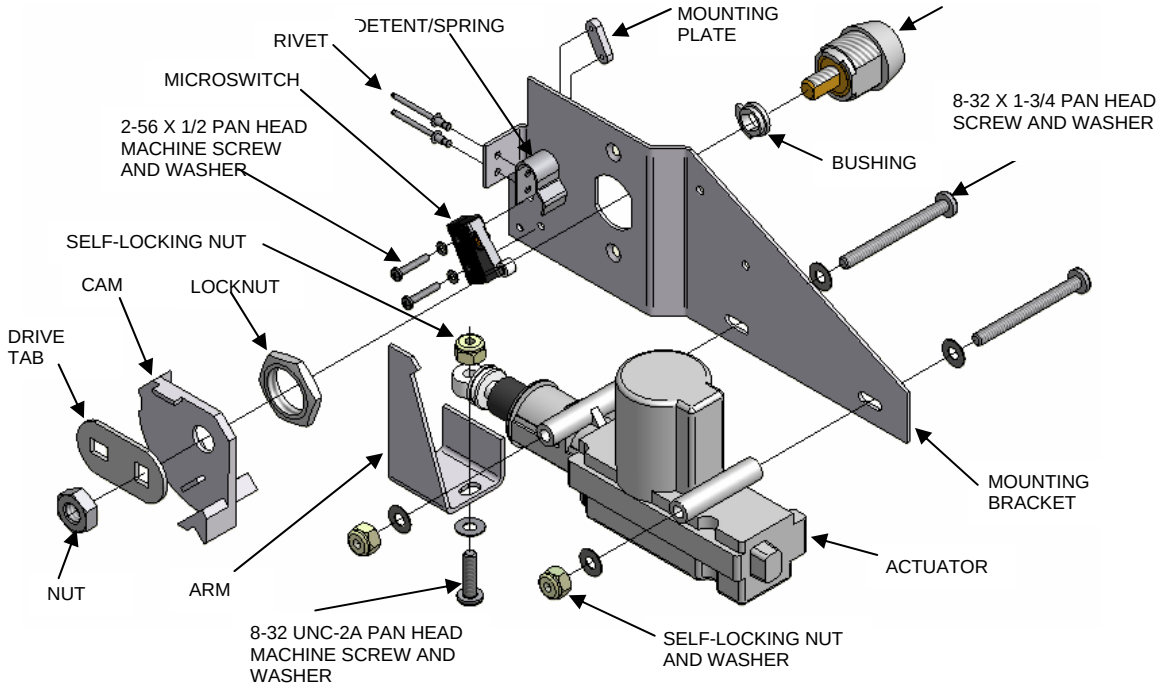

Figure 52 - 13 Remote Keyless Entry Assembly

2. Installation – Installation is the reverse of removal. See paragraph 52-10 for additional instructions.
3. Maintenance – The components of the remote keyless entry assembly require no maintenance.

b. Receiver and Relay
   1. Removal
      a) Remove the overhead console and aft roof panel per Chapter 25.
      b) Disconnect the electrical connectors.
      c) Remove the flat washer, lock washer, and nut (1 ea. for the relay and 2 ea. for the receiver) affixing the respective component to the overhead console or aft roof panel. See Figure 52 - 14 and Figure 52 - 15.
      d) Remove the receiver or relay.
   2. Installation – Installation is the reverse of removal.
   3. Maintenance – The receiver and relay are self-contained components and require no maintenance.

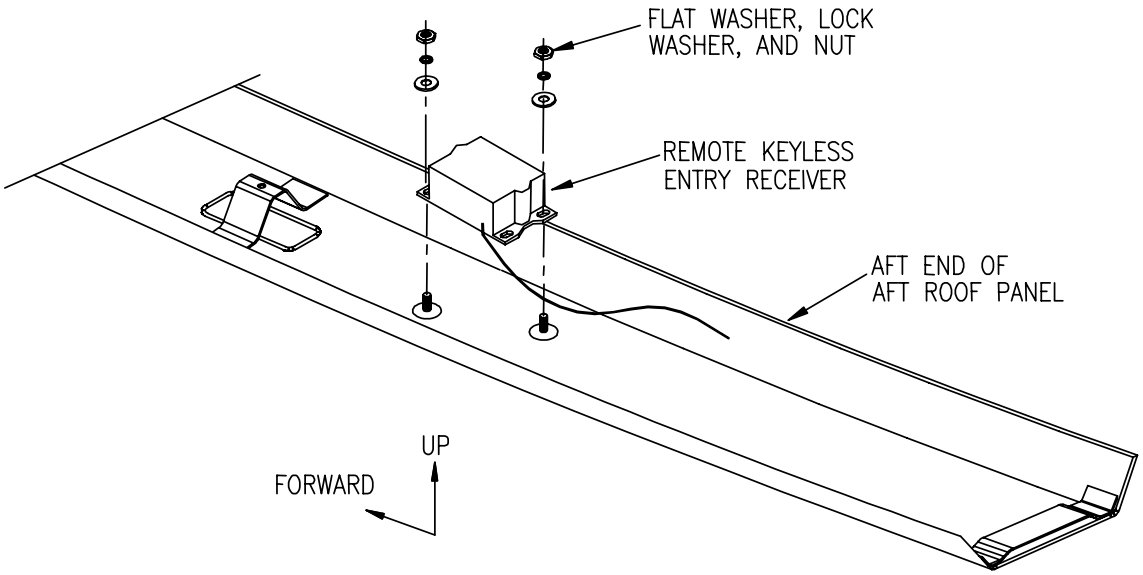

Figure 52 - 14 Remote Keyless Entry Receiver

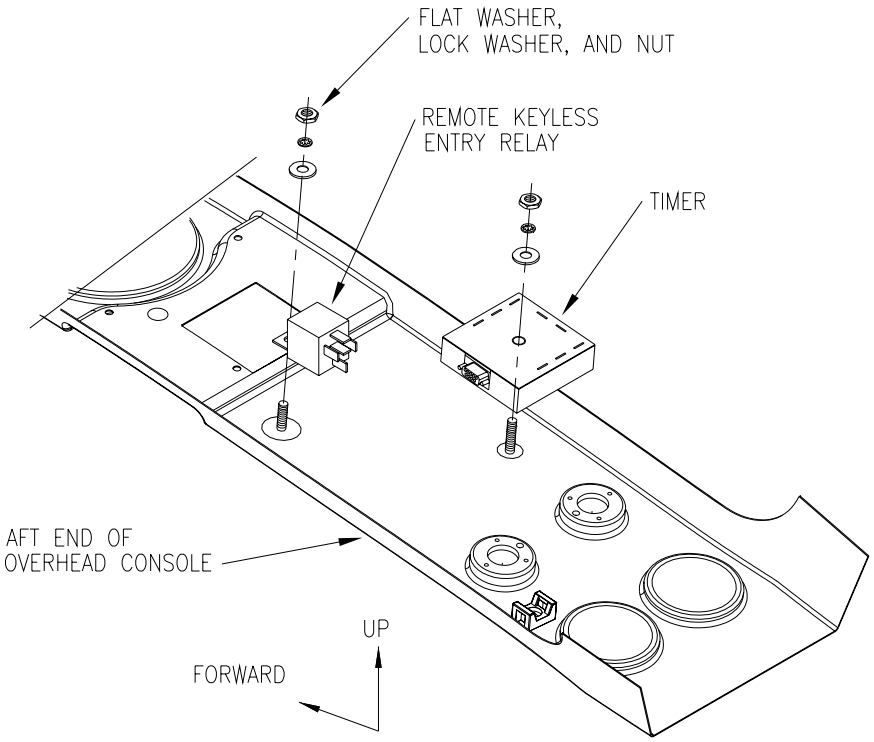

Figure 52 - 15 Remote Keyless Entry Relay

52-15. DOOR SEAL SYSTEM

a. **General** – The door seals are inflated by an electric pump located forward of the pilot's side rudder pedals near the firewall. Two other devices are associated with the electric pump: a pressure switch and a dump valve, which are mounted to the same surface as the pump. The electric pump draws air from the cabin, through a filter, and then pumps it to a pressure switch. From that point the air moves to a T-fitting. One part of the T-fitting supplies pressure to a dump valve, and the other part supplies pressure to the door seals. See Figure 52 - 16 through Figure 52 - 20.

b. **Pressure Switch** – The pressure switch maintains the system pressure at between 12 and 15 psi and turns the pump on and off at these two limits, respectively.

c. **Dump Valve** – The dump valve is a solenoid that requires two conditions to be energized and not dump pressure from the system. The master switch must be on and the Door Open Annunciator must be extinguished. If one of the cabin doors is opened or the master switch is turned off, the solenoid will de-energize and pressure is dumped from the system.

d. **Supply Lines** – The supply lines for the door seal are routed from the pump assembly, behind the lower kick panel, up through the rollover structure, and then aft, above the emergency release cables, to the aft part of the door openings. At this point a T-fitting routes air to the right and left doors. A coil assembly transfers the flow from the cabin to the door.

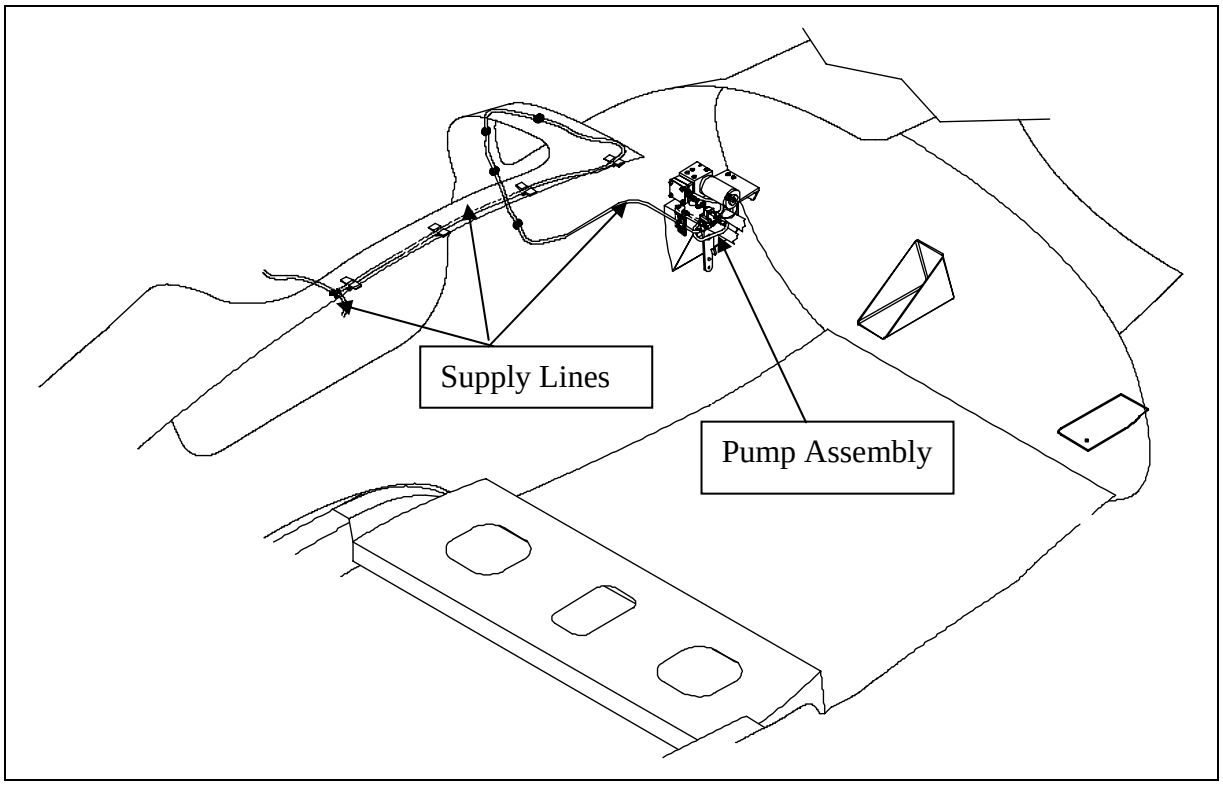

Figure 52 - 16 Location of Door Seal Pump and Lines

e. **Location of Nipples** – There are 14 plastic nipples in the system to which plastic hosing is connected. Since system leaks are frequently at the connection points, the location of each nipple is important.
   1. Seven nipples are located in the pump assembly.
   2. Three nipples are located in the overhead panel.
   3. Two nipples are associated with each door, one at the cabin connection and one at the door connection.

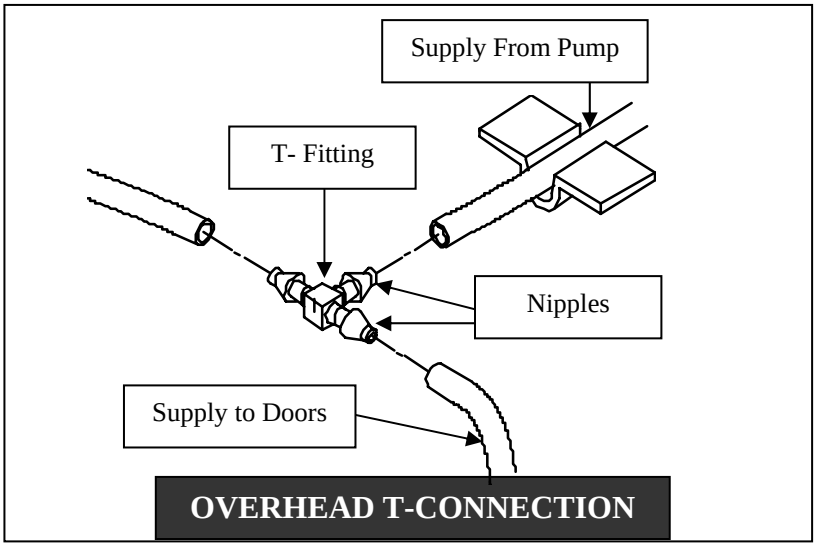

Figure 52 - 17 Door Seal T-Connector

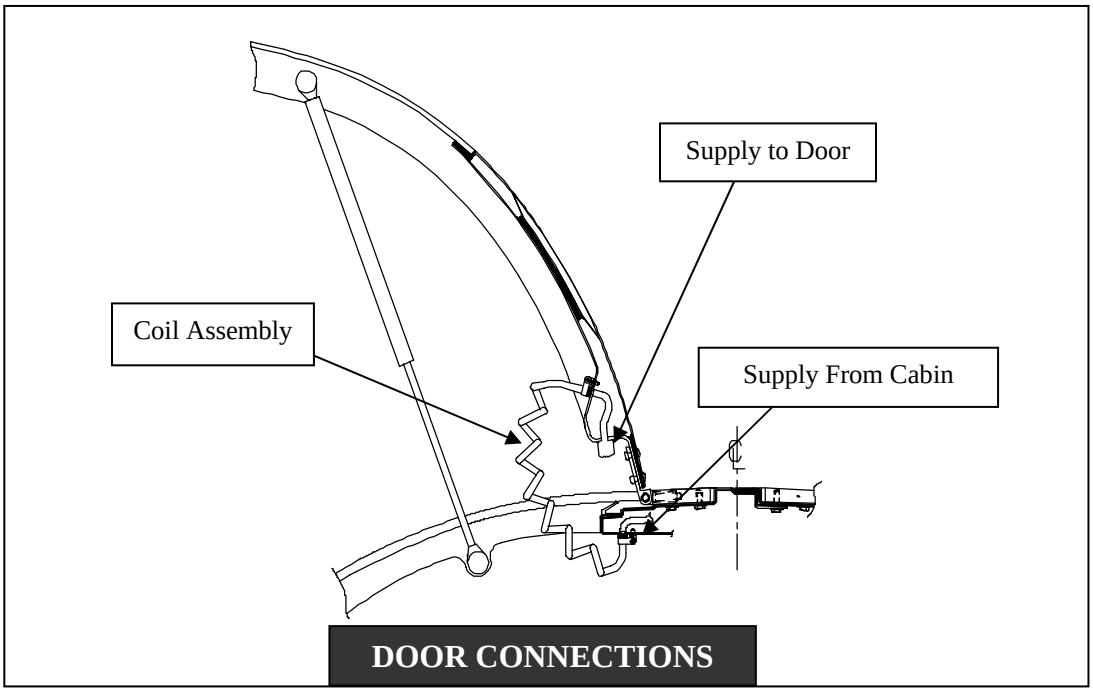

Figure 52 - 18 Door Seal Door Connections

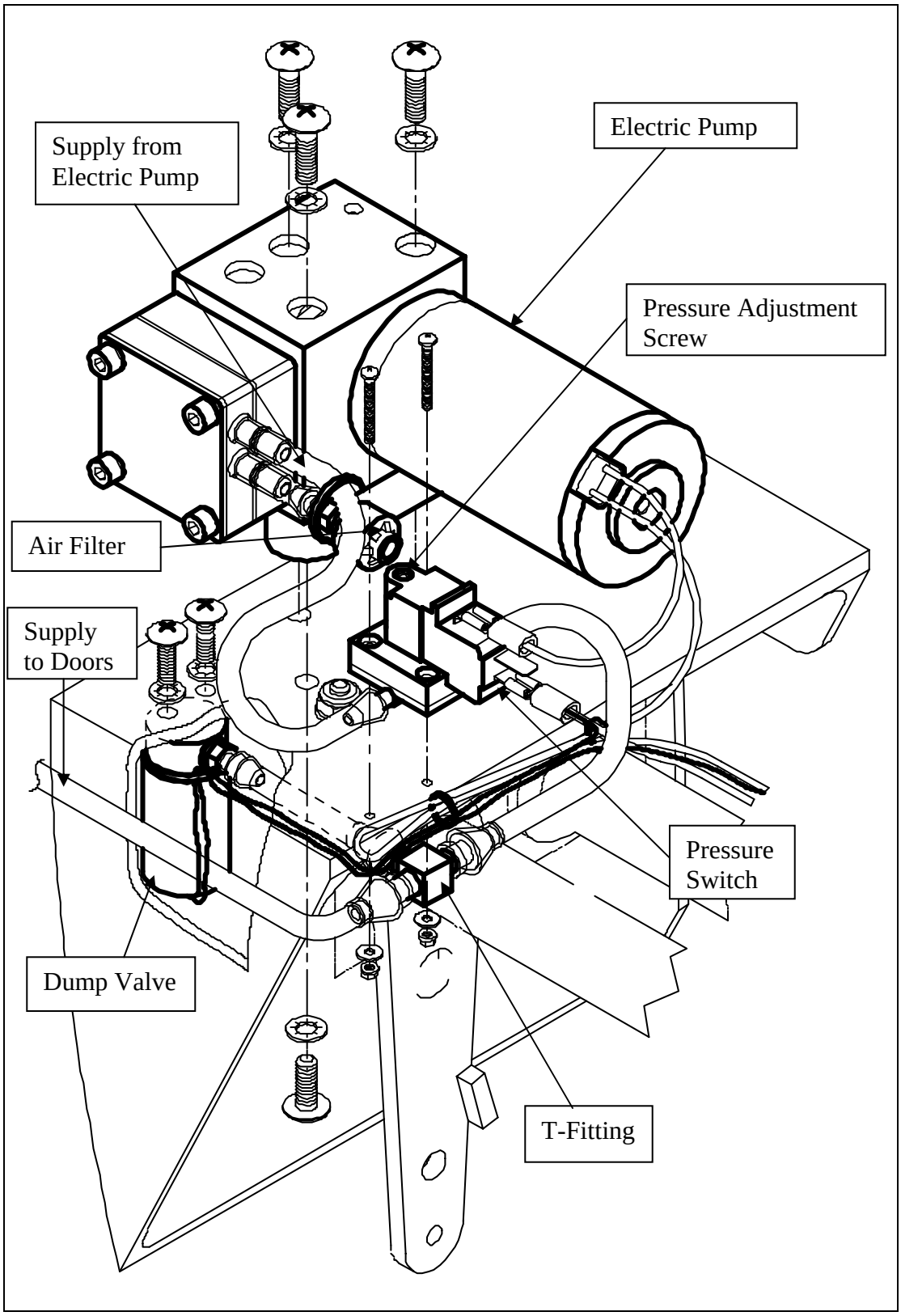

Figure 52 - 19 Door Seal Pump Assembly (Pump Type A)

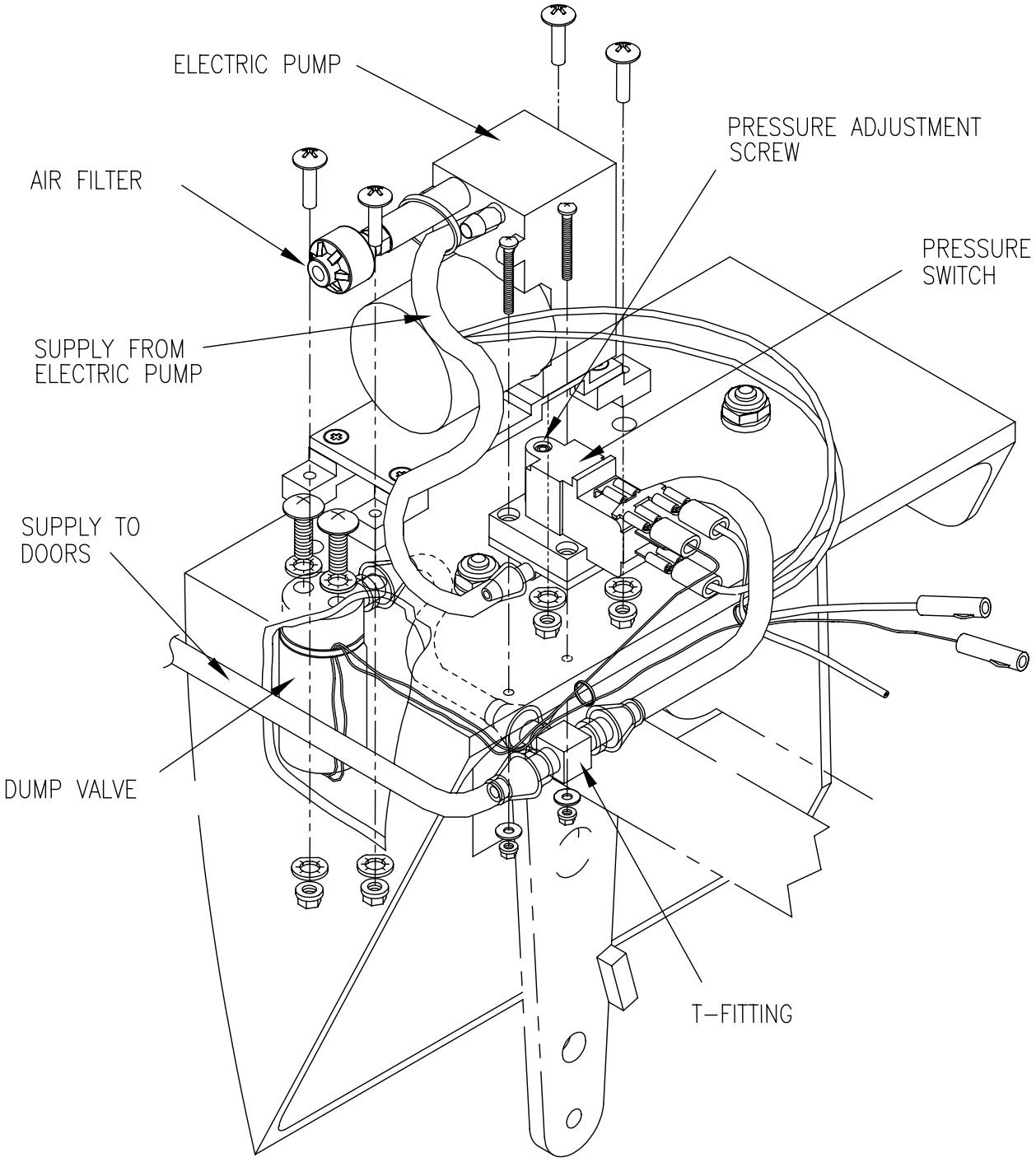

Figure 52 - 20 Door Seal Pump Assembly

(Pump Type B)

52-16. TROUBLESHOOTING THE DOOR SEAL SYSTEM

a. **Symptom** – Door seal system cycles on and off every few seconds.
   1. **Probable Cause** – Supply line between the pump and pressure switch has a leak.
   2. **Action** – Replace the supply hose, and recheck the system.

b. **Symptom** – Pump runs continuously or turns on and off every minute or so.
   1. **Probable Cause** – A leak somewhere in the system
   2. **Action** – Isolate the area of the leak.
      a) Use a hemostat to isolate a leak in the system. First, clip off the system at a point between the output of the pump assembly and the supply to the door seals. It will be necessary to remove the left side kick panel (See Chapter 25) or the right FOSP. See 52-3.a.2. Operate the system with the hemostat in place. If it holds pressure for more than two minutes, the problem is in one of the following areas:
         1) The overhead T-fitting supply lines
         2) The supply lines to the doors
         3) The supply line from the pump to the overhead T-Fitting, somewhere forward of the hemostat.
      b) If the system leaks when the hemostat is applied between the output of the pump and the supply to the system, the leak is probably in one of the following areas:
         1) The supply line between the hemostat and the pump assembly
         2) The supply line between the pump assembly T-fitting and the dump valve
         3) The supply line between the pressure switch and the pump assembly T-fitting
      c) Depending on which side of the system the leak is first isolated, continue to move the hemostat in the appropriate direction until the exact location is determined, and replace the supply line/Clippard hose per instructions in paragraph 52-17. The following points should be remembered:
         1) There may be more than one leak in the system.
         2) To access the pump assembly, it works best to remove the engine instrument panel.

c. **Symptom** – Door Seal System will not turn on.
   1. **Probable Cause** – The most probable causes are listed below in their most likely order:
      a) Door latch plunger-type switches need adjustment.
      b) Doors are not completely closed.
      c) Circuit breaker is popped.
      d) Pressure switch is defective.
      e) Pump is defective.
   2. **Action** – Perform the actions below in the order listed.
      a) Ensure the Door Open annunciator is operative so that when the door latching mechanism is activated the Door Open light turns off.
      b) Ensure that the door seal circuit breaker is in.
      c) If the doors are functioning properly, jumper past the pressure switch. If the pump turns on, the problem is a defective pressure switch. Replace switch per instructions in paragraph 52-18.
      d) If jumpering past the pressure switch does not turn on the pump, the pump is defective. Replace the pump per instructions in paragraph 52-18.

d. **Symptom** – There are no leaks in the system, however, the pump runs but will not adequately inflate the door seals.
   1. **Probable Cause** – The pump filter is clogged or the dump valve is defective.
   2. **Action** – Clean the filter. If this does not cure the problem, place a hemostat on the line between the pump assembly T-fitting and the dump valve. If the system pressurizes the door seals, the problem is a defective dump valve. Replace the dump valve per instructions in paragraph 52-18.

52-17. REPLACING AIR SUPPLY LINES/CLIPPARD HOSE

a. If a supply line leak is isolated, it must be replaced with Clippard hose Part No. 3814-6.
b. The fit of the hose to the supply nipple is very tight, and in most cases, it will require cutting the hose off. The hose can be heated slightly with a heat gun, however, extreme care must be used to not damage the nipple or other components.
c. Ensure the new hose is routed through the same plastic guides as the old hose. Normally, if the hose is inserted past the first barb on the nipple, this is sufficient.
d. To replace the door coil, use the following steps:
   1. Remove the Adel clamps from the cabin and the door. See Figure 52 - 4.
   2. Gently pull on the door coil so that it draws the nipple and Clippard hose from the overhead cabin hole and the hole in the door.
   3. Remove the door coil from each nipple, and install the new one.
   4. Feed the Clippard hose back into the overhead panel and door, and reinstall the Adel clamps.

52-18. REPLACING PUMP ASSEMBLY COMPONENTS

a. **Dump Valve** – Removal and installation of the dump valve requires removal and installation of the two No. 6 screws, which are inserted into threaded holes in the dump valve. See Figure 52 - 19 or Figure 52 - 20. Ensure the Clippard hosing and the electrical connection are removed. Torque screws 10 to 12 in.-lbs.
b. **Pressure Switch** – The pressure switch is removed and installed through removal and installation of the four No. 2 screws with self-locking nuts and washers. The electrical connections and the Clippard hosing must be removed. See Figure 52 - 19 or Figure 52 - 20. To **adjust the switch pressure**, T-in a calibrated pressure reading device to the system, and turn the pressure adjustment screw until the output pressure is 14 psi (pump type A) or 10 psi (pump type B). Torque 2 to 3 in.-lbs.
c. **Pump** – The pump is removed and installed by removal/installation of the three top screws and washers (pump type A) which are attached to a threaded rubber isolator, or the four screws, nuts, and washers (pump type B) attaching the pump to the flange. See Figure 52 - 19 or Figure 52 - 20. Ensure that all electrical connections are removed and that the Clippard hosing is removed. Torque screws 20 to 25 in.-lbs.
d. **Nipples** – Nipples for the dump valve and the air filter are installed at the factory. If it is necessary to replace the nipples, they should be hand tightened plus 1/4 of a turn.

CHAPTER 53

FUSELAGE

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Chapter 53

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53-1. General

This chapter will provide information on the fuselage structure and attach items for the Cessna 350 (LC42-550FG) aircraft. The fuselage of the Cessna 350 is a composite semi-monocoque design with an integral vertical stabilizer structure. Doors and access panels allow access to the interior areas for inspection and maintenance. Most of the structural components are permanently bonded together.

The fuselage structure carries loads from the engine mount, wing, horizontal, and main landing gear. This structure consists of the fuselage skins, vertical shear web, vertical shear closeout, antenna bulkhead, baggage bulkhead, rear floor, footwell, forward floor, and forward floor under-floor bulkhead. Hardware hardpoints exist for the seatbelts, engine mount, and instrument panel. Aileron, elevator, and rudder controls also run in the fuselage, and support structure and hardpoints allow mounting provisions for these components.

The fuselage is primarily constructed from glass fabric composites with honeycomb cores. Carbon fiber fabric is used on the engine mount attachment gussets. Uni-carbon fiber is used in the fuselage longerons and rollover structure. The fuselage longerons are located in two strips along the fuselage sides above the wing saddle. The rollover stiffeners are located in the “A” and “B” pillars on the fore and aft sides of the doors. The bulkheads and floors are permanently bonded to the fuselage shells with specially thickened epoxy adhesive and an epoxy primer. Moisture drain holes are located in the bottom of the fuselage along the centerline.

CAUTION

Only experienced personnel should attempt the repair work described in this manual. The repair of composite materials is not difficult but requires certain hands-on skills where experience is necessary. These composite repairs will be time limited due to resin gel time and must be completed in a timely fashion.

WARNING

To prevent personal injury, refer and adhere to instructions contained in the appropriate Material Safety Data Sheet (MSDS) accompanying the material you are using for any repair.

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53-2. IDENTIFICATION AND INSPECTION OF FUSELAGE COMPONENTS

a. **Inspection of Fuselage Primary Structure** – Damage to the composite fuselage structure can usually be identified visually. Damaged areas will show a characteristic white or milky color. Undamaged areas will be translucent brown to green. The use of tap testing and other non-destructive methods may be required.

Carbon fiber damage is more difficult to identify. The use of tap testing and other non-destructive methods will be required.

Tap testing and other non-destructive methods can identify damage to the bonded joints in the fuselage. Reference Chapter 20 and 51 for specific inspection techniques.

53-3. PAINTING

Painting of the fuselage should follow the requirements in Chapter 51.

53-4. LIGHTNING STRIKE PROTECTION

Lightning Strike Protection must comply with the requirements of Chapters 20 and 51.

53-5. FUSELAGE PRIMARY STRUCTURE DESCRIPTION

a. **Firewall** – The fuselage firewall is a structural composite member. (See Figure 53 - 1.) The firewall features co-cured carbon fiber upper engine mount gussets and lower carbon fiber engine mount gussets, together with the forward floor, that are bonded to the firewall. On the engine side, the firewall is covered with a stainless steel firewall plate with fiberfrax and phenolic insulators and phenolic hard points. The firewall plate is sealed around the edge with silicone rubber. The hardware attached to the firewall holds the stainless steel plate and phenolic fittings in place. The engine mount bolts to the firewall. Engine controls, fuel lines, electrical wiring, and heating and ventilation air pass through the firewall. The cockpit side of the firewall is covered by sound barrier foam.

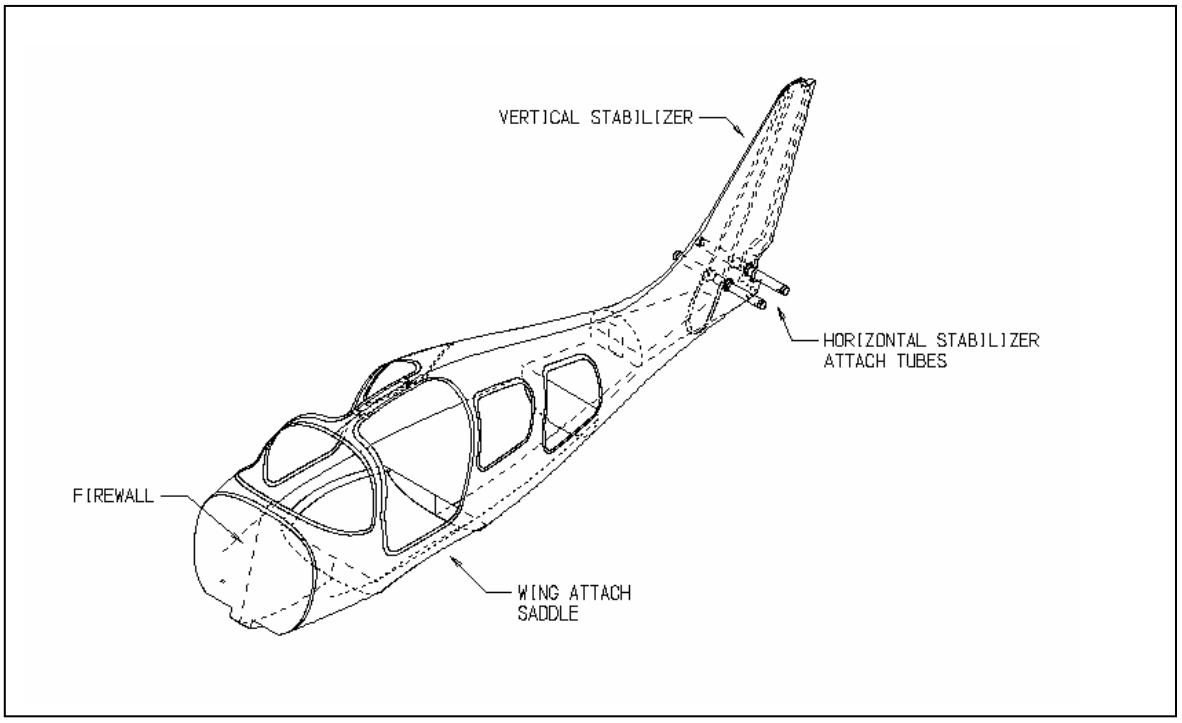

Figure 53 - 1 Fuselage Structure – External Features

b. **Forward Cabin Floor** – The forward cabin floor extends from the firewall to the footwell. The forward cabin floor is bonded to the fuselage skins, firewall, and wing mount saddle. Sound insulation is bonded to the bottom side of this floor.

c. **Forward Cabin Floor Support** – The forward cabin floor support is located under the forward floor and provides stiffening for the forward floor and instrument panel radio rack assembly. The support is bonded to the forward cabin floor and fuselage skins. There is no access to this structure.

d. **Cabin Footwell** – The footwell is a structural member that is attached to the fuselage floors by screws. The center attachment bracket for the front seat belts is bolted to the footwell.

e. **Aft Cabin Floor and Aft Baggage Floor** – The aft cabin and aft baggage floors are bonded together, and the joint is reinforced with glass cloth. These floors cover the main gearbox. The main gearbox can only be removed from the aircraft by cutting away these floor bulkheads.

NOTE

Removal of these floors can only be done with factory approval and specific instructions. These floors are bonded to the fuselage sides, wing attach saddle, and aft baggage bulkhead.

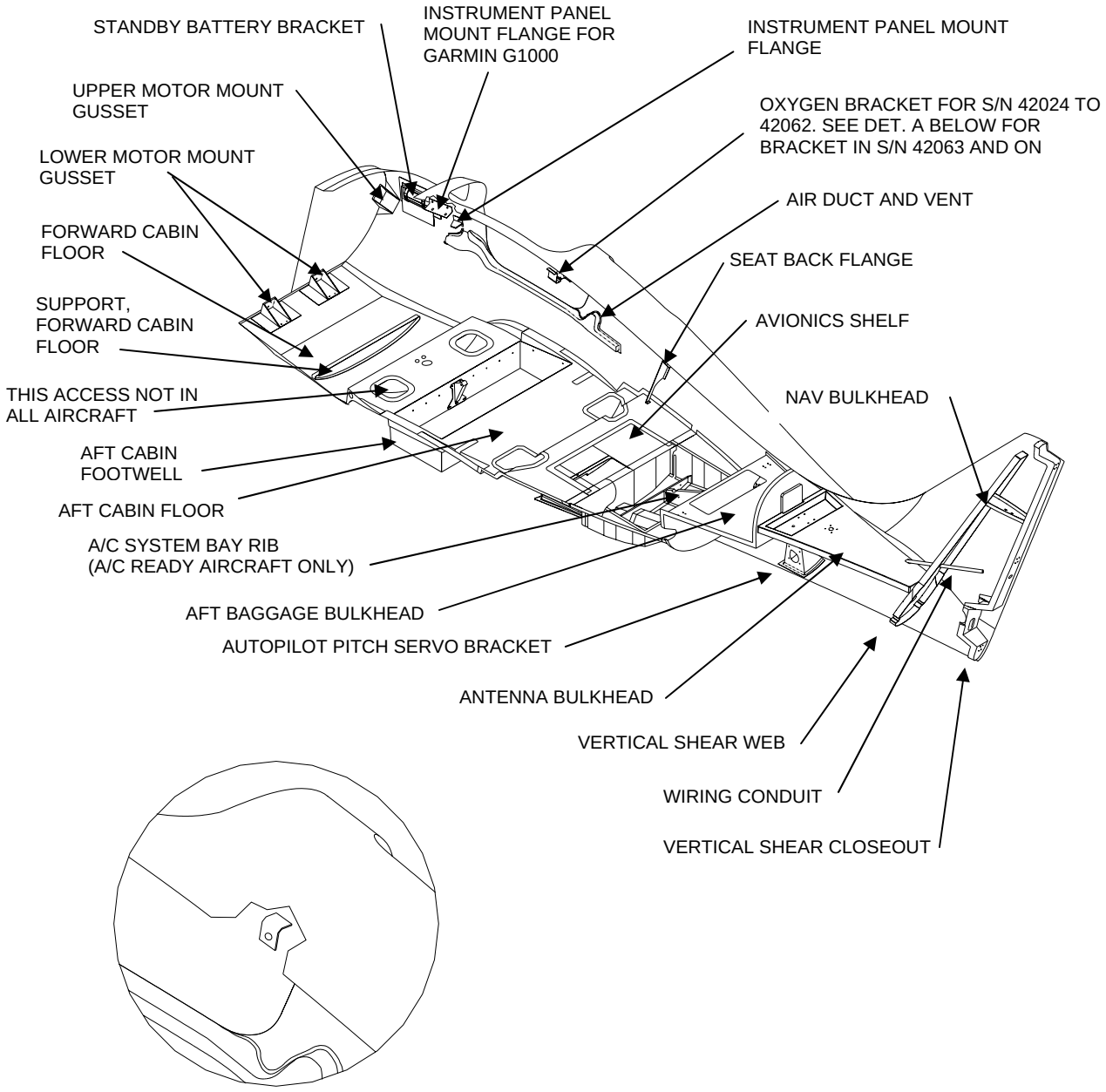

Figure 53 - 2 Fuselage Structure – Internal Features

f. **Avionics Shelf Frame** – The avionics shelf frame supports the avionics shelf. This shelf is designed to hold a multitude of components and hardware. The shelf frame is bonded to the fuselage skins.

g. **Aft Baggage Bulkhead** – The Z-shaped aft baggage bulkhead carries the elevator crossover torque tube. It also serves as the hat rack baggage floor. The bulkhead features two access panels, the baggage bulkhead and antenna bulkhead panels. The bulkhead is bonded to the fuselage sides, aft baggage floor, and antenna bulkhead.

h. **Antenna Bulkhead** – The antenna bulkhead is a structural member that provides lateral and torsional shear carry-through. It is bonded to the aft baggage bulkhead, fuselage sides, and vertical shear web.

i. **Vertical Shear Web** – The vertical shear web is the main structural shear web in the vertical stabilizer. It is bonded to the fuselage sides, which form the vertical stabilizer, antenna bulkhead, and NAV bulkhead.

j. **Vertical Shear Closeout** – The vertical shear closeout is a shear web and provides hardpoints for the rudder hinges and elevator up stop. This shear web is bonded to the fuselage sides and NAV bulkhead.

k. **NAV Bulkhead** – The NAV bulkhead provides a mount for the NAV antenna. It spans the NAV antenna access panel opening and ties into the vertical shear webs.

l. **Seat Back Flange** – The seat back flange is a bonded support for the rear seat backs. It is bonded to the fuselage skins and aft cabin floor.

m. **Instrument Panel Mounting Brackets (S/N 42001 to 42500)** – The instrument panel is mounted to three bonded-on composite brackets and two threaded inserts. The brackets are located on the top center and the left and right sides just forward and below the bottom of the “A” pillar. The threaded inserts are located on the left and right sides, forward near the firewall.

n. **Instrument Panel Mounting Brackets (S/N 42501 and on)** – The instrument panel is mounted to two bonded-on composite brackets. On the left and right sides of the fuselage.

o. **Oxygen Bracket** – For S/N 42024 to 42062, the cabin oxygen distribution manifold is mounted to a composite bracket bonded to the fuselage behind the forward center overhead light panel (FCOLP). For S/N 42063 and on, the cabin oxygen distribution manifold is mounted directly to the FCOLP. An oxygen quick disconnect bracket behind the right aft overhead side panel (AOSPR) replaces the oxygen bracket behind the FCOLP.

53-6. FUSELAGE ACCESS PANELS

The fuselage has numerous access panels located on the exterior and interior of the fuselage as shown in Figure 53 - 3. This paragraph describes the panels and how to remove and replace them.

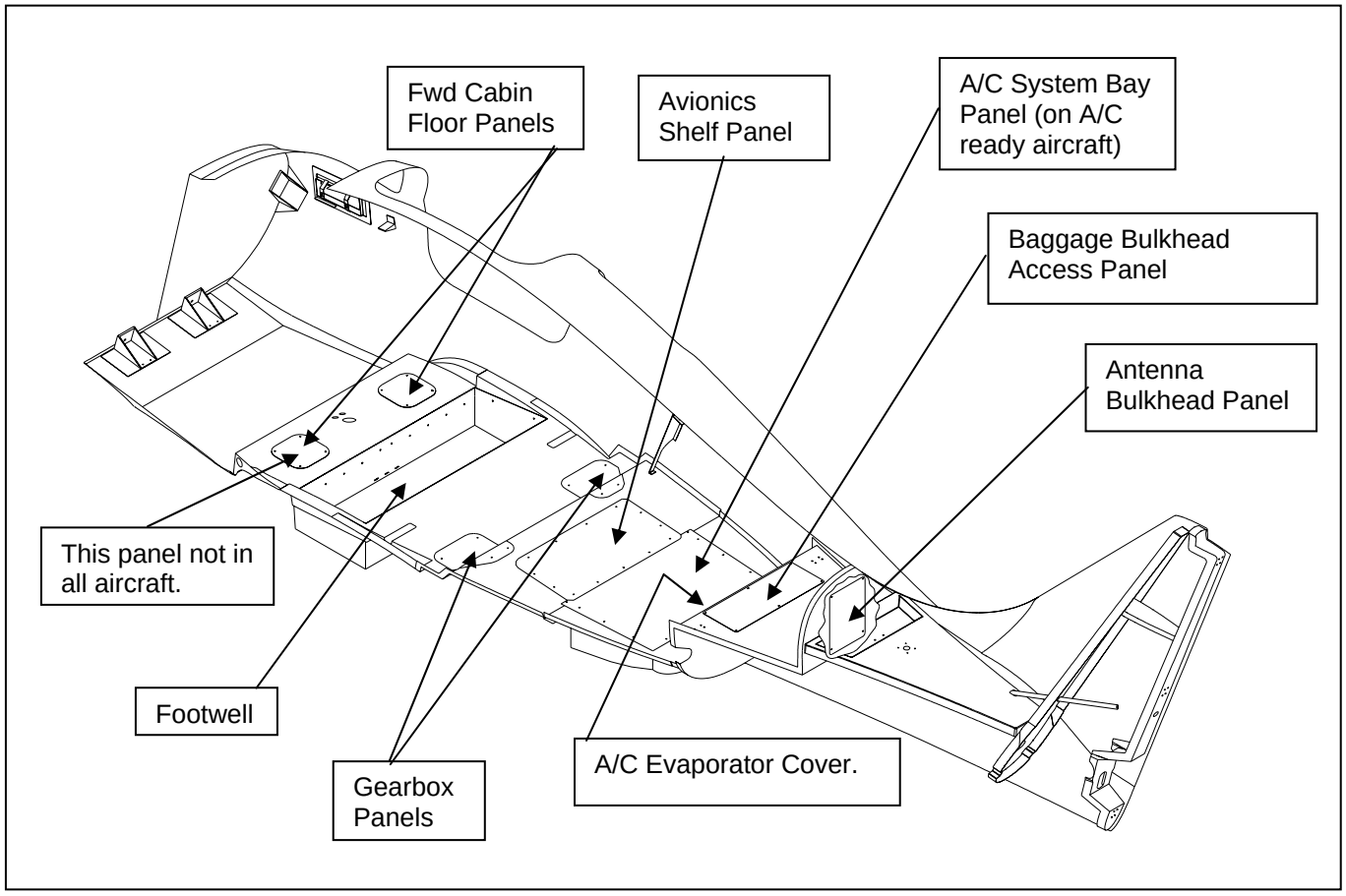

Figure 53 - 3 Fuselage Access Panels

a. **NAV Antenna Access Cover** – The NAV antenna access cover is located on the left side of the vertical stabilizer about mid span and provides access to the NAV antenna. This panel is removed by unscrewing the attachment screws. Installation is a direct reversal of the removal process.

b. **Air Outlet Vent Access Panel** – The air outlet vent access panel is located under the horizontal stabilizer on the right hand side of the aircraft. It can be used to inspect the fuselage between the vertical stabilizer shear webs. The aft-most end of the elevator push rod can also be seen from this location. The panel is removed by unscrewing the attachment screws. Installation is a direct reversal of the removal process.

c. **Antenna Bulkhead Access Panel** – The antenna bulkhead access panel is located on the baggage bulkhead. It provides access to the antennas mounted on the shelf. Access to both the top and bottom of this shelf is accomplished through the same access panel. The panel is removed by unscrewing the attachment screws. Installation is a direct reversal of the removal process.

d. **Baggage Bulkhead Access Panel** – The baggage bulkhead access panel is located in the rear of the baggage area. It can be used to access the elevator crossover and its three elevator pushrod fittings, the Comm antenna, and the autopilot pitch servo. The panel is removed by unscrewing the attachment screws. Installation is a direct reversal of the removal process.

e. **Air Conditioning (A/C) System Bay Access Panel** – This panel is in aircraft that are air conditioning ready. The panel allows access to the a/c system and the a/c condenser.

f. **Air Conditioning (A/C) Evaporator Cover** – This cover is in aircraft that are equipped with air conditioning and encloses the evaporator.

g. **Avionics Shelf Access Panel** – The avionics shelf access panel is located in the middle of the forward baggage floor. It is used to access the avionics shelf, Comm antenna, step bolts, and rear gearbox attachment bolts. The panel is removed by unscrewing the attachment screws. Installation is a direct reversal of the removal process.

h. **Main Gearbox Access Panels** – Left and right side main gearbox access panels are located on the rear seat floor. These panels can be used to access the gearbox and forward attachment bolts for inspection. These panels are removed by unscrewing the attachment screws. Installation is a direct reversal of the removal process.

i. **Footwell** – The footwell can be removed to provide access to the wing attach bolts. This is a structural member of the fuselage and contains the front seat belt attach bracket. It is installed with AN526C1032R10 screws torqued 30 to 36 in.-lbs. with NAS1149F0363P washers (AN960-10). The length of these screws shall be appropriate for the attachment. See Chapter 25 for removal and installation procedures.

j. **Forward Cabin Floor Access Panel** – The forward cabin floor access panels are located under the pilot and co-pilot's seat. The panels provide access for the front seat rail mounting. They also allow inspection of the fuel system components mounted on the front wing spar from above. The panels are removed by unscrewing the attachment screws. Installation is a direct reversal of the removal process. On some aircraft the panel under the pilot's seat is absent.

53-7. AERODYNAMIC FAIRINGS

Aerodynamic fairings attached to the fuselage consist of left and right wing fillets.

a. **Wing Fillets** – The wing fillets are bonded to the fuselage skins outside surface. They form an aerodynamic transition between the fuselage and the wing. The fillets are not bonded to the wing. The main gear leg fuselage transition panel is located at the junction of the main gear legs and the fuselage. They extend out to the root of the flaps and butt up to the wing belly at the forward edge. The fillets have a small panel for the ingress and egress steps. These steps are optional. If the steps are not present, this panel will not serve any function. The step panel allows access to the step attachment bolts. The wing fillets should be inspected periodically for cracks and impact damage.

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53-8. CONTROL SYSTEM SUPPORTS

This section describes the location of control system supports in the fuselage as seen in Figure 53 – 4.

a. **Rudder Pedal Crossover Tube Support Brackets** – Three bonded composite brackets support the rudder pedal control system. The brackets are located on the left and right fuselage sides under the instrument panel and at the firewall centerline. The left bracket provides mounting for the rudder limiter system.

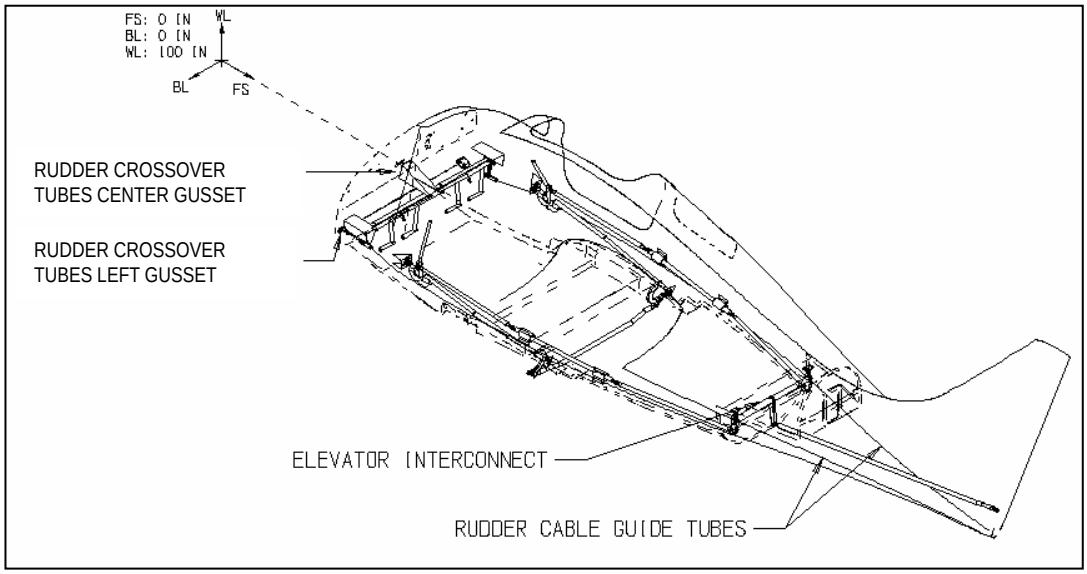

Figure 53 – 4 Control System Support

b. **Aileron Torque Tube Gussets** – Two gussets located at the forward and aft ends of each aileron torque tube support the stick assembly. The spherical bearing cups for the aileron torque tubes are bolted to these gussets.

c. **Rudder Cable Guide Tubes** – Rudder control cable guide tubes are co-cured with the fuselage skins. These guide tubes normally hold a control cable sleeve secured with hose clamps on either end. The rudder control cables run fore and aft in these sleeves. Replacement of the sleeves may only be performed with replacement of the cables.

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53-9. REMOVAL, INSTALLATION, AND MAINTENANCE PRACTICES FOR FUSELAGE COMPONENTS

This section describes the basic inspection, removal, and installation of components attached to the fuselage.

a. **Horizontal Stabilizer Attach Tubes** – The horizontal stabilizer mounting is accomplished with two steel tubes permanently attached to the fuselage structure.

1. The horizontal tubes should be securely bonded to the fuselage. No free play should be evident. Any cracks between the tubes and the fuselage structure will be cause to contact the factory for repair instructions.
2. These tubes should be visually inspected for corrosion and nicks.
3. Surface corrosion and shallow nicks should be removed with careful application of fine grit sanding paper or crocus cloth. The surface finish of the reworked area must be 63 micro inches per inch or better. This process will remove the protective plating from the tubes. These areas should be protected with zinc chromate primer. Care must be taken with the application of any protective coatings as they may prevent the reinstallation of the horizontal stabilizers.
4. Deep nicks and corrosion will require factory approval for repair. Replacement of the horizontal attach tubes will also require factory approval.

b. **Fuselage Seat Belt Attach Hardpoints** – Each shoulder belt inertial reel is mounted to the fuselage. The forward inertial reels mount to steel brackets on the “B” pillar. The rear seat shoulder belt inertial reels are tang mounted to a through bolt in the fuselage skin. (See Chapter 25 for details on removal and replacement of seat belts.)

c. **Cabin Door Attach Hardpoints** – The cabin doors mount with two hinges and are secured with two pins. The door hinges are mounted on the top of the door. The hardpoints provide a structural mounting for the metallic door hinges. Emergency release cable guides are bonded to the fuselage and run from the belly to the door hinges. Each door is secured front and aft by latch pins. The latch pins interface with the fuselage hardpoints in the fuselage door flanges. See Chapter 52 for details on inspection, removal, and replacement of the cabin door and components.

d. **Leading Edge Tape** – Optional leading edge anti-chafe tape may be added to the aircraft. The anti-chafe tape is 1 in. wide, adhesive-backed, clear, urethane tape that prevents pitting and wear on the leading edges of the vertical and horizontal stabilizer.

To apply the tape, thoroughly clean the leading edges using isopropyl alcohol. The 1 in. wide tape should be centered along the leading edge of the surface. The tape should end at the beginning of the vertical and horizontal stabilizer tip curve. Use a rubber scraper/squeegee as required to smooth the tape and remove air bubbles from underneath the tape. To remove the tape, slowly pull the tape back on itself, and clean the surface with isopropyl alcohol.

The anti-chafe tape should be inspected at each 100 hr. or annual inspection to ensure that it is not peeling up and is properly attached.

e. **Fuselage Handles –** Optional fuselage handles are available with certain aircraft to assist entering the aircraft. The handles are located behind the passenger windows. Certain aircraft may be retrofitted with handles. Contact the manufacturer to determine if the aircraft may be retrofitted and for instructions. The handles may not be permanently removed without getting additional instructions from the manufacturer.

1. **Temporary Removal of Handle**
   a) Remove the portion of the headliner on the inside of the aircraft to gain access to the handle screws.
   b) Remove the two 10-32 screws and washers, and remove the handle.
2. **Handle Installation or Tightening of Loose Handle**
   a) Apply 262 threadlocker to the two 10-32 screws, and install the washer and screw. Torque 20 to 25 in.-lbs.
   b) Re-install the headliner using Keystone ADH105P adhesive as necessary.

f. **Fuselage Steps –** Optional fuselage steps are available with certain aircraft to assist entering the aircraft. The steps are located behind and below the trailing edge of the wing. Certain aircraft may be retrofitted with the steps. Contact the manufacturer to determine if the aircraft may be retrofitted and for instructions. The steps may not be permanently removed without getting additional instructions from the manufacturer.

1. **Temporary Removal of Step**
   a) Remove the exterior step access panel by removing the four countersunk screws.
   b) Remove the interior aft cabin floor access panel per instructions in Chapter 25 to gain access to the nuts, or nutplates, securing the steps.
   c) For steps secured with flanged nuts
      1) Remove the two lower hex-head bolts, flanged nuts, and washers, and mark the hardware so that it can be put back in the same lower holes.
      2) Remove the two upper hex-head bolts, flanged nuts, and washers, and mark the hardware so that it can be put back in the same upper holes.
   d) For steps secured with nutplates
      1) Remove the two lower hex-head bolts and washers, and mark the hardware so that it can be put back in the same lower holes.
      2) Remove the two upper socket head cap screws, washers, and nutplates, and mark the hardware so that it can be put back in the same upper holes.
   e) Remove the step and any shims that may be present.
2. **Step Installation**
   a) Position step against fuselage and align using shims.
   b) For steps secured with flanged nuts
      1) Install the two upper hex-head bolts, flanged nuts, and washers. Tighten snug plus ¼ turn.
      2) Install the two lower hex-head bolts, flanged nuts, and washers. Tighten snug plus ¼ turn.
   c) For steps secured with nutplates
      1) Install the two upper socket head cap screws, washers, and nutplates. Hand tighten loosely.
      2) Align the nutplates and install the two lower hex-head bolts and washers.
      3) Torque the two upper socket head cap screws 72 to 84 in.-lbs.
      4) Torque the two lower hex-head bolts 30 to 36 in.-lbs.
   d) Replace the exterior step access panel by installing the four countersunk screws. Torque 12 to 15 in.-lbs.
   e) Replace interior aft cabin floor access panel per instructions in Chapter 25.

CHAPTER 55

STABILIZERS

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55-1. GENERAL – SCOPE AND DEFINITION

The Cessna 350 (LC42-550FG) empennage assembly consists of a horizontal and vertical stabilizer. The vertical stabilizer is an integral part of the fuselage. The horizontal stabilizer is a two-piece unit that is fully removable for service and maintenance.

The horizontal stabilizer is of a carbon fiber composite construction. The internal structure consists of two ribs that run the full length of the stabilizer chord line at their respective stations and two shear webs: one between the two ribs and one that is outboard of the outboard rib. The horizontal stabilizer is attached to the vertical stabilizer via two high-strength steel tubes and two locking pins.

The elevator and rudder are also of a carbon fiber composite construction. The elevator has an electrically actuated trim tab located on the right side. The elevator is attached to the horizontal stabilizer with two piano hinges. The rudder has a fixed trim tab located mid-span along the trailing edge and is attached to the vertical closeout at three locations. Both the elevator and rudder have balance weights installed for stabilization during flight.

55-2. Horizontal Stabilizer

The left and right sides of the horizontal stabilizer structure consist of an inboard rib, an outboard rib, an inboard shear web, and an outboard shear web, as shown in Figure 55 - 1. The ribs and shear webs are a pre-molded carbon composite material. The inboard and outboard ribs contain bushings for the horizontal tubes. The bushings are preinstalled in the ribs.

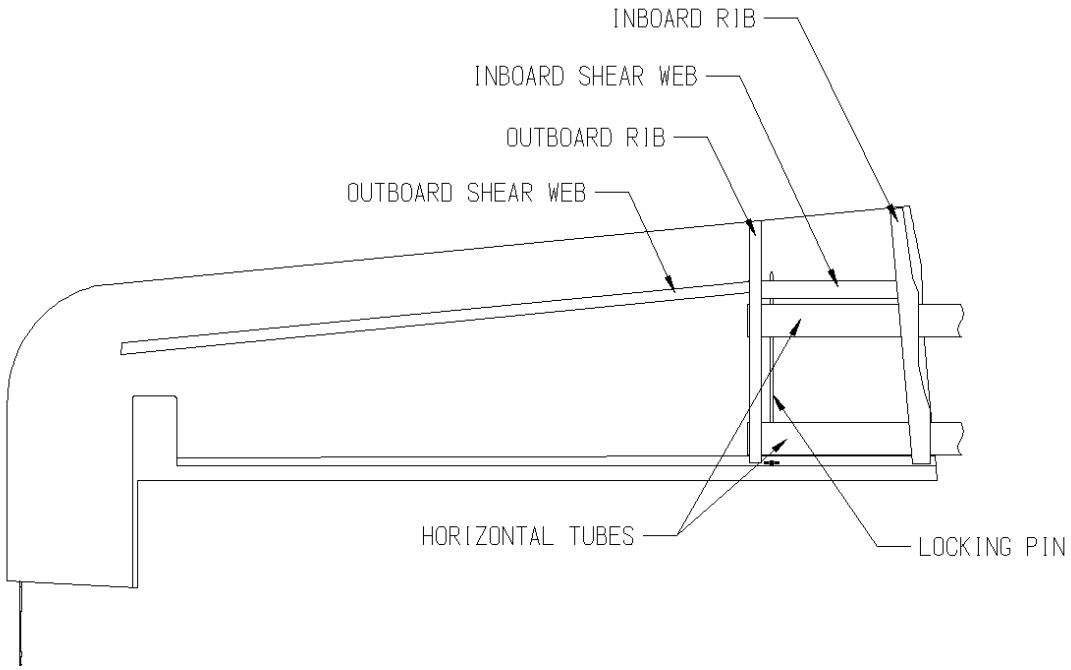

Figure 55 - 1 Horizontal Stabilizer Internal Structure

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55-3. HORIZONTAL STABILIZER REMOVAL AND INSTALLATION

CAUTION

Removal of the horizontal stabilizer may require more than one individual due to its size and relatively tight fit.

a. Use the following procedure to remove the horizontal stabilizer:
   1. Remove the rudder and elevator per the instructions in Chapter 27.
   2. Remove the two AN526C1032R8 screws and two NAS1149F0332P (AN960-10L) washers attaching the locking pin to the aft face of the horizontal stabilizer.
   3. Using a pair of vice grips or similar tool, pull the locking pin, part number LA55702003, straight aft out of the horizontal stabilizer.

CAUTION

Take care not to bend the locking pin while removing it. If the pin becomes bent, installation will be difficult, if not impossible.

   4. Remove the horizontal stabilizer by gently rocking it fore and aft while sliding it straight outboard from the fuselage until the horizontal tubes are free of the outboard rib and bushings. Once the tubes are free of the outboard rib, the horizontal stabilizer will slide freely until the end of the tubes contact the inboard rib and bushings. Again, gently rock the stabilizer fore and aft to clear the inboard bushings.

WARNING

Take care not to rock the stabilizer too hard and cause undue stress on the internal ribs and bushings, as this may cause the bushings to break loose. Should the bushings break loose, contact the manufacturer for further instructions.

   5. Reinstall the horizontal stabilizer sides in the reverse order. When reinserting the locking pin, take care not to bend it, as installation will become difficult or impossible. Should the locking pin become bent and impossible to install, contact the manufacturer for a replacement. Torque the locking pin attachment screws 30 to 36 in.-lbs.

55-4. MAINTENANCE PRACTICES

Regular maintenance of the horizontal stabilizer is not required. The holes in the trailing edge for the elevator hinge attachment screws should be inspected occasionally for wear. The horizontal tubes do not require maintenance.

55-5. DETECTION PROCEDURES FOR DELAMINATION AND LATENT DAMAGE

Due to its composite construction, the horizontal stabilizer should be inspected for delamination and latent damage per the procedures outlined in Chapter 20. The horizontal stabilizer does not have any access panels and needs to be removed in order to inspect the internal structure. Follow the instructions outlined in this chapter for removal of the horizontal stabilizer. Common inspection tools that may be used are mirrors, bore scopes, etc.

55-6. INSPECTION OF HORIZONTAL TUBES

The horizontal tubes should be inspected per the procedures outlined in Chapter 53.

55-7. PAINT SCHEMES

The horizontal stabilizer is to be painted in accordance with instructions in Chapter 51.

55-8. LEADING EDGE TAPE

Optional leading edge anti-chafe tape may be added to the aircraft. The anti-chafe tape is 1 in. wide, adhesive-backed, clear, urethane tape that prevents pitting and wear on the leading edges of the horizontal surfaces.

To apply the tape, thoroughly clean the leading edges using isopropyl alcohol. The 1 in. wide tape should be centered along the leading edge of the surface. The tape should end at the beginning of the horizontal stabilizer tip curve. Use a rubber scraper/squeegee as required to smooth the tape and remove air bubbles from underneath the tape. To remove the tape, slowly pull the tape back on itself, and clean the surface with isopropyl alcohol.

The anti-chafe tape should be inspected at each 100 hr. or annual inspection to ensure that it is not peeling up and is properly attached.

CHAPTER 56

WINDOWS

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56-1. General

a. This chapter covers procedures for the removal and installation of cabin door windows, passenger windows, and the windshield. Care of the windows is covered in this chapter and is repeated in Chapter 12.

1. **Windshield and Windows –** The proper care of the windshield and windows (sometimes referred to as transparencies) is one of the more important exterior care items on the airplane and often the least understood. Refer to the Transparency Repair section of Chapter 51 for approved cleaning materials and procedures.

CAUTION

Do not use anything containing ammonia, aromatic solvents like methyl ethyl ketone, acetone, lacquer thinner, paint stripper, gasoline, benzene, alcohol, anti-ice fluid, hydraulic fluid, fire extinguisher solutions, or window cleaner on the acrylic window surfaces. The use of these substances may cause the surface to craze.

56-2. OVERVIEW OF WINDOW LOCATIONS

Figure 56 - 1 shows all of the windows on the airplane.

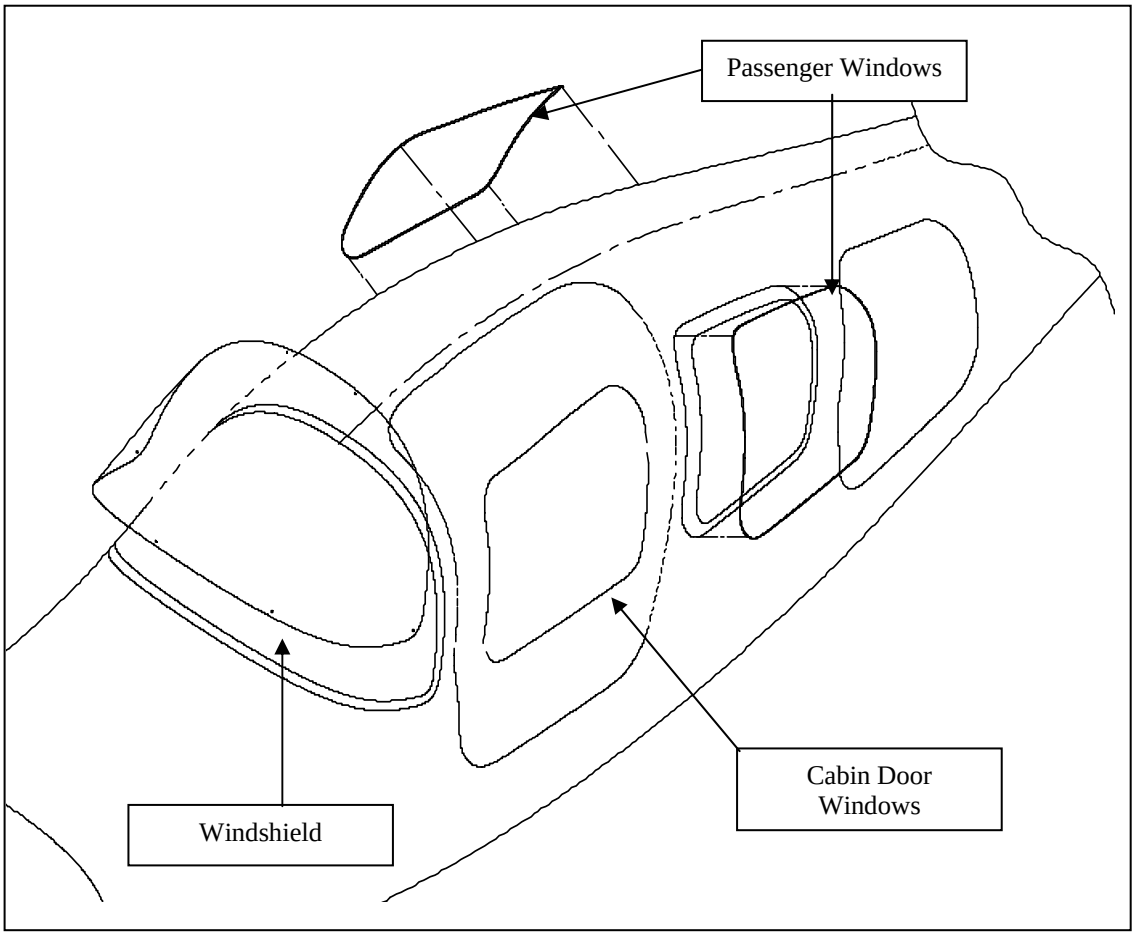

Figure 56 - 1 Aircraft Windows

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56-3. REMOVAL OF WINDOWS

a. All the windows, including the windshield, are bonded to the fuselage. See Figure 56 - 2**.** Removal by the sanding method is the technique recommended in this manual since there is less likelihood of damage to the airplane. Two final points to consider are:
   1. Removal of the windows is a very tedious job. The sanding alone can take as long as eight hours.
   2. The airplane must be transported to a facility with expertise in this area.

b. Use the following steps for window removal.
   1. If the cabin door window is to be replaced, remove the door from the airplane, as described in Chapter 52, using the quick removal method.
   2. If the windshield or passenger window is to be removed, remove the appropriate panels and/or glare shield (see Chapter 25), and completely cover the interior of the airplane to minimize damage from dust and other particles. The importance of doing a thorough job of protecting the interior cannot be over-emphasized.
   3. If the cabin door is removed for window replacement, ensure the surface of the door is protected from accidental scratching. Either remove the locks, handles, and door mechanism, or make sure all openings are sealed from dust and particles.
   4. Cut out the window using a fine-toothed jig-saw. Get as close to the window frame as possible, but not so close that the composite surface could be inadvertently damaged.
   5. Use a power sander and sand off the Plexiglas ring around the frame of the window. For S/N 42030 to 42050, there is glue on the beveled edge that must be removed (see Figure 56 - 2). Use course-grade paper until near the polyurethane glue between the windshield or window and the fuselage flange, then switch to finer grain sandpaper. It is permissible, and even desirable, to leave some of the original glue on the fuselage flange since this aids the adhesion process.

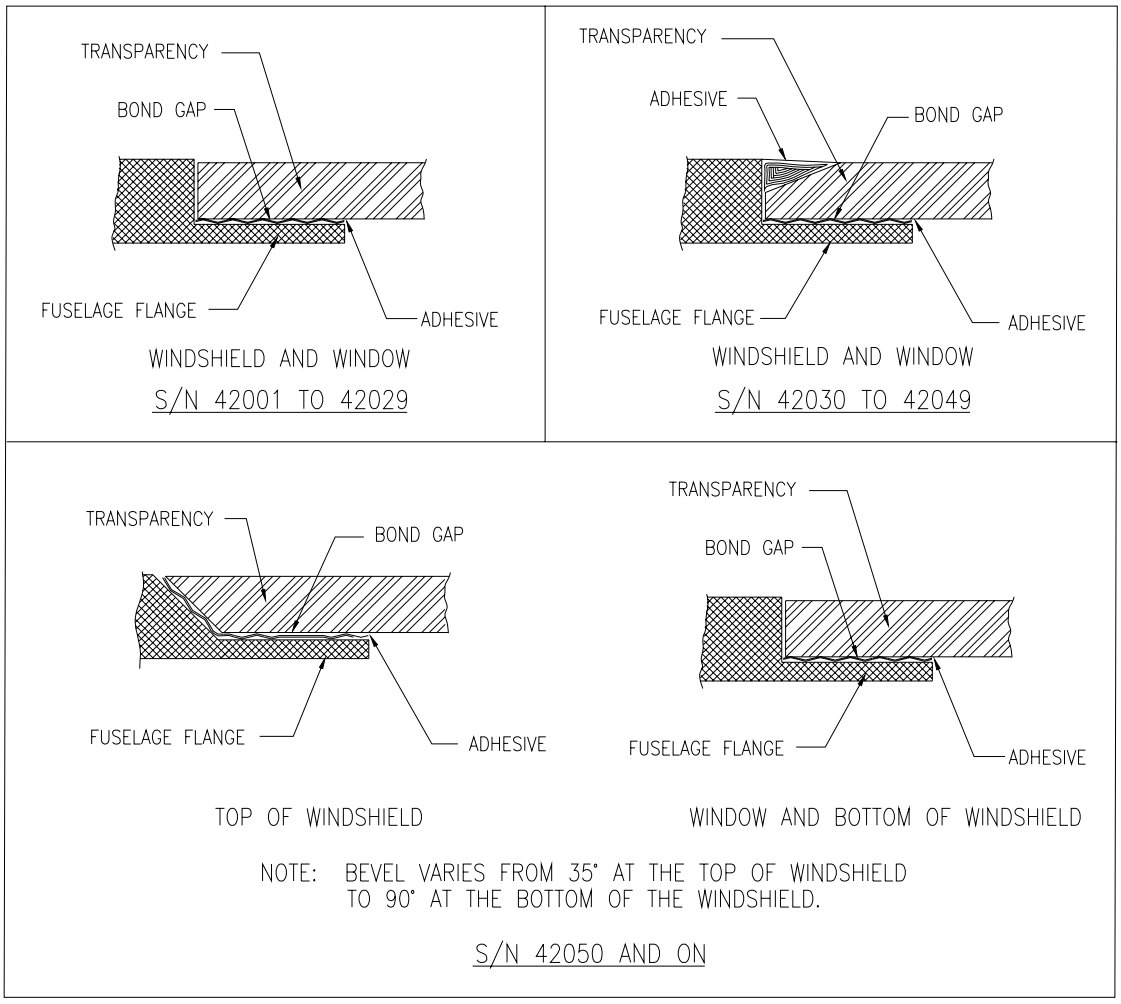


Figure 56 - 2 Existing Windshield or Window Installation at Fuselage Flange

56-4. SURFACE PREPARATION FOR SIDE WINDOW INSTALLATION

a. Prepare the bonding surfaces as follows:
   1. Trim window to fit within the fuselage flange.
   2. Peel off the protective coating on the inside surface of the window approximately 2 in. from the edge.
   3. Bevel the edge of the side window as shown in Figure 56 - 3. Surface roughness not to exceed 250 m-in. Water break test not required on the beveled surface.
   4. While the window is sitting in the fuselage flange, mark the inside edge of the flange on the window using a fine tipped permanent marker.
   5. Using black electrical tape, mask off the inside of the window $^{1}/_{16}$ in. to the inside of the marked flange line. Continue to layer tape on the inside of the window until an overlap on the protective plastic coating is established.
   6. Remove the fine tipped permanent marker line from the inside surface of the window using isopropyl alcohol, 99% anhydrous.
   7. Using black electrical tape, mask off the outside of the fuselage around the window flange. Masking should be 1 to 1 ¼ in. wide.

8. Roughen both the fuselage window flange and the window surface with 80 to 100 grit sandpaper. When sanding the window flange, use a circular sanding motion taking care to only remove the surface gloss of the resin and not to cut into the underlying reinforcement fibers. Abrade the window only within the area that will be bonded to the flange.
9. Remove abrading dust from the bonding surfaces with clean, dry compressed air.
10. Dry-wipe the bonding surfaces with a clean, dry, lint-free cloth or paper towel. Using a solvent-saturated, clean cloth, wet a section of the bonding surface (either Isopropyl Alcohol, 99% anhydrous for the window, or Methyl Ethyl Ketone (MEK) for the initial cleansing of the composite flange followed by a final wipe with Isopropyl Alcohol, 99% anhydrous). Immediately wipe the bonding section with a clean, dry, lint-free cloth or paper wipe, using a single stroke. Natural evaporation of the solvent will result in dispersing contaminants across the bonding surface. After each swipe, turn the drying cloth to a clean surface. Do not reuse a contaminated drying cloth surface. Continue to solvent wash the surface until the drying cloth wipes away clean. A final cleansing of the surface should be performed using deionized water and a clean wiping cloth. Continue until both bonding surfaces are clean.
11. The water-break test should be used to ensure proper bonding surface preparation (see below). If the water-break test fails, repeat steps 8 through 10 until the bonding surface passes the test.

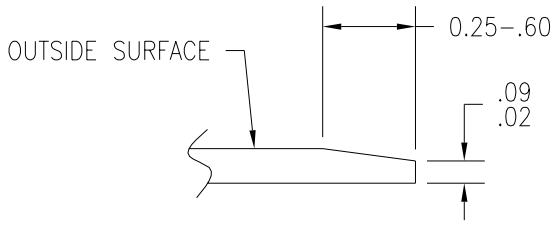
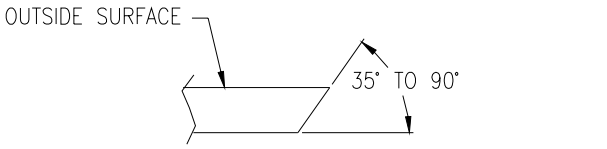


Figure 56 - 3 Side Window Bevel

56-5. SURFACE PREPARATION FOR WINDSHIELD INSTALLATION

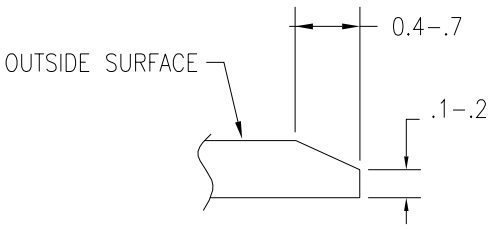
a. Prepare the bonding surfaces as follows:
   1. Trim windshield to fit within the fuselage flange. For S/N 42001 to 42049 do not bevel the edge. For S/N 42050 and on bevel the edge as shown in Figure 56 - 4.
   2. Drill six 0.215 to 0.230 in. diameter holes as defined in Figure 56 - 5 and using Cessna Tool #TA53560000.
   3. Peel off the protective coating on the inside surface of the windshield approximately 2.5 in. away from the edge.
   4. While the windshield is sitting in the fuselage flange, mark the edge of the flange on the windshield using a fine tipped permanent marker.
   5. Using black electrical tape, mask off the inside surface of the windshield $^{1}/_{16}$ in. to the inside of the marked flange line. Continue to layer tape on the inside of the windshield until an overlap on the protective plastic coating is established.
   6. Remove the fine tipped permanent marker line from the inside surface of the windshield using isopropyl alcohol, 99% anhydrous.
   7. Using black electrical tape, mask off the outside of the fuselage around the windshield flange. Masking should be 1 to 1 ¼ in. wide.

8. Roughen both the windshield flange and the windshield surface with 80 to 100 grit sandpaper. When sanding the windshield flange, use a circular sanding motion taking care to only remove the surface gloss of the resin and not to cut into the underlying reinforcement fibers. Abrade the windshield only within the area that will be bonded to the flange.
9. Remove abrading dust from the bonding surfaces with clean, dry compressed air.
10. Dry-wipe the bonding surfaces with a clean, dry, lint-free cloth or paper towel. Using a solvent-saturated, clean cloth, wet a section of the bonding surface (either Isopropyl Alcohol, 99% anhydrous for the window, or Methyl Ethyl Ketone (MEK) for the initial cleansing of the composite flange followed by a final wipe with Isopropyl Alcohol, 99% anhydrous). Immediately wipe the bonding section with a clean, dry, lint-free cloth or paper wipe, using a single stroke. Natural evaporation of the solvent will result in dispersing contaminants across the bonding surface. After each swipe, turn the drying cloth to a clean surface. Do not reuse a contaminated drying cloth surface. Continue to solvent wash the surface until the drying cloth wipes away clean. A final cleansing of the surface should be performed using deionized water and a clean wiping cloth. Continue until both bonding surfaces are clean.
11. The water-break test should be used to ensure proper bonding surface preparation (see below). If the water-break test fails, repeat steps 9 through 11 until the bonding surface passes the test. It is not necessary for the beveled surface to pass the water-break test.



NOTE: BEVEL VARIES FROM 35° AT THE TOP OF THE WINDSHIELD TO 90° AT THE BOTTOM OF THE WINDSHIELD.

S/N 42050 to 42501



S/N 42502 AND ON

Figure 56 - 4 Windshield Bevel

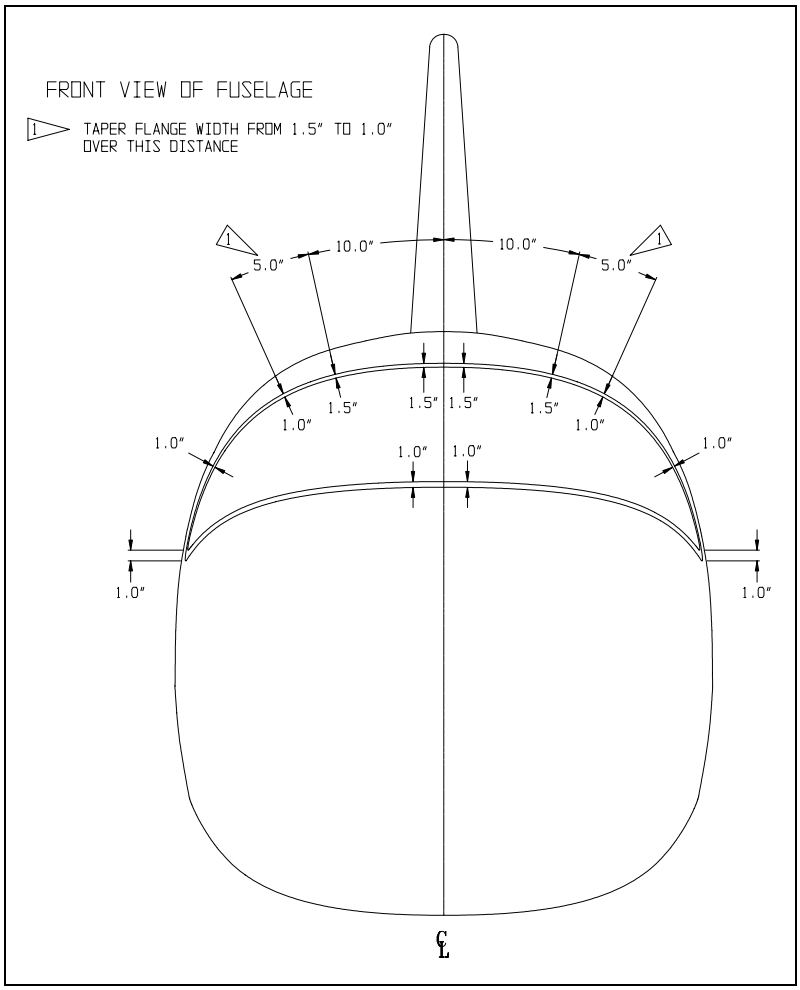


Figure 56 - 5 Windshield Hole Locations

56-6. ADHESIVE PREPARATION

a. Adhesive material must remain sealed and free of contamination. In a clean container, mix the adhesive using one of the ratios specified in Figure 56 - 7.

b. Do not use bare paper or wax-coated paper containers. Use a wooden or metal spatula for mixing.

c. Use the following adhesive preparation procedure:

1. From the original batch of adhesive, keep a separate amount of pure adhesive for wetting out of the bonding surfaces. Flox may be added to the remaining adhesive in the amount specified in Figure 56 - 7, to aid in bond gap formation. The pot life for the adhesive is specified in Figure 56 - 7.

56-7. METHOD FOR BONDING THE FUSELAGE SIDE WINDOWS

a. The following steps should be completed within the working life of the adhesive as noted in Figure 56 - 7.

1. Wet out both the fuselage flange and window bonding surfaces using pure adhesive.
2. Using the remaining adhesive (with flox if specified in Figure 56 - 7), build up a bonding fill layer on the window that is at least $^{1}/_{8}$ in. thick and as wide as the bonding area.

3. Press the window into the flange using even pressure until the outer surface of the window is flush with the outer surface of the fuselage.
4. The window must be clamped flush with the fuselage surface to ensure proper bond gap thickness. The bond gap should measure 0.065 ± 0.015 in.
5. After the window is placed within the flange and satisfactorily held in place, ensure that adhesive fills the bevel to provide a smooth transition between fuselage and window. Excess adhesive should be removed from the inside and outside of the window using a spatula and Isopropyl Alcohol, 99% anhydrous.
6. Remove the black electrical tape on the inside of the window and the fuselage.
7. Any remaining excess adhesive should be wiped from the surface of the window or fuselage using Isopropyl Alcohol, 99% anhydrous.

56-8. METHOD FOR BONDING THE FUSELAGE WINDSHIELD

a. The following steps should be completed within the working life of the adhesive.
   1. Wet out both the fuselage flange and windshield bonding surfaces using pure adhesive.
   2. Using the adhesive with flox, build up a bonding fill layer on the windshield that is at least $^{1}/_{8}$ in. thick and as wide as the bonding area.
   3. Press the windshield into the flange using even pressure.
   4. Insert the clamping screws, washers, and wing nuts as shown in Figure 56 - 6 .
   5. Uniformly tighten the screws until the outer surface of the windshield is flush with the outer surface of the fuselage. Bond gap (see Figure 56 - 2) thickness requirements are met when the windshield remains flush with the outside of the fuselage during curing. The bond gap should measure 0.065 ± 0.015 in..
   6. After the windshield is satisfactorily clamped in place, ensure that adhesive fills the bevel to provide a smooth transition between fuselage and windshield. Excess adhesive should be removed from the inside and outside of the window using a spatula and Isopropyl Alcohol, 99% anhydrous.
   7. The black electrical tape on the inside of the window and the fuselage can now be removed.
   8. Any remaining excess adhesive should be wiped from the surface of the window or fuselage using Isopropyl Alcohol, 99% anhydrous.

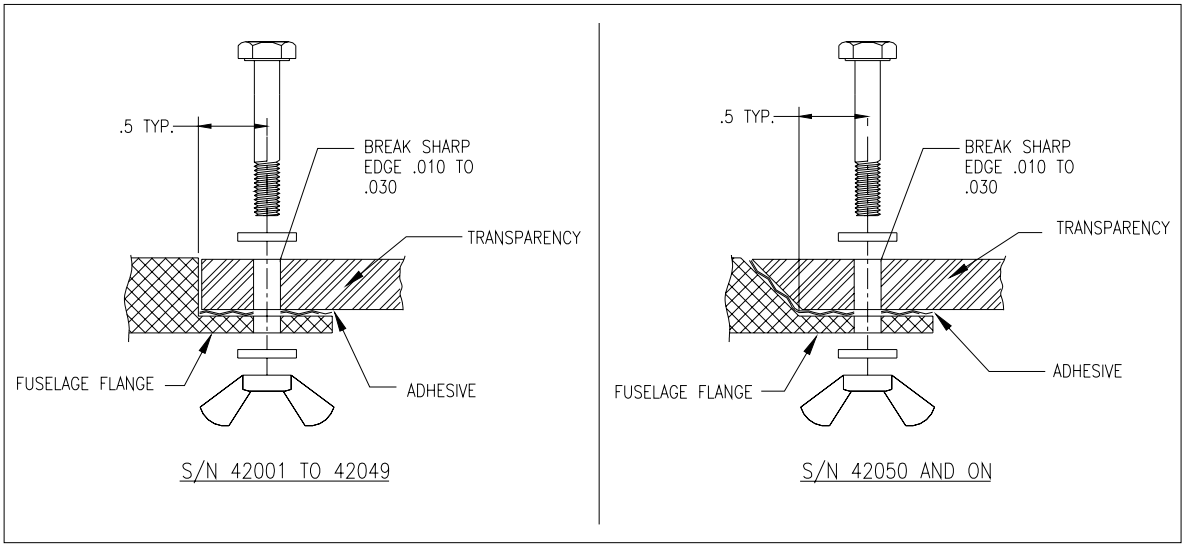


Figure 56 - 6 Windshield Clamping Mechanism

56-9. REPAIRING PAINT CRACKS AROUND WINDOWS

a. Small cracks may appear in the paint around the edge of the windows. The cracking does not involve glass plies or other structure and can be repaired by removing the cracked paint and filler, and refinishing. Use the following procedure to remove the cracked paint and filler:

1. Sand affected area down to the bottom of the exposed crack, paying particular attention to the depth of the crack and making sure that the glass plies underneath are unaffected. If the glass plies are affected, a structural repair is required, and TLC must be notified immediately.
2. Body work, prime, and paint with matching color per instructions in Chapter 51.

56-10. CURING

a. Cure the adhesive as specified in Figure 56 - 7.

Product Designation	Mix Ratio	Cure Cycle
Casco Than 1805 Adhesive with 1806 Curing Agent, mixed with 15% flock (flox)* by weight	100:33 by weight or 100:47 by volume, 1805:1806 **Pot Life:** approx. 60 min.	Apply 200 g/cm (2.85 psi) pressure to joint during handling cure, if desired. **Handling Cure**: Two hours minimum at 65-73 °F and 60% RH or below. **Post Cure**: 7 Days at 60-100 °F or 21 hours at 135 °F.
Casco Than 1805 Adhesive with 1806 Curing Agent	100:33 by weight or 100:47 by volume, 1805:1806 **Pot Life:** approx. 60 min.	Apply 200 g/cm (2.85 psi) pressure to joint during handling cure, if desired. **Handling Cure**: Two hours minimum at 65-73 °F and 60% RH or below. **Post Cure**: 7 Days at 60-100 °F or 21 hours at 135 °F.
Casco Than 1805 Adhesive with 1821 Curing Agent	100:33 by weight or 100:47 by volume, 1805:1821 **Pot Life:** approx. 120 min.	Apply 200 g/cm (2.85 psi) pressure to joint during handling cure, if desired. **Handling Cure**: Two hours minimum at 65-73 °F and 60% RH or below. **Post Cure**: 7 Days at 60-100 °F or 21 hours at 135 °F.
Macroplast UK8303A with UK5400 Curing Agent	100:20 by weight UK8303A:UK5400 **Pot Life**: Approx. 50 minutes	Apply 200 g/cm (2.85 psi) pressure to joint during handling cure, if desired. **Handling Cure**: Four hours minimum at 65-73 °F and 60% RH or below or until the adhesive is hard to the touch. **Post cure**: 60 minutes ramp to 140 ºF, 5 hours at 140 ºF, cool down at a rate not to exceed 10 ºF/min.
Macroplast UK8303A with UK5400 Curing Agent mixed with 2% flock (flox)* by weight	100:20 by weight UK8303A:UK5400 **Pot Life**: Approx. 50 minutes	Apply 200 g/cm (2.85 psi) pressure to joint during handling cure, if desired. **Handling Cure**: Four hours minimum at 65-73 °F and 60% RH or below or until the adhesive is hard to the touch. **Post cure**: 60 minutes ramp to 140 ºF, 5 hours at 140 ºF, cool down at a rate not to exceed 10 ºF/min.

***** Approved flock : W200 manufactured by IFC.

Figure 56 - 7 Window Adhesive Information

56-11. POST ASSEMBLY COMPLETION

a. For post assembly completion, use the following steps.
   1. Remove the screws from the windshield installation after curing has been completed.
   2. Fill holes with 3MK15 MICRO K15 GLASS BUBBLES by 3M.
   3. The protective plastic coating on the windows and windshield should be left on as long as possible to protect the optical quality.

56-12. WATER-BREAK TEST

a. The adequate surface preparation of the fuselage bonding flanges for the installation of the side windows and windshield is extremely important. Inspection of the surface preparation can be accomplished by using the water-break test. The water-break test is extremely simple, time efficient, qualitative, and cost effective. The following steps should be taken to complete the water-break test:

1. Use a squirt bottle to apply a thin layer of deionized water on the prepared surface. A very thin layer of water is best since a thick layer could bridge and conceal an area of contamination, such as a fingerprint. An application of the least amount of water possible is best.
2. The different types of water-break test results are listed below:
   a) Complete beading of water caused by no surface preparation, mold release, or other oily contaminate on the composite surface. This result indicates complete failure of the water-break test. See Figure 56 - 8.

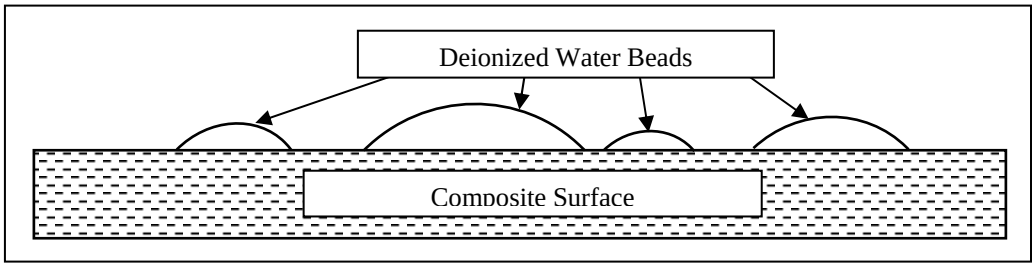


Figure 56 - 8 Water-Break Test Failure Indication

   b) Semi-uniform flow of deionized water over surface. Small areas where the deionized water has been repelled, usually caused by oily fingerprints or drips or splashes of an oily substance such as a beverage. This result indicates failure of the water-break test. See Figure 56 - 9.

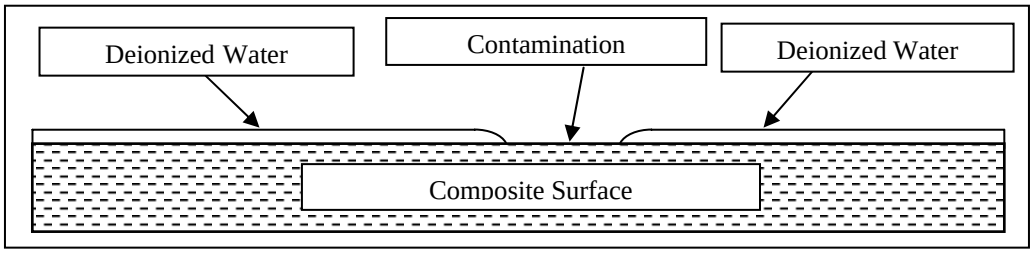


Figure 56 - 9 Water-Break Test Failure Indication

   c) Uniform flow of deionized water over the complete surface. This result indicates that the surface preparation has passed the water-break test. See Figure 56 - 10.

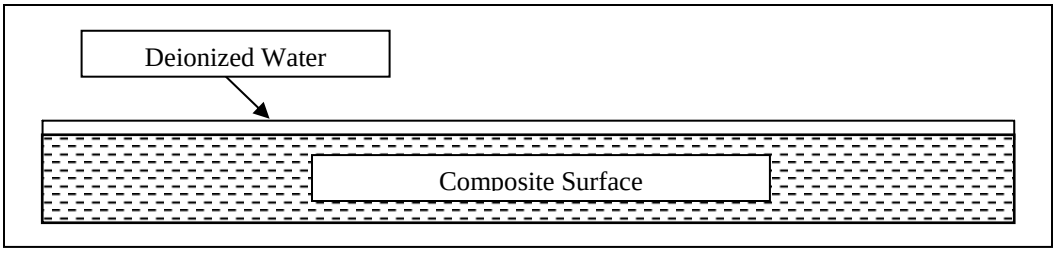


Figure 56 - 10 Water-Break Test Passing Indication

3. After completion of the water-break test, the water can be left to evaporate naturally or can be force-dried using an electric air dryer. The air temperature at the surface should not exceed 120° C (250° F).
4. Once the part has passed the water-break test and dried, it must only be handled using clean white cotton gloves outside of the bonding region. Bonding or priming of the part should occur shortly after step 3 in order to avoid contamination of the bonding surface.

CHAPTER 57

WINGS

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Chapter 57

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57-1. GENERAL WING OVERVIEW

The one-piece wing of the Cessna 350 (LC42-550FG) is composite in construction utilizing both carbon and fiberglass components bonded together using structural adhesive. The attachment to the fuselage is provided by a bolted saddle joint between the wing and the fuselage at the wing to fuselage intersection using a total of 28 bolts, 14 on each side. The primary wing structure includes the upper and lower wing skins, two redundant full span spars, and ribs.

Additional structure includes bellcrank brackets attached to the aft spar, flap and aileron hinges, and the flap and aileron control surfaces. Both of the continuous spars run from WS 198.0 on the left to WS 198.0 on the right. The forward spar is located at FS 105.7, and the aft spar is located at FS 128.8. The fuselage attachment is bolted through root ribs located between BL 20 and BL 24 on both the left and right sides, in front, between, and aft of the spars. Ribs located at WS 46 and WS 145 bind the fuel tank between the forward and aft spars. In the center of the fuel tank is an additional rib at WS 96 that also works as a fuel tank baffle. The aileron and flap control rods run behind the aft spar and through the aft ribs at WS 66.5, WS 105.5, and WS 151.0. On the right wing, a rib is located at WS 151.0, and the left wing has a joggle for the landing and taxi lights. On aircraft in which the wing was constructed with an integrated cuff, the forward rib at WS 151.0 will be absent. The aft rib at WS 151.0 will be absent in some aircraft built in late 2007 and all aircraft built after.

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57-2. PROCEDURES FOR INSTALLATION AND REMOVAL OF THE WING

The major wing subassembly is removable from the fuselage subassembly, as are the ailerons and flaps. For the removal and installation of the wing, a wing cradle is required. The cradle is required to support the wing below the root ribs and should be placed approximately mid-span to prevent tipping of the wing with only the center cradle. Additional supports are required for the fuselage when the fuselage mounted main landing gear is removed to aid in the removal of the wing.

CAUTION

The removal and installation of the wing requires the use of a cradle. Additional support is recommended for the wing approximately mid-span. An additional cradle is required for the fuselage when the fuselage mounted main gear is removed to aid in the removal or installation of the wing.

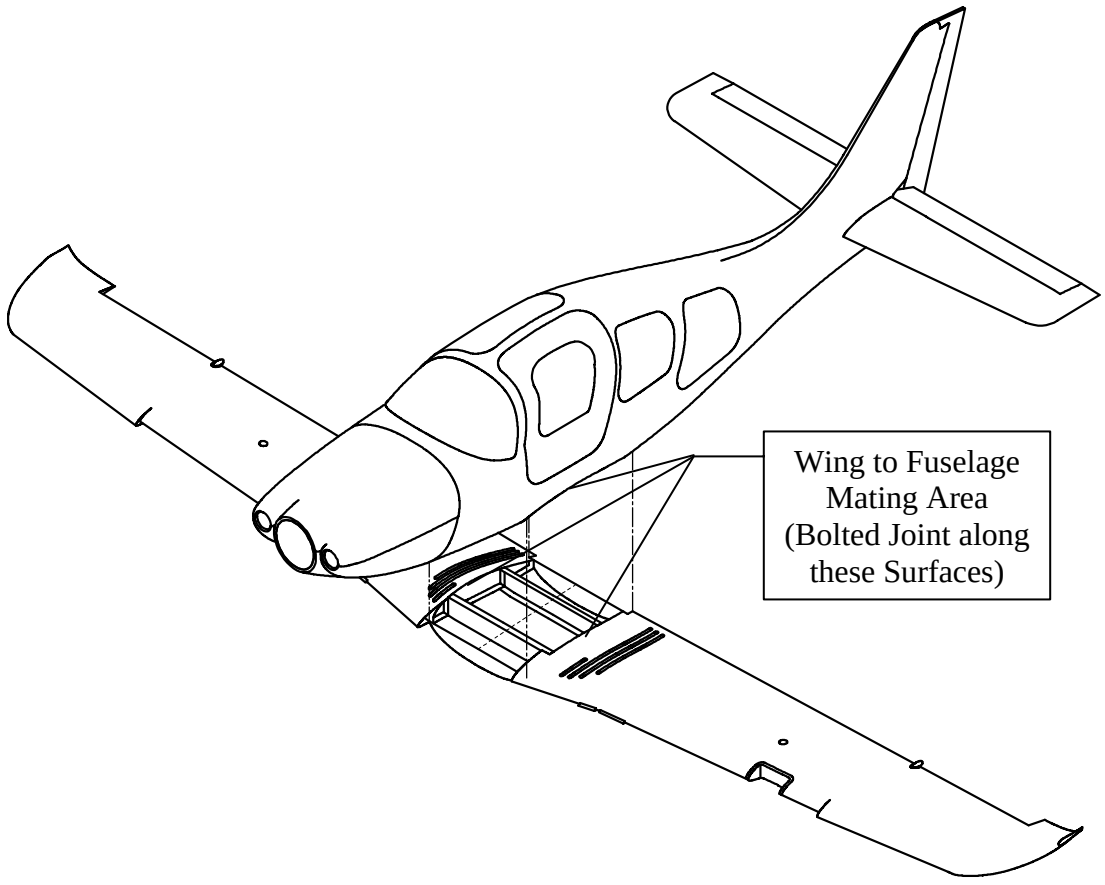


Figure 57 - 1 Wing Fuselage Attachment

57-3. WING REMOVAL

a. Drain all the fuel from the aircraft. This includes both wing tanks, fuel lines in the wings and fuselage, and the sumps and gascolator. When draining/de-fueling the aircraft, fire precautions should be taken and the process performed in a ventilated area. See Chapter 12 for detailed instructions.
b. Remove static wicks to prevent damaging them during the process of removing the wing.
c. Remove the front and aft seats, side interior panels below the doors, carpeting, rear seat footwell, and wing belly panels. See Chapter 25 for detailed instructions.
d. Disconnect and remove the aileron crossover tube (located in the aft wing belly access opening) from the aileron torque tubes. Disconnecting the roll servo of the autopilot may be necessary if installed. See Chapter 27 for detailed instructions.
e. Disconnect the aileron control rods from the aileron torque tubes. See Chapter 27 for detailed instructions.
f. It is recommended that the flap and aileron control surfaces be removed to prevent damaging them during the removal process. See Chapter 27 for detailed instructions.
g. Disconnect all fuselage to wing wiring. Note that primary wiring bundles are located along the side of the fuselage next to the front passenger seat.
h. Disconnect all antenna wiring in the wing belly section of the wing and pull the wiring into the fuselage. Note that any wire ties must be removed prior to pulling the cables.
i. Disconnect the pitot lines between the wing and the fuselage. See Chapter 34 for detailed instructions.
j. Disconnect the fuel feed and return lines from the fuselage to the wing. See Chapter 28 for detailed instructions.
k. Support the wing and fuselage using cradles.
l. Remove the main landing gear, wheel pants, fairings, and landing gear panels per instructions in Chapter 32.
m. With the wing and fuselage supported and all lines, wiring, and controls disconnected, remove the 28 bolts from the wing to fuselage attachment. Note any non-standard or customized washers or spacers must be marked and used upon installing the wing. See Figure 57 - 2.
n. After removing all bolts, nuts, and washers, carefully lower the wing and remove from the fuselage.

NOTE

The hardware removed from the wing to fuselage attachment may be specific to an individual aircraft. This includes, but is not limited to: the washers and spacers if modified from standard hardware and the front seat belt attachments. It is recommended to number and label all the wing attachment bolts, washers, and spacers, for reassembly. Bolts may not be reused but should be labeled so that replacement bolts of identical length to the original may be used.

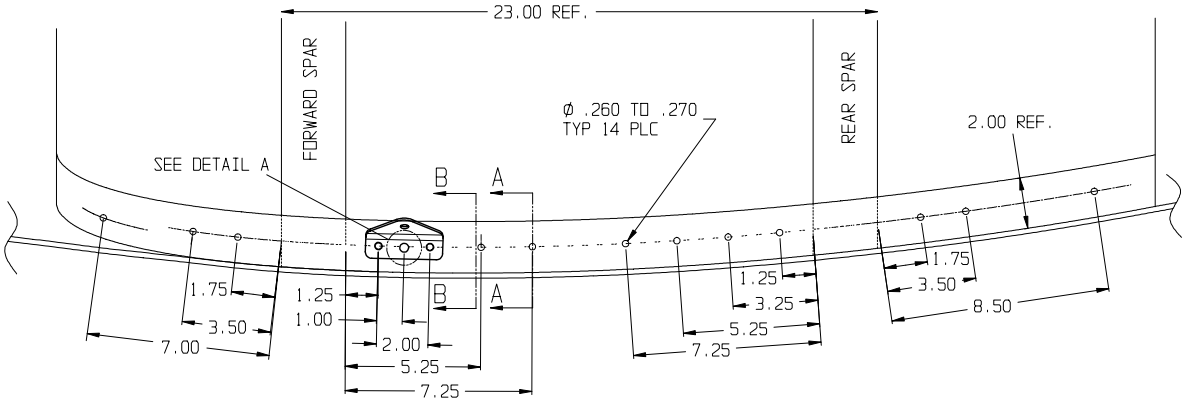


Figure 57 - 2 Wing to Fuselage Attachment Bolt Locations

57-4. WING INSTALLATION

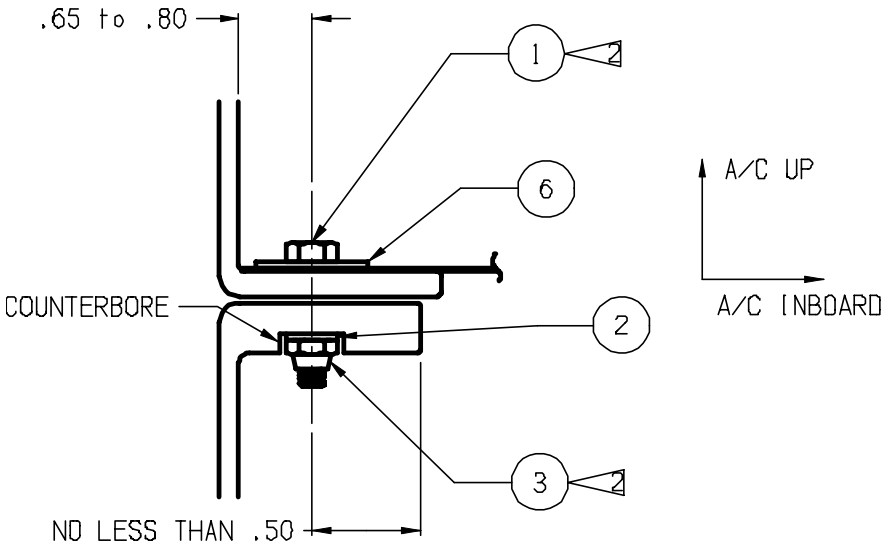
WARNING

Contact the factory for wing installation instructions if installing a new replacement wing, a wing from a different aircraft, or a wing that has a modified/repaired wing attachment area that affects the contour of the wing or fuselage.

a. Wing installation is a direct reversal of the removal when the wing being installed was the same as the one removed from the same aircraft. Bolts are to be replaced with new bolts when installing the wing. Bolts as shown in Figure 57 - 4 labeled 1, are a standard AN4-xxA, Aircraft Undrilled Machined Bolts (125 ksi). Note the use of large area washers on the fuselage and the mounting of the forward seatbelt attachment. Length of each bolt varies from aircraft to aircraft, and replacement bolts should be the same as those removed. Torque values for the bolts and hardware installation are shown in Figure 57 - 3 and Figure 57 - 4. Figure 57 - 5 shows the installation of the bolt, nut, and washer for the center hole of the lap belt bracket. Remove and repair any corrosion on existing hardware prior to installation per this manual. RTV 5818 silicone caulk may be applied to the exterior mating edges between the wing and fuselage (fore and aft), the wing strake, and the fuselage after the wing is attached to provide a seal against vapor, oil spray, etc.

CAUTION

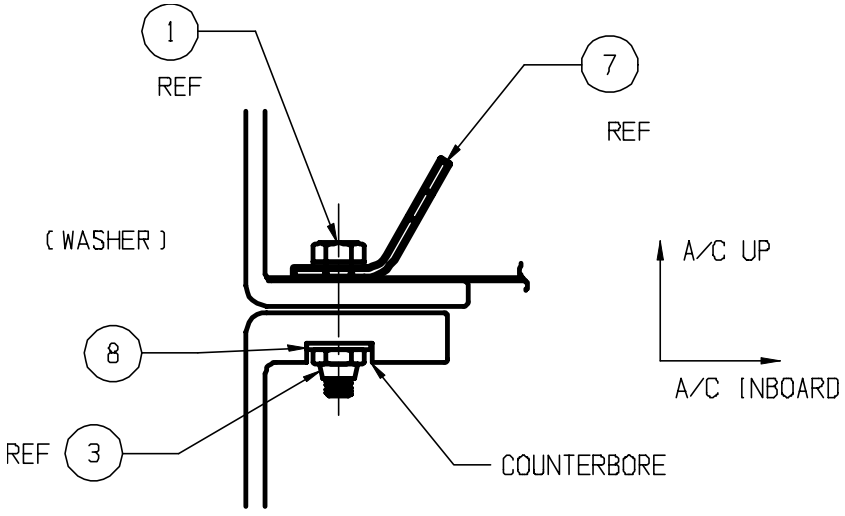
When installing a wing, the bolts must be replaced with new standard AN4-xxA, Aircraft Undrilled Machined Bolts (125 ksi) of the same length as those removed when the wing was removed. No threads should be below the edge of the composite structure when installed.



SECTION A-A
.5 SCALE

NOTES:
1. TOLERANCE ON ALL DIMENSIONS IS ± 0.10
2 TORQUE NUTS 72 TO 84 LB.-IN., WHERE NUTS ARE NOT ACCESSIBLE, TORQUE BOLT HEADS 79 TO 92 LB.-IN.

Figure 57 - 3 Section A-A from Figure 57 - 2



SECTION B- B
SEAT BELT (LAP)
INSTALLATION

NOTES:
1. TOLERANCE ON ALL DIMENSIONS IS ± 0.10
2 TORQUE NUTS 72 TO 84 LB.-IN., WHERE NUTS ARE NOT ACCESSIBLE, TORQUE BOLT HEADS 79 TO 92 LB.-IN.

Figure 57 - 4 Section B-B from Figure 57 - 2

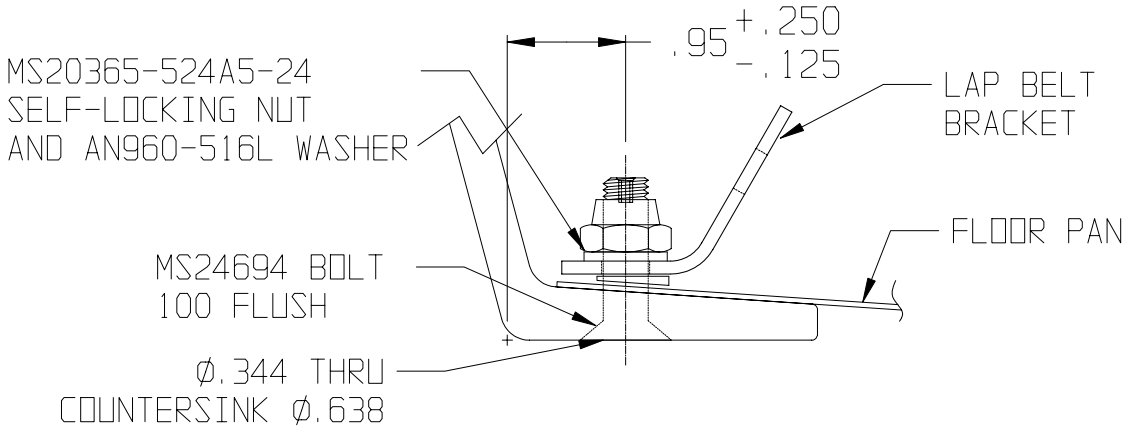


Figure 57 - 5 Detail A from Figure 57 - 2

57-5. MAIN WING STRUCTURE

The one-piece wing of the Cessna 350 is a bonded composite structure providing a strong and aerodynamically clean structure. On the upper wing surface are two fuel filler caps (see Figure 57 - 6), and on the lower surface are various joggles and 14 access panels for inspection, maintenance, and repair of the wing structure and systems. (See Figure 57 - 7.)

The primary load carying structure is comprised of the upper and lower wings skins made of epoxy/fiberglass and honeycomb core. Bonded to the skins are the wing spars made also of epoxy/fiberglass with honeycomb core with additional use of carbon for the spar caps. All ribs are made from epoxy/fiberglass and do not have core.

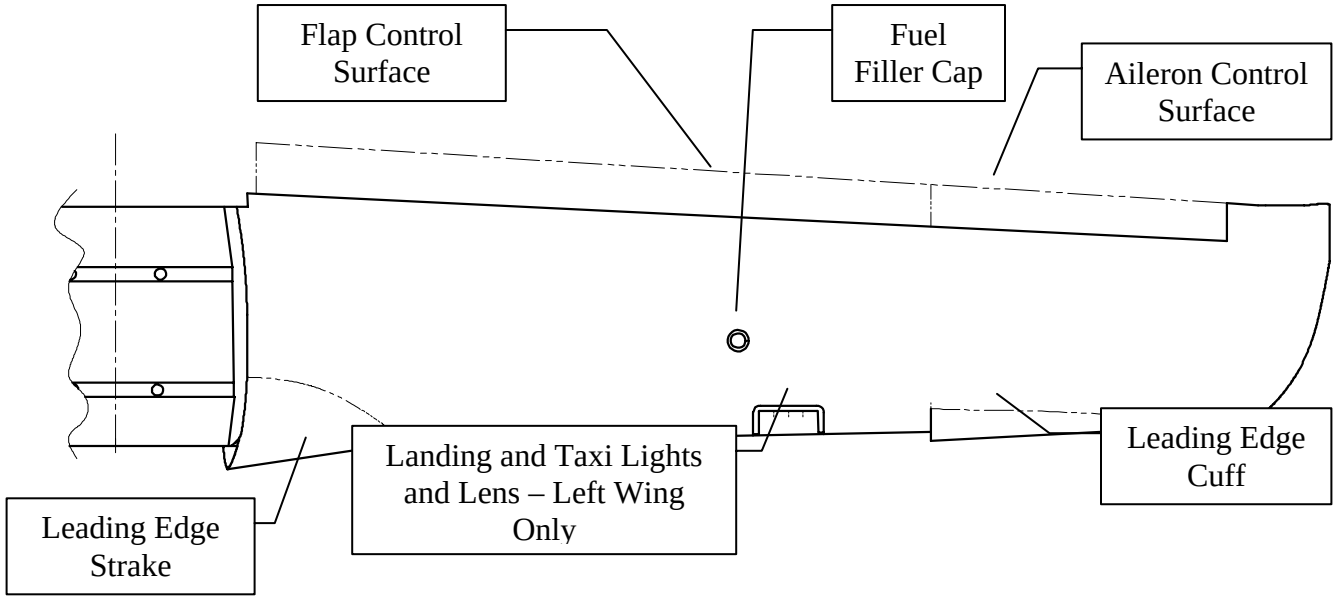


Figure 57 - 6 Upper Wing Surface Key Components (Left Wing Shown)

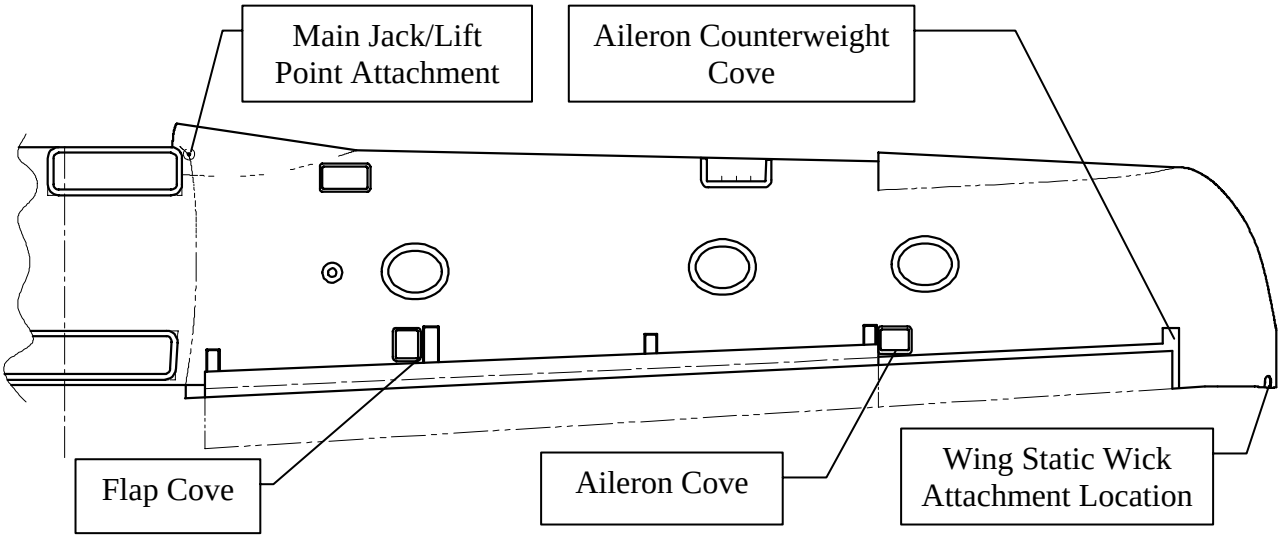


Figure 57 - 7 Lower Wing Surface Key Components (Left Wing Shown)

57-6. LOCATION OF SPARS AND RIBS

a. Figure 57 - 8 shows the wing station (WS) and fuselage station (FS) of all the primary wing structure.

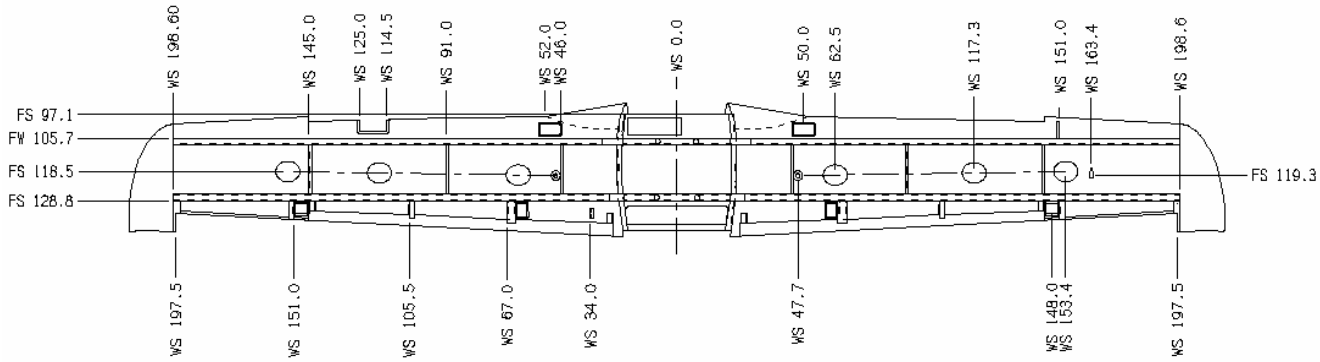


Figure 57 - 8 Wing Structure

1. The two C-section spars are full span between WS 198.6 and WS 198.6 with the shear web of the spars at FS 105.7 and FS 128.8 for the front and aft spar respectively. Both spars are 2.5 in. wide and face inward to the center of the wing.
2. The root ribs include two ribs forward of the forward spar, two center ribs between the forward and aft spars, and the aft root ribs aft of the aft spar. These root ribs provide the structure through which the wing to fuselage bolts pass.
3. The leading edge of the wing has one rib on the right wing only at WS 151.0 with a 1 in. wide bonding flange facing inboard. This rib will not be present in some aircraft. In addition, the inboard strakes also have ribs just outboard of the fuselage.

4. There are three center ribs in each wing located at WS 46.0, WS 91.0, and WS 145.0 with the 1.0 in. wide flanges all facing inboard. The ribs at WS 46.0 and WS 145.0 are the inboard and outboard closeouts for the integral fuel tanks. The center rib at WS 91.0 is also the anti-slosh baffle for the fuel tank.
5. Along the aft section behind the aft spar are three ribs in addition to the aft root rib. These ribs are located at WS 67.0, WS 105.6, and WS 151.0. These ribs provide additional structure for the flap and aileron attachment hinges. The rib at WS 151.0 will be absent in some aircraft built in late 2007 and all aircraft built after.

57-7. LOCATION OF INSPECTION COVERS (ACCESS PANELS)

a. The wing of the Cessna 350 is equipped with two fuel filler caps on the upper wing surface and 16 access panels on the lower wing surface.
b. The access panels are described below, and their location is shown in Figure 57 - 9 Lower Wing Surface Access Panel Locations. There are no wing access panels on the upper surface of the wing. Refer to Figure 57 - 8 for approximate station locations of the access panels.

1	Fuel System Access Panel and Gascolator Inspection Door
2	Flap and Aileron Push-Rod Access Panel
3	Fuel Line Access Panel
4	Flap Outboard Bellcrank Access Panel
5	Inboard Fuel Bay Access Panel
6	Outboard Fuel Bay Access Panel
7	Outer Wing Access Panel and Fuel Vent
8	Aileron Outboard Access Panel
9	Aileron Linear Bearing Access Panel

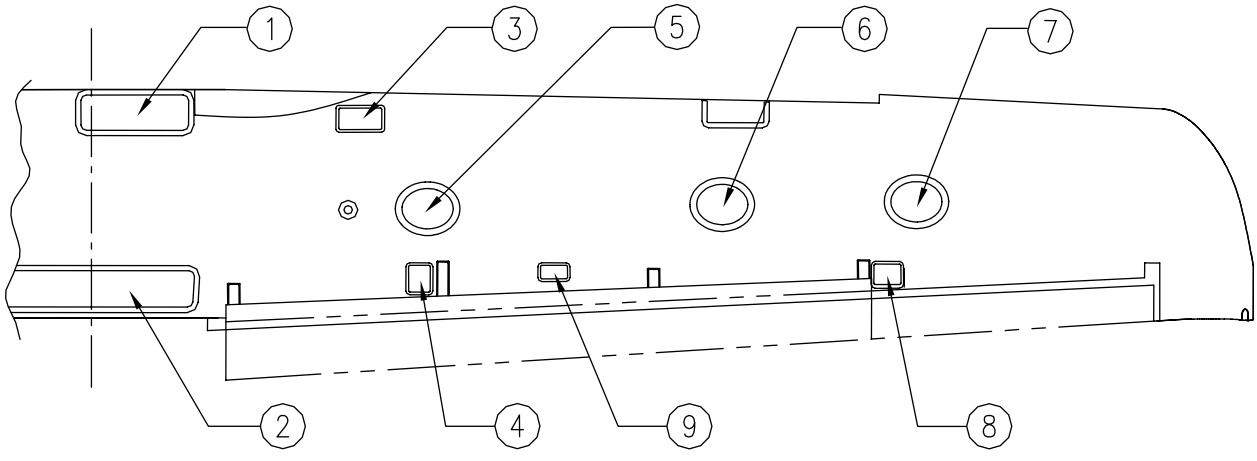


Figure 57 - 9 Lower Wing Surface Access Panel Locations

CAUTION

The inboard and outboard fuel bay access panels (Items 5 and 6 in Figure 57-8) are sealed using PRC-1428 fuel tank access door sealant. Special care must be taken during the removal and installation of these panels due to the wet-wing design of the Cessna 350. Prior to opening of the fuel tanks, de-fuel the aircraft and follow the procedures in Chapter 28 of this manual.

57-8. DETECTION PROCEDURES FOR DELAMINATION AND LATENT DAMAGE

a. Field detection of delamination and latent damage is accomplished primarily by visual inspection. Additional detection methods include tap-testing (refer to Chapter 20), ultrasonic inspection, or other approved methods. Areas where the structure is bonded or where there is a change in the composite lay-up will be apparent during tap-testing, or equivalent inspection method.
   1. Other common names for tap-testing are: coin tap, ring-tap, and metallic ring test.
   2. In general, composite honeycomb structures can be inspected using a tap test as done on metal bonded honeycomb. Use of this method can quickly determine if additional inspection is required.

b. The Cessna 350 is designed to allow for a certain amount of beyond visual damage (BVD) that is difficult to detect except with non-destructive inspection (NDI) methods as described above. Any depression, dents, cracks, gouging, scratches, or scoring that are visible must be investigated for structural damage.
   1. Depressions, Dents, and Scratches may indicate more serious damage below the paint. A minimum of a tap-test must be performed to determine if damage has been done to the structure underneath. If damage is suspected, sanding down through the paint and primer to the lightning mesh should be performed for a more in-depth inspection. See Chapter 51 for classification and repair.
   2. Scoring, Cracking, and Gouging are an indication of damage to the structure and must be inspected, classified, and repaired per Chapter 51 of this manual.

57-9. PAINT SCHEMES

WARNING

The Cessna 350 lightning protection is dependent on paint type and thickness. Only approved paints and paint colors may be used. Paint thickness and locations are also controlled. Paint in accordance with instructions in Chapter 51.

WARNING

The composite construction of the Cessna 350 requires ultraviolet protection in the paint for protection of the composite structure. The topcoat applied to the external aircraft structure provides this protection. When repainting use only paints approved in Chapter 51.

WARNING

The 175°F operating temperature limit does limit the use of dark colors on certain areas of the aircraft. Colors must meet the specifications in Chapter 51.

a. All re-painting must be done in accordance with the instructions in Chapter 51.

57-10. LEADING EDGE TAPE

Optional leading edge anti-chafe tape may be added to the wings, horizontal stabilizer, vertical stabilizer, cowling near the latch, door tread, tow bar attachment point, wheel pants, gear leg fairings, and the landing light lens.. The anti-chafe tape is either a 1 in. wide strip or of a shape and size as required; it is adhesive-backed, clear, urethane tape that prevents pitting and wear on the leading edges.

To apply the tape, thoroughly clean the leading edges using isopropyl alcohol. The 1 in. wide tape should be centered along the leading edge of the surface. The tape should end at the beginning of the wing tip curve and may not be installed over the stall strips or landing light joggle. Use a rubber scraper/squeegee as required to smooth the tape and remove air bubbles from underneath the tape. To remove the tape, slowly pull the tape back on itself and clean the surface with isopropyl alcohol.

The anti-chafe tape should be inspected at each 100 hr. or annual inspection to ensure that it is not peeling up and is properly attached.

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CHAPTER 61

PROPELLER

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Chapter 61

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61-1 GENERAL DESCRIPTION

a. The propeller is a constant speed type that operates by engine driven hydraulic pressure opposing the natural centrifugal twisting moment of rotating blades and spring force to obtain the correct pitch for the engine load. Engine oil is supplied to a power piston in the propeller hub by means of the propeller shaft. An engine driven McCauley governor controls the amount of oil pressure applied. Increasing oil pressure decreases RPM and increases pitch. Decreasing oil pressure increases RPM and decreases pitch.

b. This section discusses the description, removal, and installation of the propeller. The scope of this chapter is to help guide the service and repair technician through the description, removal, inspection, servicing, and installation of the Hartzell 3-bladed propeller and hub, model PHC-J3YF-IRF and blades model, F7691D-1.

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61-2 PROPELLER OVERHAUL

a. Refer to Hartzell Overhaul Manual No. 113B and Standard Practices Manual No. 202A for complete overhaul and maintenance instructions.

NOTE

Only approved propeller shops are authorized for propeller overhaul and major repairs.

61-3 TOOLS & EQUIPMENT

a. The mechanic should be equipped with a complete set of the necessary tools that include the following:

1. Wrenches – ¼ in. through 1 ¼ in.
2. Common and Phillips Head Screwdrivers
3. Pliers – Common, Diagonal Cutters, Needle Nose, Duck Bill, Snap Ring, Safety Wire
4. Ratchets ¼ in., $^{3}/_{8}$ in., ½ in. drive
5. Sockets – ¼ in. drive $^{5}/_{32}$ in. through ½ in., $^{3}/_{8}$ in. drive $^{3}/_{8}$ in. through 1 in., ½ in. drive $^{7}/_{16}$ in. through 1 ¼ in.
6. Sockets (Deepwell) – ½ in. drive, $^{7}/_{16}$ in. through 1 in.
7. Feeler Gauges
8. Leather or Soft Plastic Mallet
9. Torque Wrenches* 0-500 in.-lbs. and 0-100 ft.-lbs.
10. Micrometers*
11. Slide Hammer
12. Pullers
13. Thickness Gauges
14. Vernier Calipers*
15. Small Hole Gauges

* Must be currently calibrated, and the calibration must be traceable to the National Bureau of Standards.

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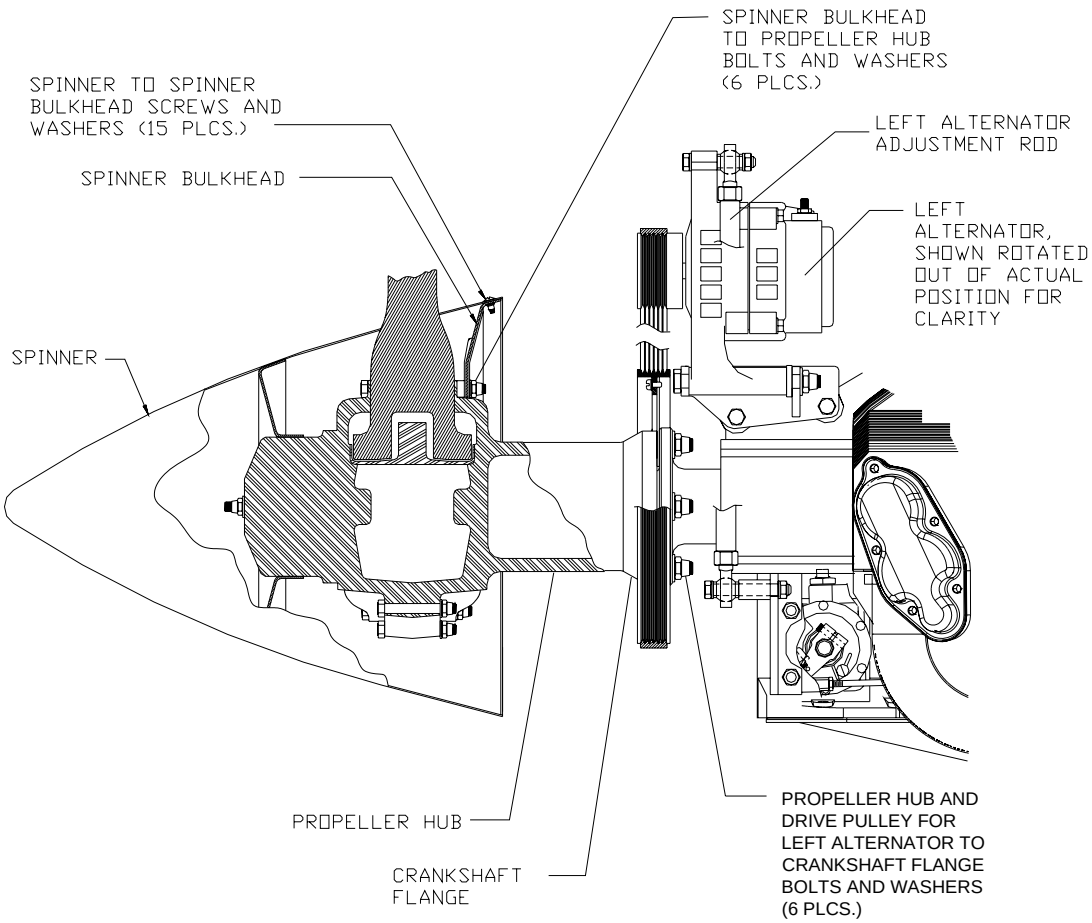
61-4 SPINNER AND PROPELLER REMOVAL AND INSTALLATION

a. **Removal of Spinner and Propeller** (see Figure 61 - 1)

1. Remove the upper and lower engine cowling per instructions in Chapter 71.
2. Disconnect the battery wires attached to the left alternator.
3. Remove 15 spinner to propeller hub attachment screws and fiber washers, and remove the spinner from the propeller hub.
4. Loosen the left alternator adjustment rod and remove the belt from the pulley. The adjustment rod ends are threaded such that rotating the rod itself will loosen or tighten the ends equally.
5. Using a black felt tip marker, mark the position of the propeller hub to the engine crankshaft.
6. Remove six propeller hub flange to engine crankshaft flange retaining nuts and washers, and remove the propeller, hub assembly, and drive pulley from the aircraft.
7. Remove six spinner bulkhead to propeller hub nuts and washers, and remove the spinner bulkhead from the propeller hub.

NOTE

Use clean rags to cover the cavity in the crankshaft and hub.



VIEW FROM PILOT SIDE

Figure 61 - 1 Propeller Hub

b. **Installation of Spinner and Propeller**

1. Install the spinner bulkhead onto the propeller hub. Install six spinner bulkhead to propeller hub retaining nuts. Torque nuts 240 to 264 in.-lbs.
2. Install a new O-ring seal into the propeller hub groove. Place alternator drive belt on propeller hub. Fit the drive pulley and propeller hub assembly onto the engine crankshaft dowl pins with index marks aligned. Install six new propeller hub to engine crankshaft flange retaining nuts and washers. Torque nuts 70 to 80 ft.-lbs. Apply torque seal to all six retaining nuts and hub studs.
3. Install the spinner over the propeller bulkhead so that spinner mounting holes align with the nutplate holes in the spinner bulkhead. Install 15 spinner to spinner bulkhead attachment screws and fiber washers. Torque screws 30 to 36 in.-lbs.
4. Align belt onto drive pulley and alternator pulley.
5. Connect a torque wrench to the alternator pulley retention nut. With an assistant holding the propeller, to prevent the engine crank shaft from rotating, tighten the adjustment rod until the belt no longer slips on the alternator pulley and force applied to the torque wrench is 15 to 25 lb.-ft. Do not over tighten belt. After final belt adjustment has been made torque jam nut on both ends of adjustment rod and nuts attaching the adjustment rod to the alternator and to the bracket attached to the engine 15 to 20 lb.-ft.
6. After the belt is tensioned torque the nut attaching the alternator to the mounting bracket 40 to 45 lb.-ft.

WARNING

Ensure that there is a minimum clearance of ½" between the upper cowling skin and the alternator case, fan, belt, and pulley.

7. Connect the battery wires to the alternator.
8. Install the upper and lower engine cowling per Chapter 71.

61-5 PROPELLER GOVERNOR REMOVAL AND INSTALLATION

a. **Removal of Propeller Governor**

1. Remove the lower engine cowling per instructions in Chapter 71.
2. Disconnect the battery wires attached to the left alternator.
3. Disconnect the governor control cable from the governor (see Figure 61 - 3 and Figure 61 - 5).
4. The left alternator adjustment is accomplished with either a rod or a banana bracket. Loosen the rod; the rod ends are threaded such that rotating the rod itself will loosen or tighten the ends equally. If the banana bracket is installed, remove the two bolts, washers, and alternator tension plate attaching the alternator to the banana bracket.
5. Remove the self locking nut, washers and bolt attaching the alternator adjustment rod or banana bracket to the bracket attached to the governor. Note the position of the washers for reinstallation.
6. Remove four governor to engine attachment nuts and washers (see Figure 61 - 2).
7. Remove the alternator adjustment rod (or banana bracket) attachment bracket (See Figure 61 - 5). Note the position of the washers for reinstallation.
8. Remove the governor from the engine (see Figure 61 - 2).

b. **Installation of Propeller Governor**

1. Install a new governor gasket over the mounting pad studs on the engine (see Figure 61 - 2).
2. Install the governor onto the mounting pad studs (see Figure 61 - 2).
3. Install the alternator adjustment rod (or banana bracket) attachment bracket onto the mounting pad studs (See Figure 61 - 5). Reinstall washers in the reverse order of removal.
4. Install four governor to engine attachment nuts and washers. Torque nuts 180 to 220 in.-lbs. lubricated with TCM Part # 646943 Anti Seize Lubricant or Loctite Anti-Seize Lubricant 767 (see Figure 61 - 2).
5. **Adjustment Rod**
   a) Ensuring that the alternator adjustment rod ends are assembled into the rod equal distance attach the adjustment rod to the attachment bracket.
   b) Connect a torque wrench to the alternator pulley retention nut. With an assistant holding the propeller, to prevent the engine crank shaft from rotating, tighten the adjustment rod until the belt no longer slips on the alternator pulley and force applied to the torque wrench is 15 to 25 lb.-ft. Do not over tighten belt. After final belt adjustment has been made torque jam nut on both ends of adjustment rod, and nuts attaching the adjustment rod to the alternator and to the bracket attached to the engine 15 to 20 lb.-ft.
   c) After the belt is tensioned, torque the nut attaching the alternator to the mounting bracket 40 to 45 lb.-ft.
6. **Banana Bracket**
   a) Insert the alternator tension plate into the banana bracket and hand tighten the hardware attaching the assembly to the alternator.
   b) Connect a torque wrench to the alternator pulley retention nut. With an assistant holding the propeller, to prevent the engine crank shaft from rotating, tighten the alternator tension plate until the belt no longer slips on the alternator pulley and force applied to the torque wrench is 13 to 17 ft.-lbs. Do not over tighten the belt.

Torque the alternator tension plate bolts 12 to 15 ft.-lbs. and ensure that a minimum of two threads extend past the alternator body.

c) After the belt tension has been fully adjusted, torque the nut attaching the banana bracket to the propeller governor bracket and the nut attaching the alternator to the mounting bracket 15 to 20 ft.-lbs.

WARNING

Ensure that there is a minimum clearance of ½" between the upper cowling skin and the alternator case, fan, belt, and pulley.

7. Ensure a minimum of 5 threads engagement of the governor control cable into the rod end of the cable eyelet. Using a new cotter pin, connect the governor control cable to the governor control arm in reverse order as disassembled (see Figure 61 - 3 or Figure 61 - 4).
8. Connect the battery wires to the alternator.
9. Install the lower engine cowling per instructions in Chapter 71.

61-6 GOVERNOR ADJUSTMENT

a. The following governor adjustment procedures are required to allow for proper maximum engine RPM adjustment. Before any adjustment is made to the McCauley governor, the engine oil supply must be at normal operating temperature. For safety reasons it is necessary that two technicians perform the governor adjustment procedure described below.

b. **Propeller Governor Adjustment**

1. Remove engine cowling as described in Chapter 71.
2. Using wheel blocks, block the airplane's main gear.
3. With the emergency brake engaged and feet on the rudder pedals, start the engine and allow the engine oil to warm up to a normal operating temperature of 100°F minimum.
4. With the engine at normal operating temperature and the brakes fully engaged, push the propeller control knob into the full high RPM position.
5. Slowly push in the throttle control until maximum engine RPM is achieved. If maximum RPM is not 2,700 RPM (+0, -25 RPM), shut down the engine. The RPM gauge located on the airplane instrumentation panel may be used to determine engine RPM.
6. Carefully adjust the limit screw on the McCauley governor. Extrapolate for 2,700 RPM assuming ¼ turn equals 25 RPM.
7. Start the engine and perform steps 4 through 6 until a maximum engine RPM of 2,700 is achieved or adjustments do not affect RPM.
8. Shut down the engine. Install the engine cowling as described in Chapter 71.
9. Fly the aircraft.
10. Perform steps 1 through 7, if required, to achieve 2,700 RPM maximum.
11. When the propeller is correctly adjusted there must be 1/8 in. to 1/4 in. nominal distance between the knob and the panel as shown in Figure 76 - 17.
12. Shut down the engine.

13. Safety wire the governor limit screw per MS33540.
14. Install the engine cowling as described in Chapter 71.

c. **Propeller RPM at Static Conditions –** Note that a properly set governor may not achieve 2,700 RPM static, even at full power. The propeller low pitch stop will prevent the required low pitch for full RPM. Proper static RPM is 2,650 to 2,700.

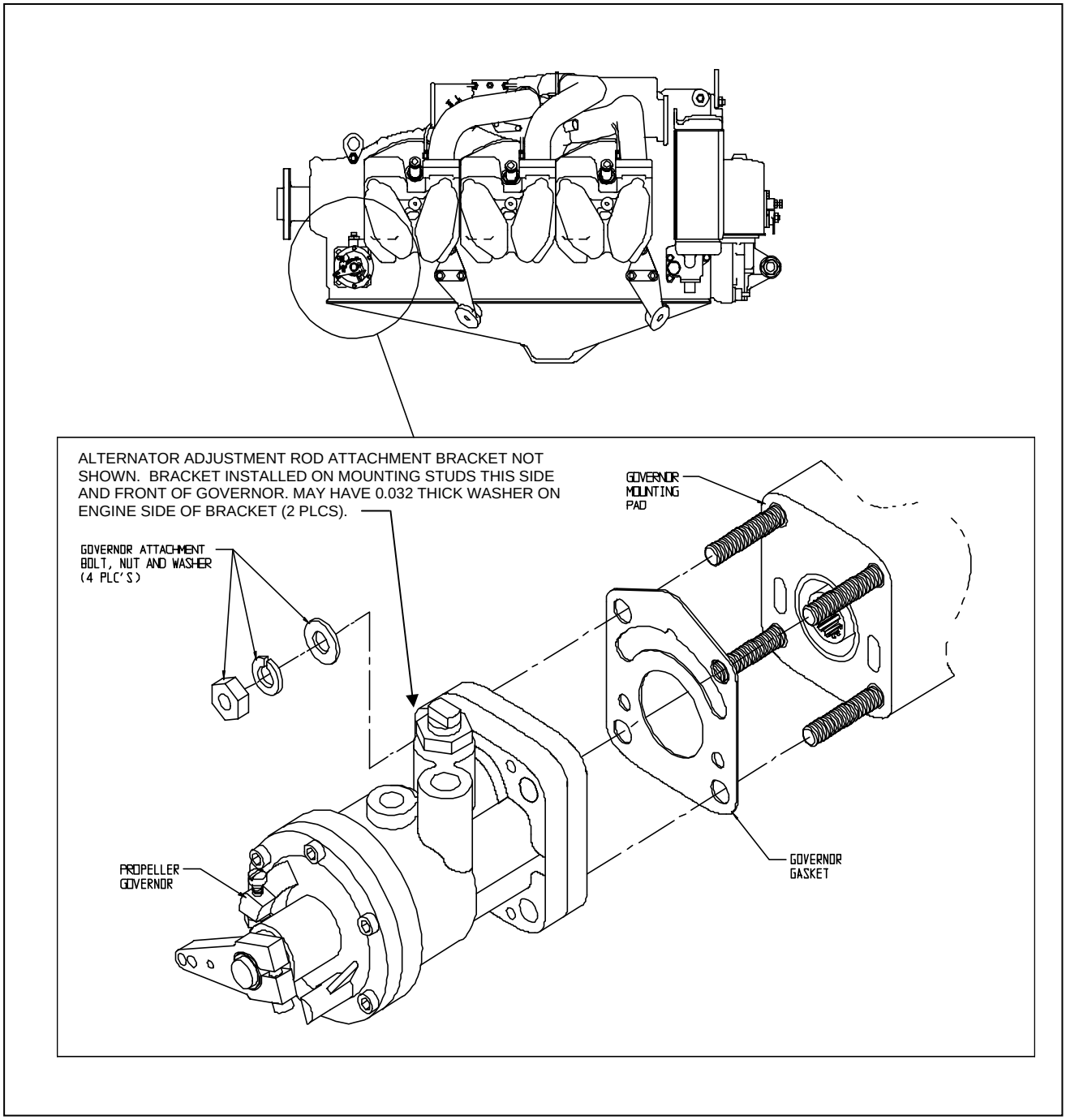


Figure 61 - 2 Governor to Engine Attachment

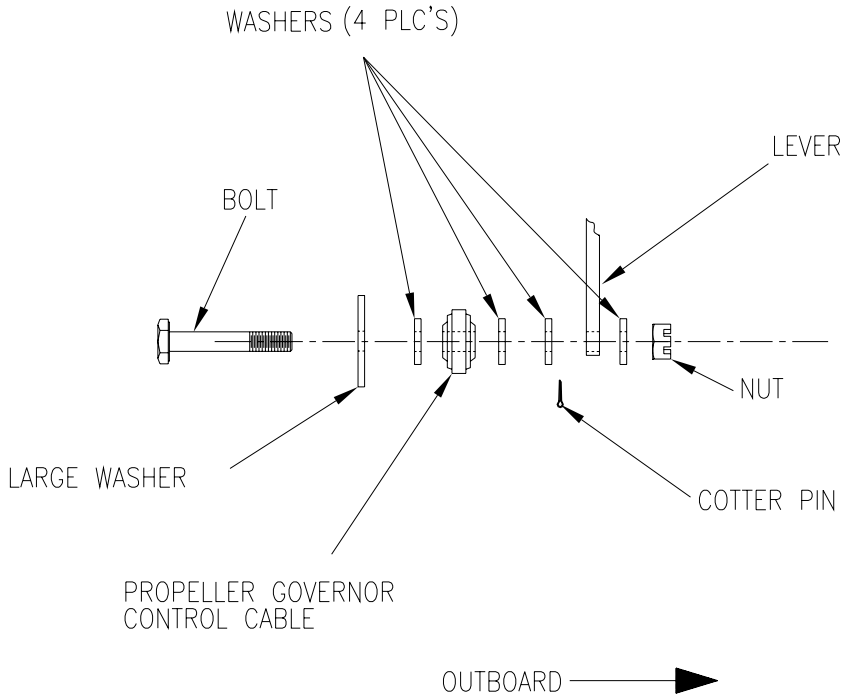


Figure 61 - 3 Governor Control Cable Attachment (S/N 42001 to 42500)

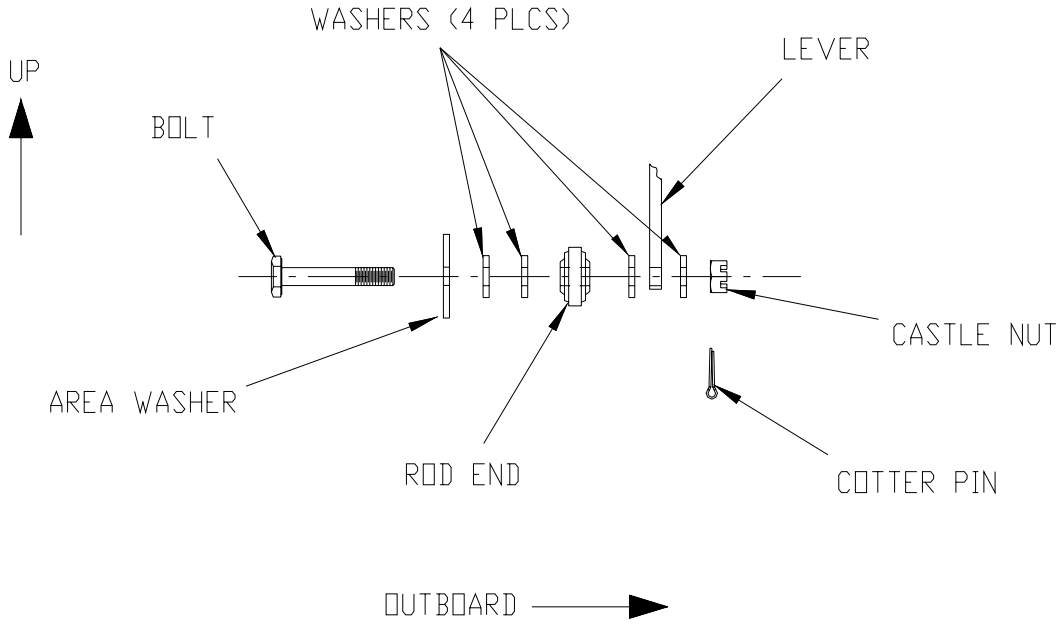


Figure 61 - 4 Governor Control Cable Attachment (S/N 42501 and on)

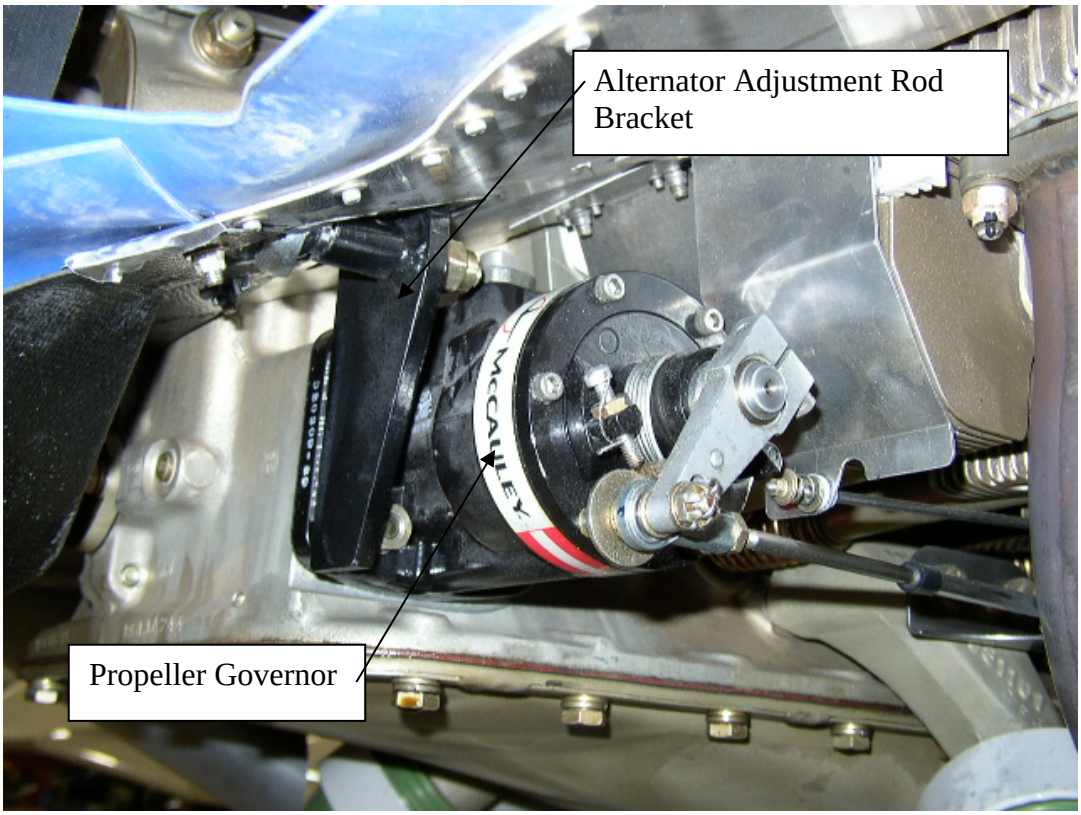


Figure 61 - 5 Alternator Adjustment Rod Bracket

61-7 TROUBLESHOOTING

a. **Symptom –** Propeller surging or wandering is observed.

- **Probable Cause No. 1 –** There is excessive transfer bearing leakage. Engines with excessive transfer bearing leakage can experience surging because the governor may not be able to get enough pressure to the propeller. This will cause a delay in propeller responsiveness, and by the time the propeller responds to earlier governor inputs, they have changed, resulting in propeller wandering.
- **Solution for No. 1 –** Perform a transfer bearing leakage test per engine manufacturer's instructions. If the test indicates a high rate of leakage (even though it may still be on the high side of "acceptable" tolerance), this may be your cause. Install the suspect governor on a known "good" aircraft. If the problem disappears, engine work may be indicated.
- **Probable Cause No 2. –** The engine oil is dirty. Contaminants in dirty engine oil can cause blockages of close tolerance passages in the governor, leading to erratic operation.
- **Solution No. 2 –** Frequent engine oil changes, as often as every 30 hours, should eliminate the problem.

- **Probable Cause No. 3 –** There is excessive play in the aircraft propeller control linkage. Excessive "play" in the linkage between the governor and the cockpit control often leads to erratic operation. Specifically, if the propeller RPM is suddenly changing and holding a new setting on its own, this could indicate a loose linkage.
- **Solution No. 3 –** Trace the linkage, locate unsecured sections, and tighten as needed. Please note that although linkages may appear to allow full governor control while the engine is off, it may not in the air. Engine vibration and "stretch" of the mount during operation can often aggravate the condition; therefore, it is important the entire length of the linkage be properly secured, even if the ends alone are tight.
- **Probable Cause No. 4 –** There is excessive propeller friction. The propeller may be over-resistant to pitch movement. Either excessively tight shimming of the propeller blades, internal corrosion, or part failure causing binding can cause friction. (**NOTE**: This is rarely the cause of RPM malfunction.)
- **Solution No. 4 –** Check amount of blade play. A total lack of blade "shake" may indicate excessively tight blade shims. If this is suspected, have the propeller checked by a qualified FAA approved propeller repairman. Note that this check and any needed correction can usually be performed with the propeller installed on the aircraft.

CHAPTER 71

POWERPLANT

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71-1. General Description

This document describes the troubleshooting, maintenance, removal, installation, and storage of the IO-550-N engine. For engine related information not found in this document, refer to applicable Teledyne Continental Motors maintenance manuals.

71-2. Tools & Equipment

The mechanic should be equipped with a complete set of the necessary tools that include the following:

a. Wrenches – ¼ in. through 1 ¼ in.
b. Common and Phillips Head Screwdrivers
c. Pliers – Common, Diagonal Cutters, Needle Nose, Duck Bill, Snap Ring, Safety Wire
d. Ratchets ¼ in., 3/8 in., ½ in. drive
e. Sockets – ¼ in. drive 5/32 in. through ½ in., 3/8 in. drive 3/8 in. through 1 in., ½ in. drive 7/16 in. through 1 ¼ in.
f. Sockets (Deepwell) –1/2 in. drive, 7/16 in. through 1 in.
g. Feeler Gauges
h. Leather or Soft Plastic Mallet
i. Torque Wrenches* 0-500 in.-lbs. and 0-100 ft.-lbs.
j. Micrometers*
k. Slide Hammer
l. Pullers
m. Thickness Gauges
n. Vernier Calipers*
o. Small Hole Gauges

* Must be currently calibrated, and the calibration must be traceable to the National Bureau of Standards.

71-3. Engine Description and Operation

IO-550-N Technical Description
Manufacturer – Teledyne Continental Motors
Model – IO-550-N

Cylinders:
Arrangement – horizontally opposed
Compression Ratio – 8.5:1
Firing Order – 1-6-3-2-5-4
Number of Cylinders – 6
Bore (inches) – 5.25
Stroke (inches) – 4.25
Piston Displacement (in^3) – 550
Brake Horsepower – 310

Operating Limits:
Crankshaft Speed (RPM, rated maximum continuous operation) – 2700
Crankshaft Speed (RPM, idle) – 600 Min.

Fuel Control System – Continental Continuous Flow Injector
Oil Pressure at Idle (psi @ 75° F) – 10
Oil Pressure at Normal operation (psi @ 200° F) – 30 to 60
Oil Sump Capacity (U.S. Quarts) – 8
Oil Temperature Limits:
Minimum for Takeoff – 100° F
Limit – 240° F
Recommended Operational – 170° to 220° F

Ignition Timing (compression stroke, breaker opens):
Right Magneto, degrees BTC – 22° ± 1°
Left Magneto, degrees BTC – 22° ± 1°

Ignition:
Spark Plug – TCM 634675 or Champion RHB32E
Spark Plug Gap – Per manufacturer
Spark Plug Torque (in.-lbs.) – 300 to 360
Magnetos (left and right) – S6RSC-25

71-4. Engine Removal, Cleaning, and Installation

This section provides instructions to remove and install the engine and engine mount.

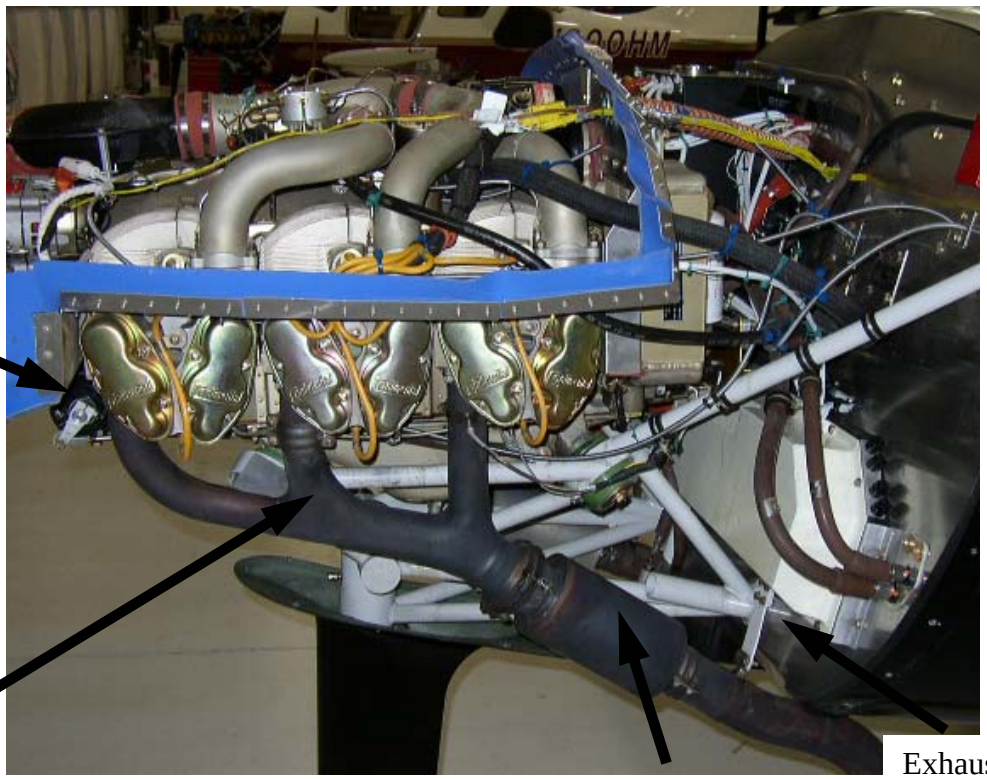

Figure 71 - 1 Engine (S/N 42001 to 42022)

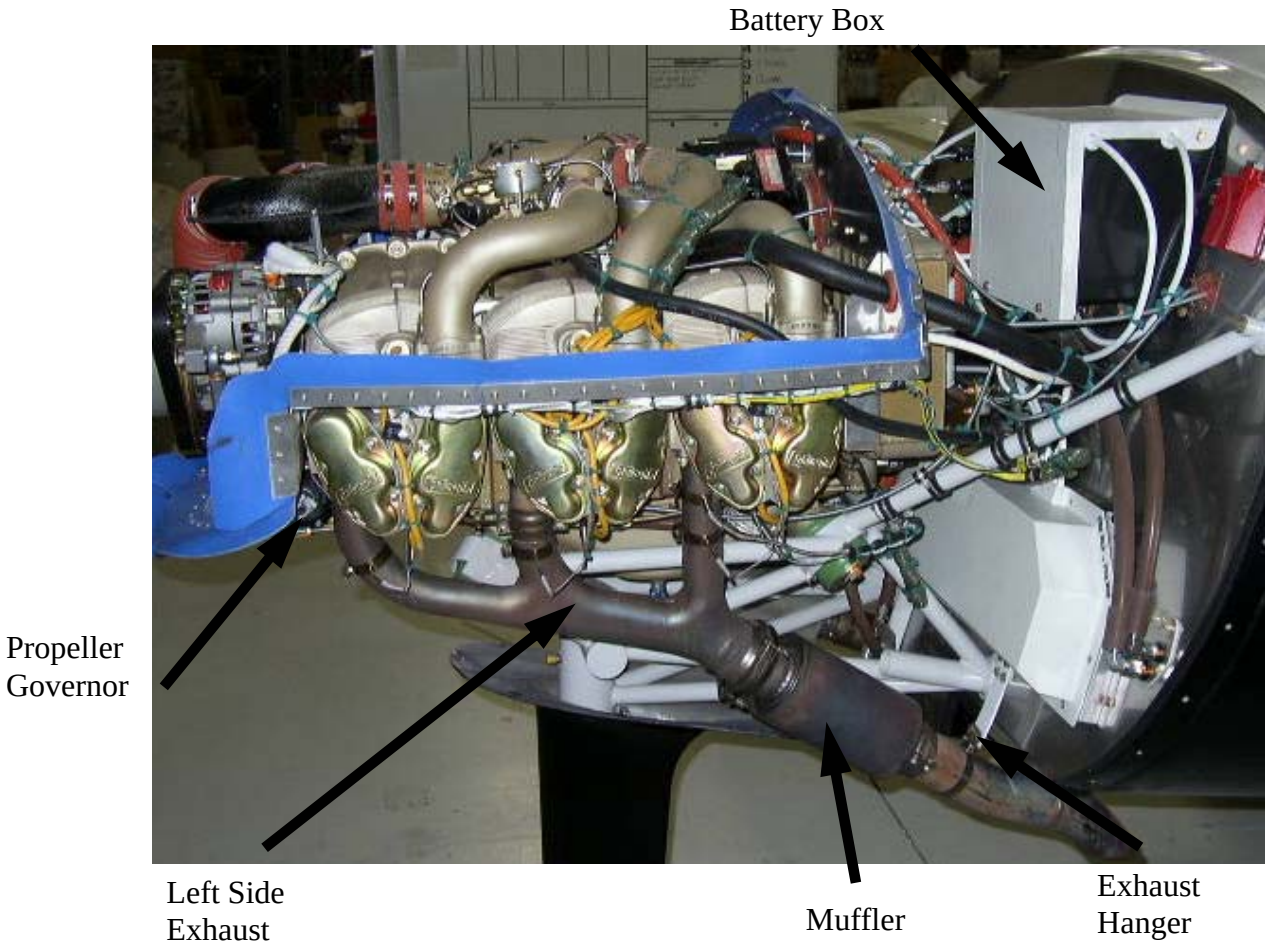

Figure 71 - 2 Engine (42023 and on)

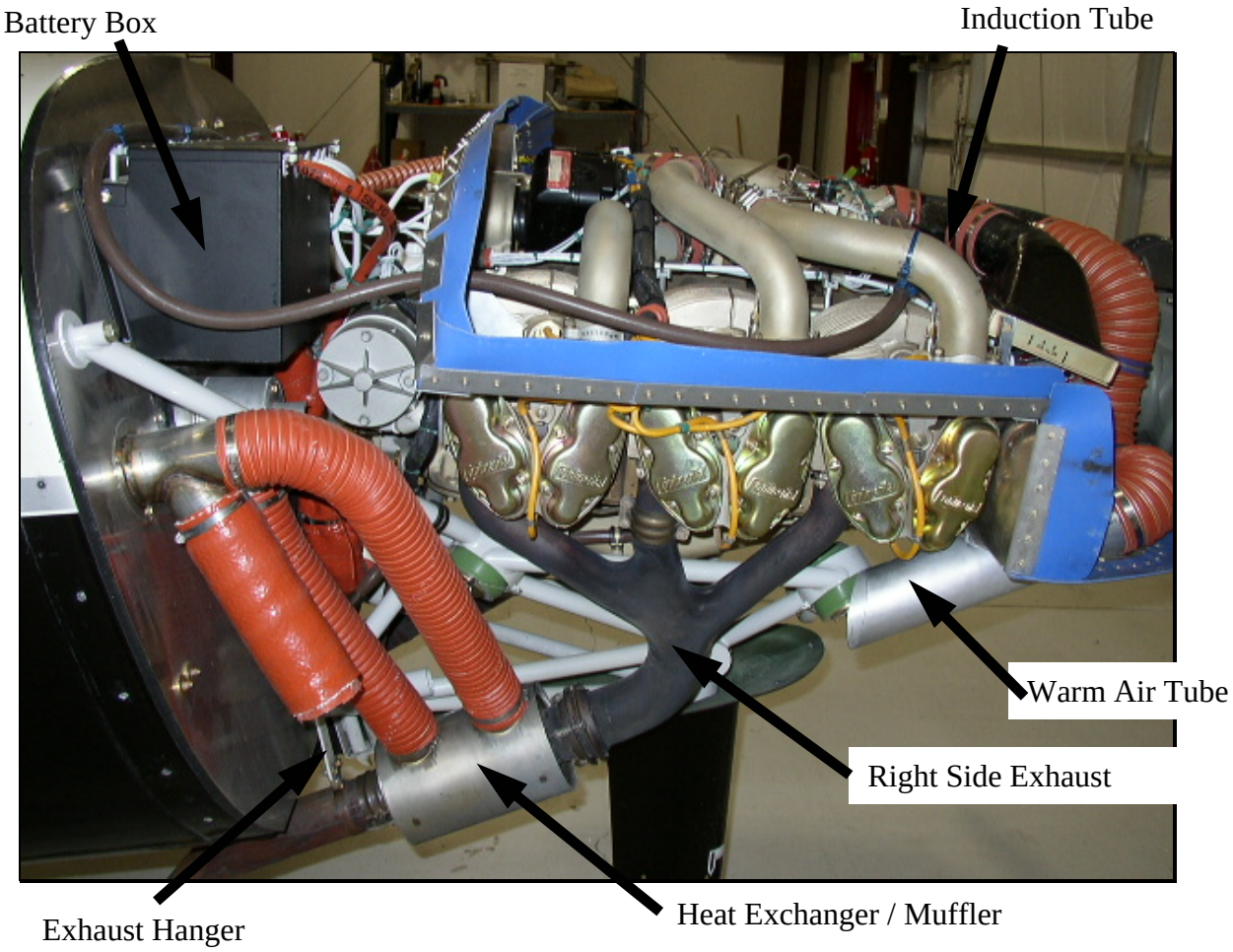

Figure 71 - 3 Engine (S/N 42001 to 42022)

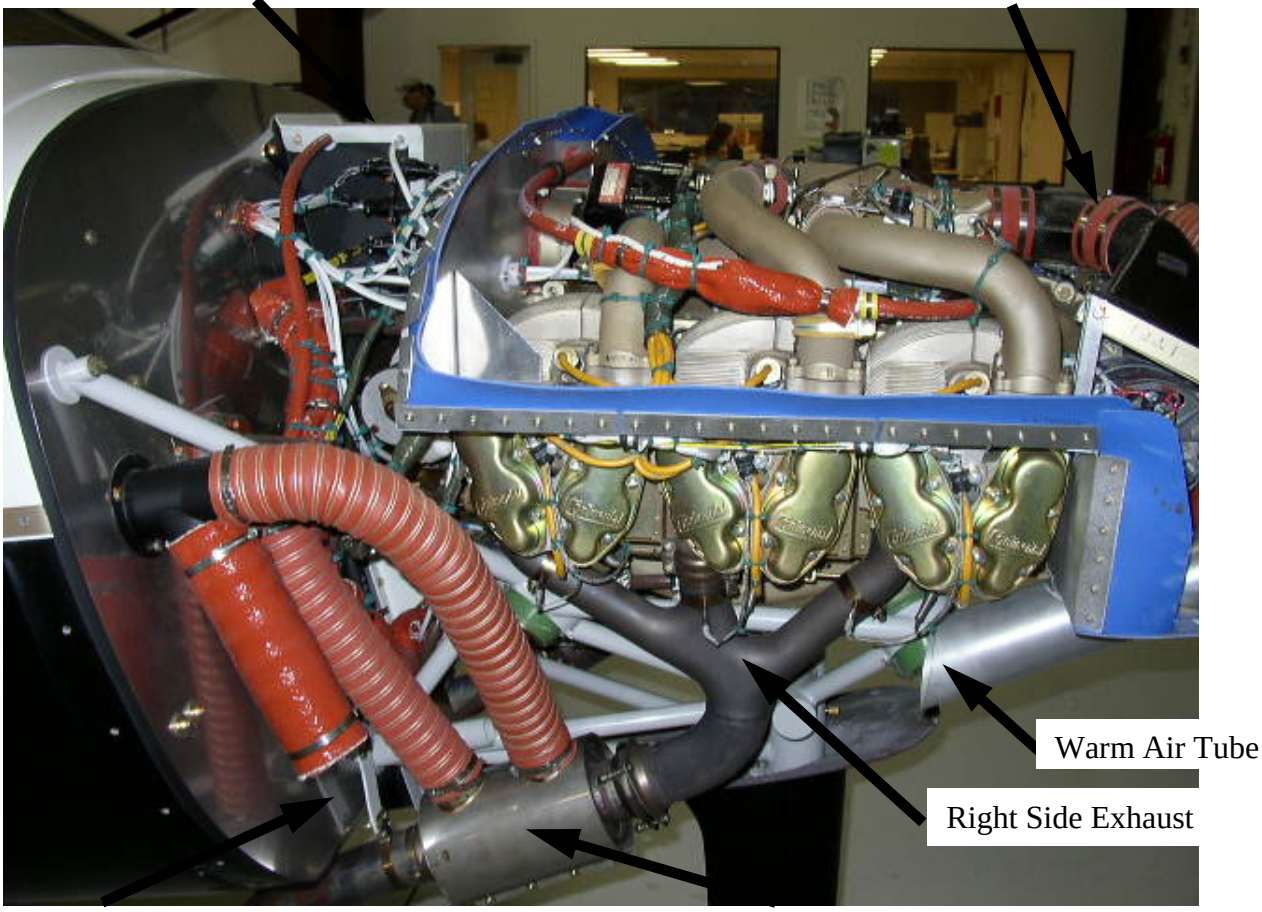

Figure 71 - 4 Engine (S/N 42023 and on)

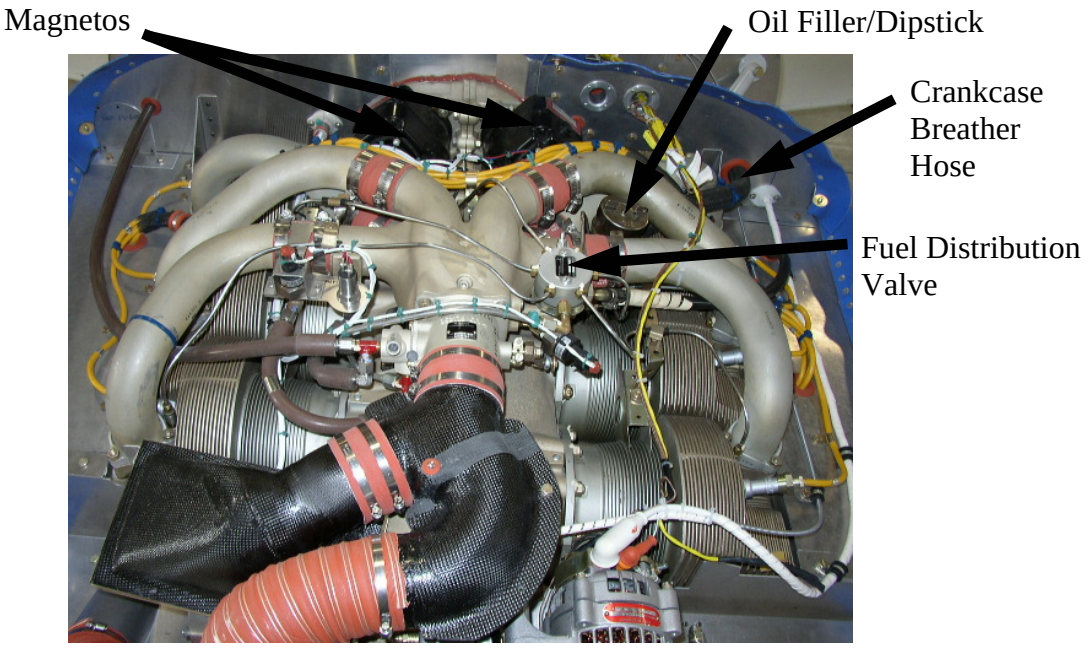

Figure 71 - 5 Engine

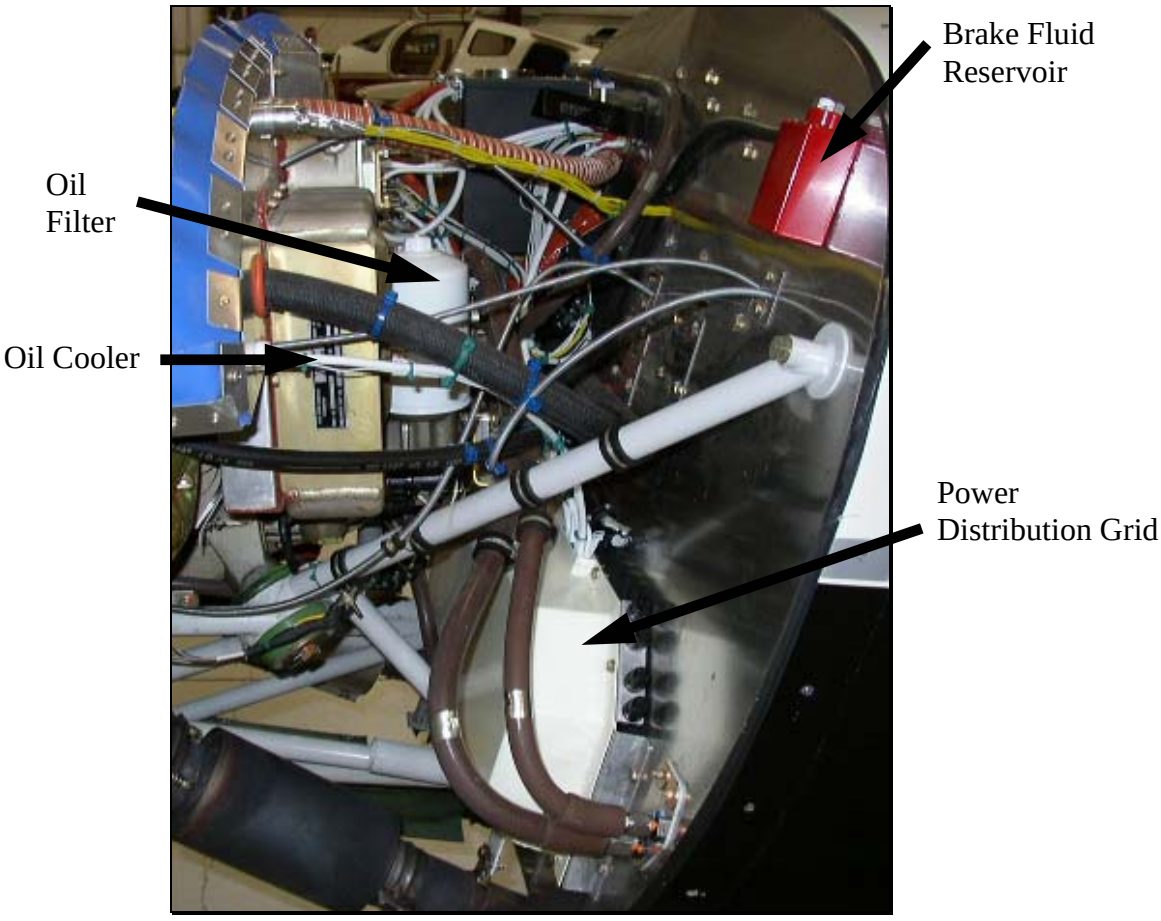

Figure 71 - 6 Engine (S/N 42001 to 42022)

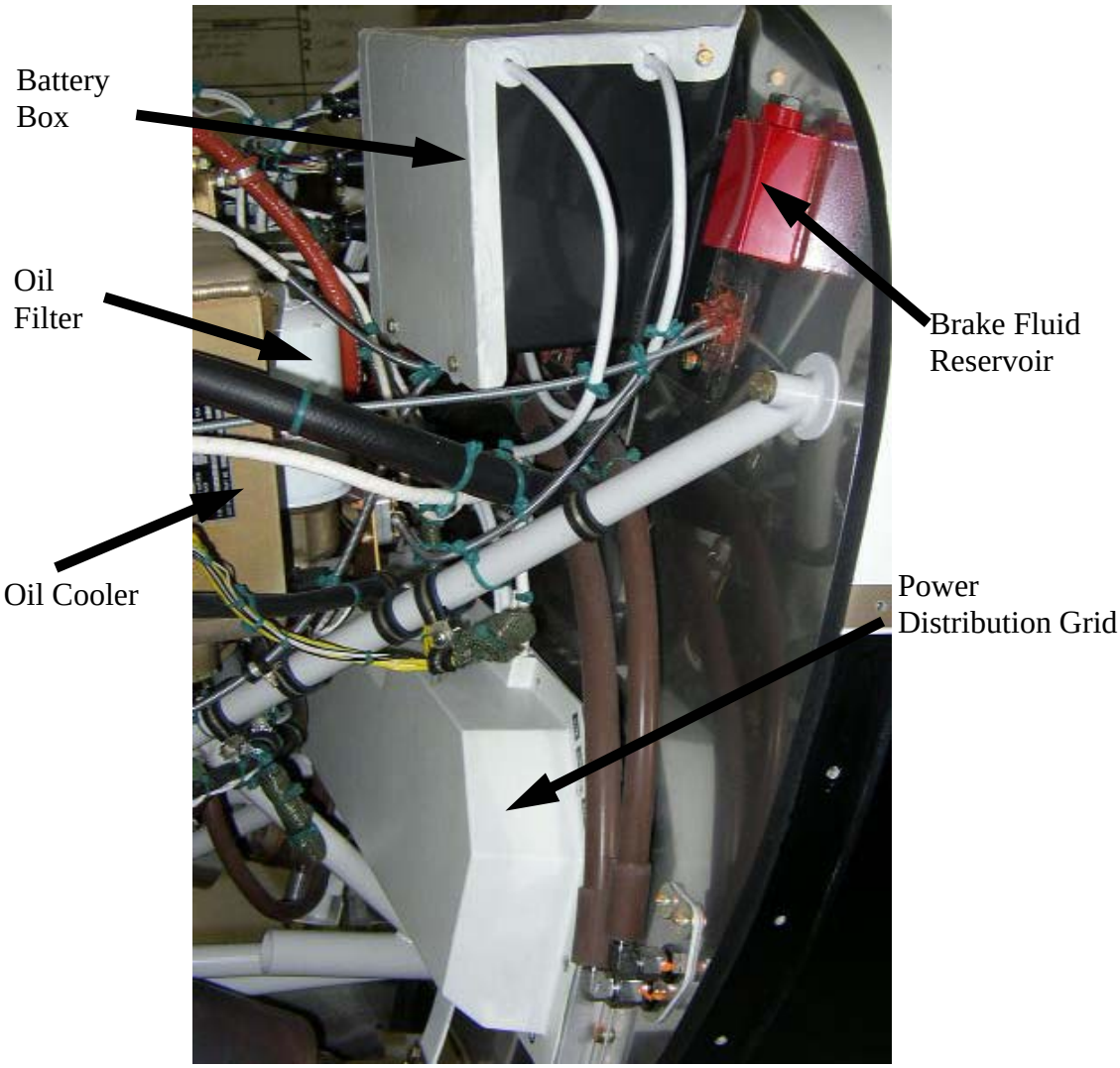

Figure 71 - 7 Engine (S/N 42023 and on)

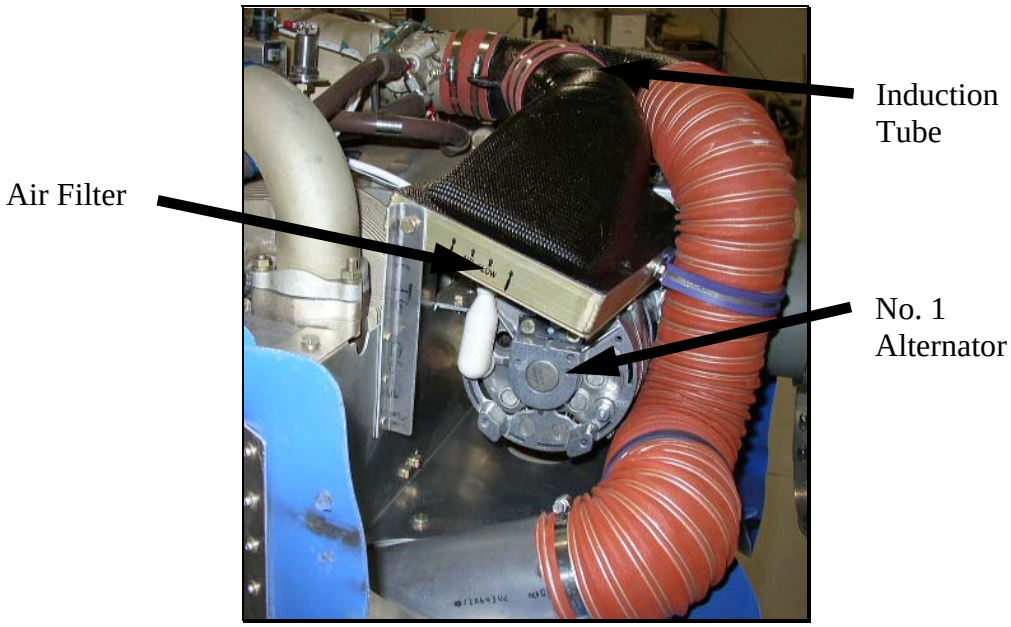

Figure 71 - 8 Engine

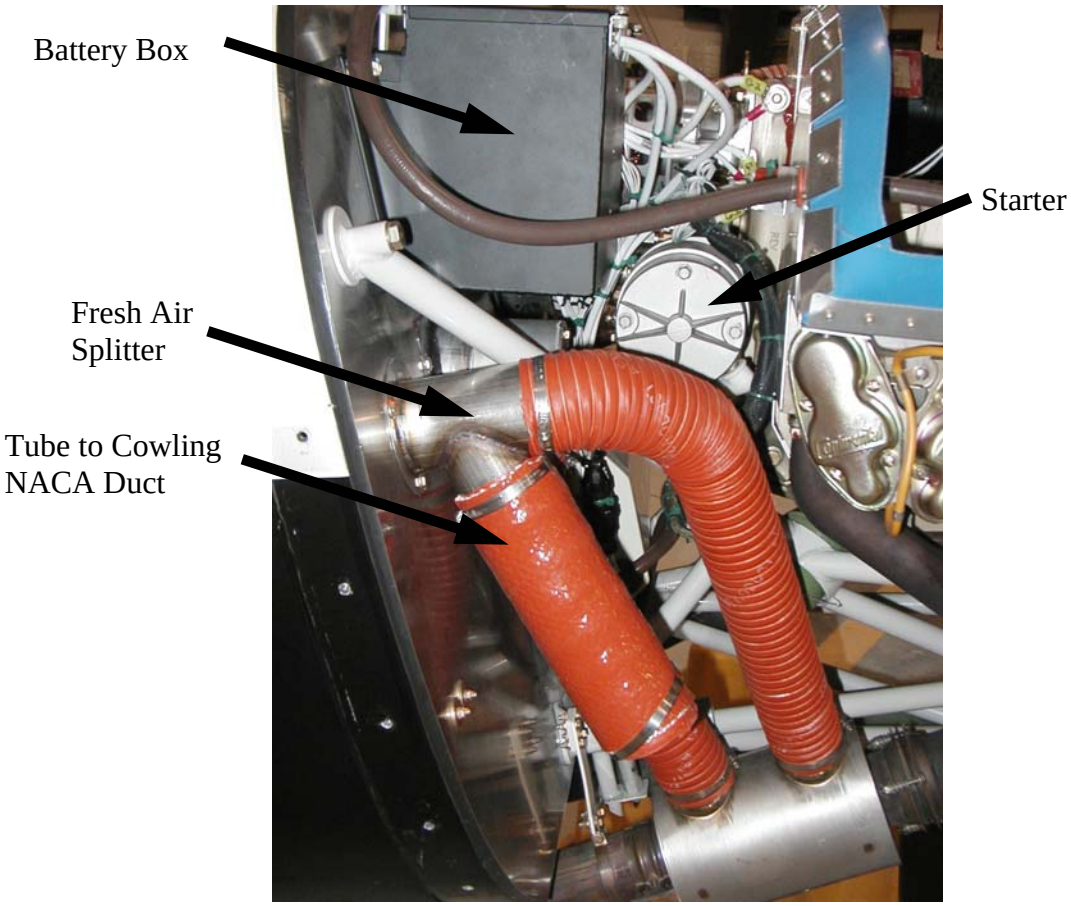

Figure 71 - 9 Engine (S/N 42001 to 42022)

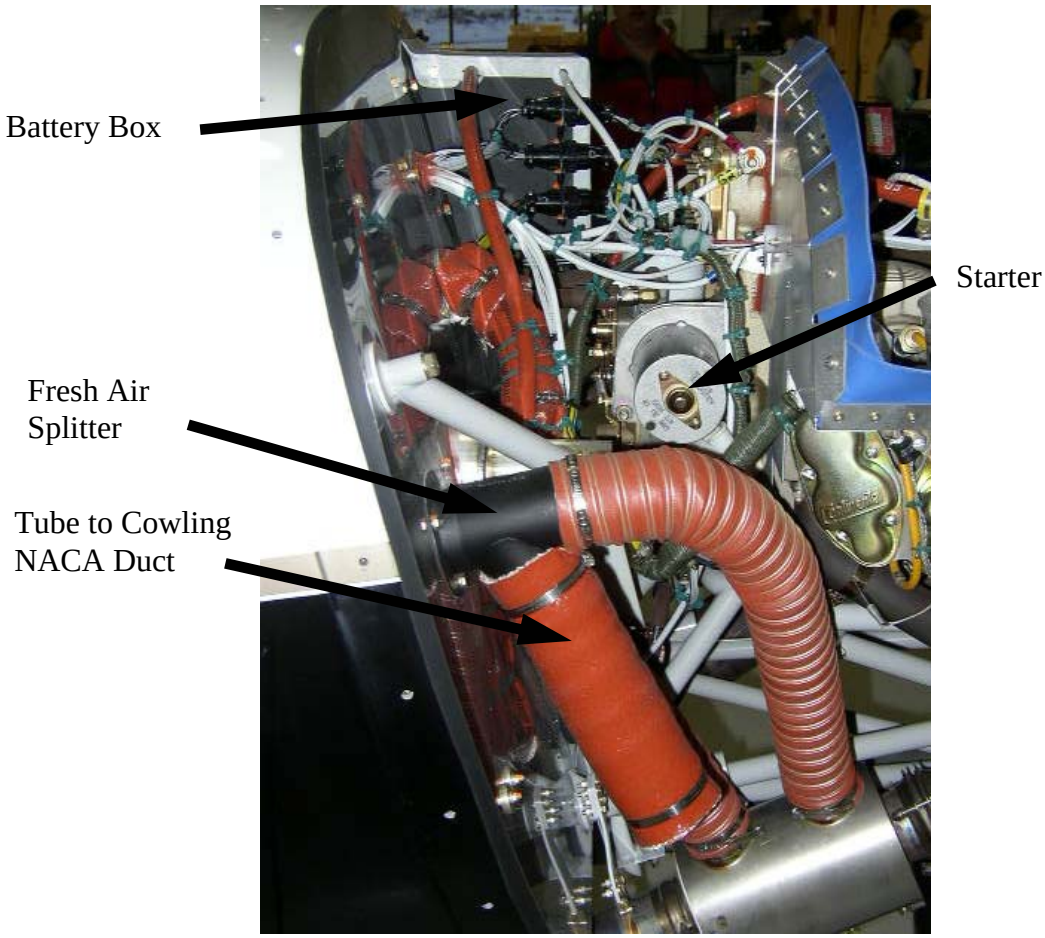

Figure 71 - 10 Engine (S/N 42023 and on)

Figure 71 - 11 Engine

a. **Engine Removal**

NOTE

Always install a new cotter pins at all control cable connections.

1. Place all cabin switches and the fuel shutoff valve in the off position.
2. Secure the tail of the aircraft using a jack stand and an undertail tiedown configuration.
3. Remove the upper and lower engine cowling per paragraph 71-5 of this chapter.
4. Remove the battery box lid and disconnect the positive and negative battery leads from the batteries. The negative lead does not need to be disconnected from the engine case.
5. Disconnect the ground strap between the engine mount and engine case at the engine mount side of the ground strap.
6. Loosen the No. 2 (left) alternator adjustment rod and slip the drive belt off the alternator pulley.
7. Remove the propeller, spinner, hub assembly, and alternator drive pulley from the engine (see Chapter 61).
8. Remove the exhaust hangers from the tailpipes by removing the steel clamps from the hanger buckle.
9. Disconnect the P1, P2, and P3 power leads from the power distribution grid.
10. Disconnect the alternator output wire from the power distribution grid.
11. Disconnect the starter power lead at the starter terminal stud.
12. Disconnect the main and return fuel lines from the firewall bulkhead fittings.
13. Disconnect the fuel injection head drain hose from the drain tube.
14. Disconnect the engine fuel pump drain hose from the drain tube.
15. Remove the manifold drain hose sniffle valve from the drain tube flange leaving hoses attached.
16. Disconnect the crankcase breather hose from the drain tube.
17. Remove the fresh air ventilation ducting hose from the fresh air manifold on the firewall.
18. Remove the inlet and outlet hot air ventilation ducting hoses from the exhaust heat exchanger.
19. Remove the throttle cable from the throttle control lever and cable attachment bracket.
20. Remove the fuel mixture cable from the engine fuel pump mixture control arm and cable attachment bracket.
21. Remove the propeller governor cable from the governor control lever and cable attachment bracket.
22. Cut the safety wire, and disconnect the warm air control cable from the induction tube butterfly valve lever arm. Pull the warm air cable out from the induction tube retaining ears.
23. Remove all clamps and tie straps used to fasten down wiring, cables, fuel lines, and hoses that will interfere with engine removal.
24. Remove the exhaust manifolds from the engine (see Chapter 78).
25. Remove four engine isolator bolts, outer isolator cushions, spacers, and heat shield (if present on the left front isolator) from the engine.
26. With the proper hoist and hoist attachment connected to the engine lifting eyes, remove the engine from the aircraft.

b. **Engine Cleaning**

Before cleaning the engine, cover or remove all electrical components subject to damage from cleaning solution saturation. Plug all openings using suitable plastic caps, rubber plugs, and covers. Use standard solvent or mineral spirits to thoroughly clean all areas of the engine. Low air pressure may be carefully used to assist cleaning and drying of the engine components.

CAUTION

Never use high pressure washers or cleaning devices to clean the engine.

c. **Engine Installation**

With the proper hoist and hoist attachment connected to the engine lifting eyes, hoist the engine into position making sure attachment bolt holes in the engine standoff legs align with the center holes in the engine isolator cushions.

1. Install four engine outer isolator cushions, spacers, heat shield (on the left front isolator only, if applicable), safety wire tabs, and bolts. Maintain 0.40 inch clearance between the heat shield and the engine frame. Slight bending and/or trimming of the heat shield is acceptable to achieve the required clearance. Torque bolts 55 to 60 ft.-lbs.
2. Install safety wire per MS33540.
3. Install the exhaust manifolds to the engine per instruction in Chapter 78.
4. Install the warm air control cable through the induction tube retaining ears and connect the cable end to the induction tube butterfly valve lever arm. Torque the cable retaining nut and bolt configuration 20 to 24 in.-lbs. Safety wire the warm air control cable housing to the induction tube retaining ears per MS33540.
5. Install the propeller governor cable to the cable attachment bracket. Torque retaining bolts 30 to 36 in.-lbs. Connect the cable end to the propeller governor control lever.
6. Install the fuel mixture control cable to the mixture control bracket. Torque retaining bolts 30 to 36 in.-lbs. Connect the end of the cable to the engine fuel pump mixture control arm lever.
7. Install the throttle control cable to the cable attachment bracket. Torque retaining bolts 30 to 36 in.-lbs. Connect the end of the cable to the throttle control lever.
8. Install the inlet and outlet hot air ventilation ducting to the exhaust heat exchanger, and tighten the steel retaining clamp.
9. Install the fresh air ventilation ducting hose to the fresh air manifold on the firewall, and tighten the steel retaining clamp.
10. Connect the crankcase breather hose to the drain tube, and tighten the steel retaining clamp.
11. Install the manifold drain hose sniffle valve to the drain tube flange. Torque the sniffle valve retaining nut 240 to 280 in.-lbs.
12. Connect the engine fuel pump drain hose to the drain tube, and tighten the steel retaining clamp.
13. Connect the fuel injection head drain hose to the drain tube, and tighten the steel retaining clamp.
14. Connect the main and return fuel lines to the firewall bulkhead fittings. Torque the main fuel line fitting 450 to 500 in.-lbs. Torque the return fuel line fitting 270 to 300 in.-lbs.
15. Connect the starter power lead to the starter terminal stud. Torque the terminal stud retaining nut 140 to 180 in.-lbs.

16. Connect the alternators output wires to the power distribution grid. Torque the retaining nut 30 to 36 in.-lbs.
17. Connect the P1, P2, and P3 power leads to the power distribution grid.
18. Connect the exhaust hangers to the tailpipes. Torque the clamp retaining nut 12 to 15 in.-lbs.
19. Install the propeller, spinner, hub assembly, and alternator drive pulley to the engine per instructions in Chapter 61.
20. Connect the ground strap between the engine and engine mount. Torque the retaining bolt on the mount side of the ground strap 30 to 36 in.lbs.
21. Install in reverse order as disassembled all clamps and tie straps used to fasten down wiring, cables, fuel lines, and hoses which will chafe, rub, or interfere with surrounding objects during engine operation. If any hoses, wires, or lines are touching the engine induction tubes, oil radiator, or engine mount apply a dab of high temperature silicone between them.
22. Connect the positive and negative battery leads to batteries. Install the battery box lid. Install the safetey wire between the battery box lid retaining bolts.
23. Install the upper and lower cowling per section 71-5 of this chapter.
24. Remove the jack stand and undertail tie down configuration from the tail of the aircraft.

71-5. Engine Cowling Removal, Installation, and Maintenance

a. **Removal of Engine Cowling**

1. Using suitable pliers or by attaching a drill/driver, remove the hinge pins from the left and right side of the cowling by pulling the pins outward through the front inlet holes. See Figure 71 – 12.

NOTE

When using a powered drill/driver select low speed and low torque.

2. Remove the screws attaching the nose gear fairing to the lower cowling and the two screws on the outboard rear nose gear attaching the cowling to the fuselage. Do not remove the screw attaching the rear center fairing to the fuselage. This screw will hold the nose fairing in place when installing the lower cowling. See Figure 71 – 13.
3. Remove the six front cowling attachment screws. See Figure 71 – 14.
4. Remove the screws attaching the lower cowling to the fuselage. An assistant will be required to help hold the lower cowling and prevent it from falling and hitting the nose wheel pant or ground. See Figure 71 – 15.
5. Remove the lower cowling from the aircraft.
6. Remove screws attaching the upper cowling to the fuselage. See Figure 71 – 15.
7. Lift up the cowling to gain access to the fresh air tube that attaches to the NACA duct at the side of the cowl. Remove clamp at the NACA duct, then remove the tube.
8. Remove the upper cowling from the aircraft.

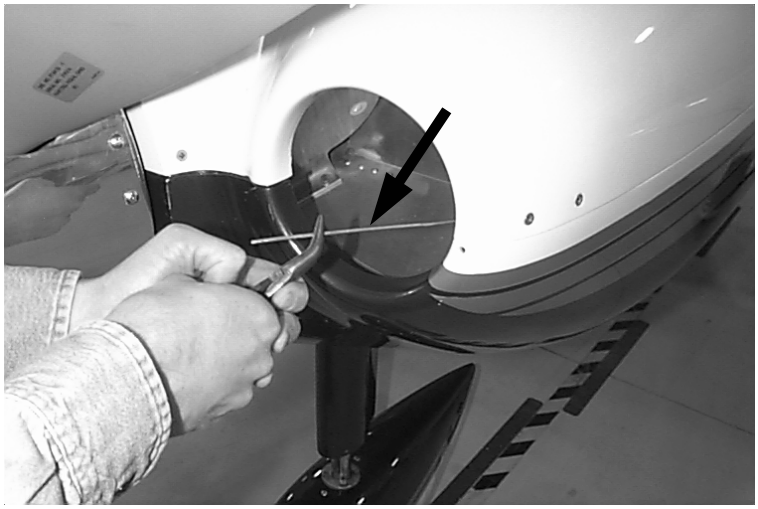

Figure 71 – 12 Removing Hinge Pins from Cowl

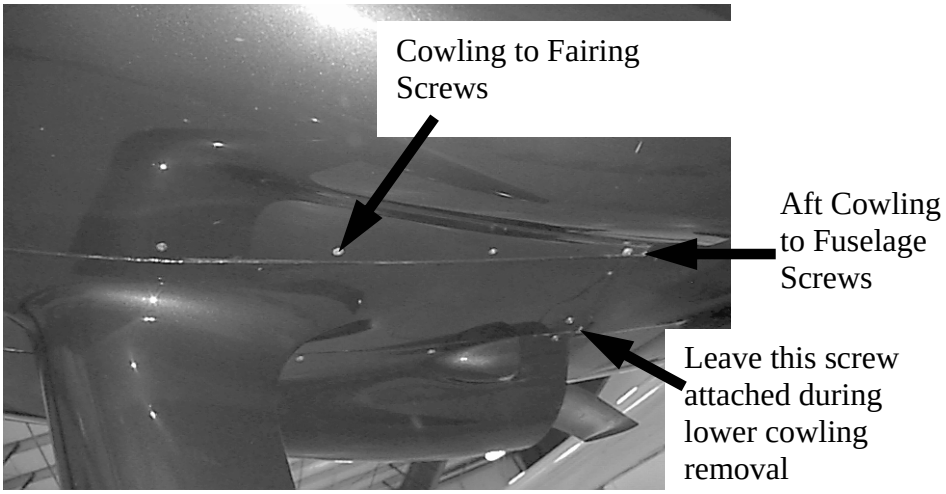

Figure 71 – 13 Cowling Screws

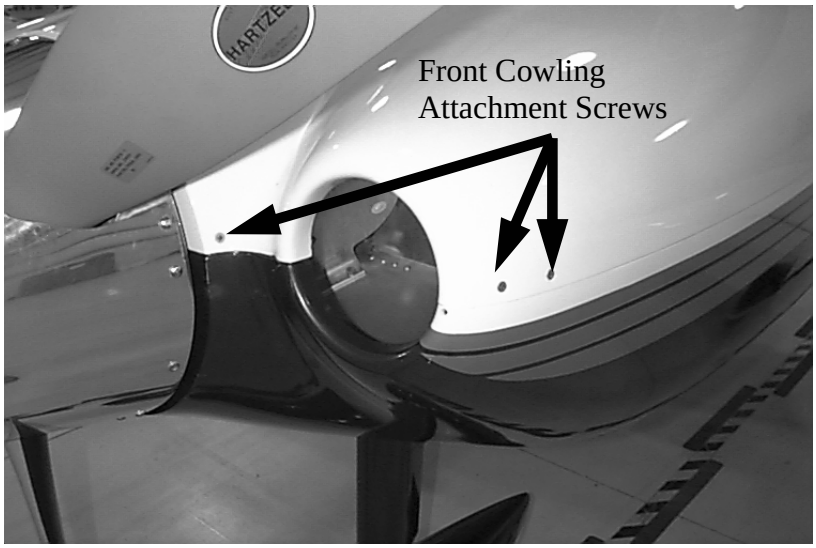

Figure 71 – 14 Front Cowling Screws

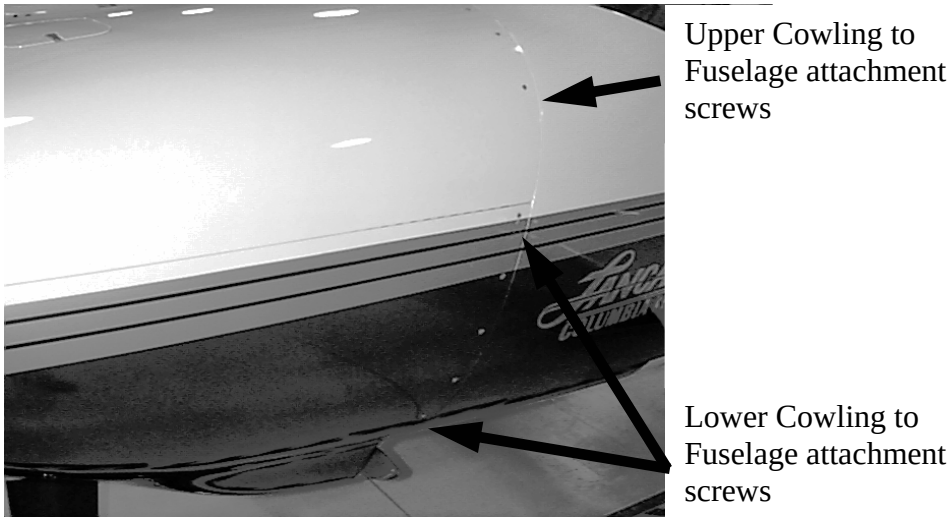

Figure 71 – 15 Screws Attaching the Cowling to the Fuselage

b. **Installation of the Cowling**

1. With an assistant, install the lower cowling onto the fuselage so that cowling attachment screw holes align with the corresponding holes in the fuselage joggle. Install the number one attachment screw (closest to the hinge) and the lowest screw that shows good alignment. Repeat for the opposite side. The screws should be loose. Do not final torque at this time. See Figure 71 – 15.
2. Position the upper cowl against the bulkhead and insert all fasteners. Do not start the threads at this time.
3. Insert the hinge pins by hand through the front pin holes, located next to the outboard areas of the air inlet holes, and into the hinge elements until they stop. The pin holes are not on all aircraft and their use is optional, if present. See Figure 71 – 16. As necessary, apply a light coating of Aeroshell 15W50 engine oil, Mil-L-7870 lubricating oil, or Dow Corning DC 7 silicone release compound to the first two inches of the hinge pin to aid installation.
4. Attach the hinge pin to a drill/driver and feed the pin into the hinge elements while simultaneously manipulating the cowl halves as required. Insert both pins completely before proceeding.

NOTE

When using a powered drill/driver select <u>low speed</u> and <u>low torque</u>.

NOTE

It is recommended to insert the copilot side hinge pin first. Hinge elements at the leading edge become obscured by engine baffling on the pilot's side.

5. Install all remaining lower cowl fasteners. Tighten all upper and lower cowl fasteners beginning with the top center an progressing circumferentially toward the nose wheel fairing. Torque screws 30 to 36 in.-lbs.
6. Align the nose fairing attachment screw holes with the corresponding holes in the lower cowling, and install the attachment screws. Torque screws 20 to 25 in.-lbs. Install two outboard rear nose gear and cowling-to-fuselage attachment screws. Torque screws 30 to 36 in.-lbs. See Figure 71 – 13.
7. Attach the fresh air tube to the NACA duct on the right side of the cowl with a clamp. Tighten clamp. See Figure 71 - 9.
8. Install the upper cowling so that the cowling attachment screw holes align with the corresponding holes in the fuselage joggle, and install the attachment screws. Torque screws 30 to 36 in.-lbs. (see Figure 71 – 15).
9. Install front six cowling attachment screws. Torque screws 30 to 36 in.-lbs. See Figure 71 – 14.

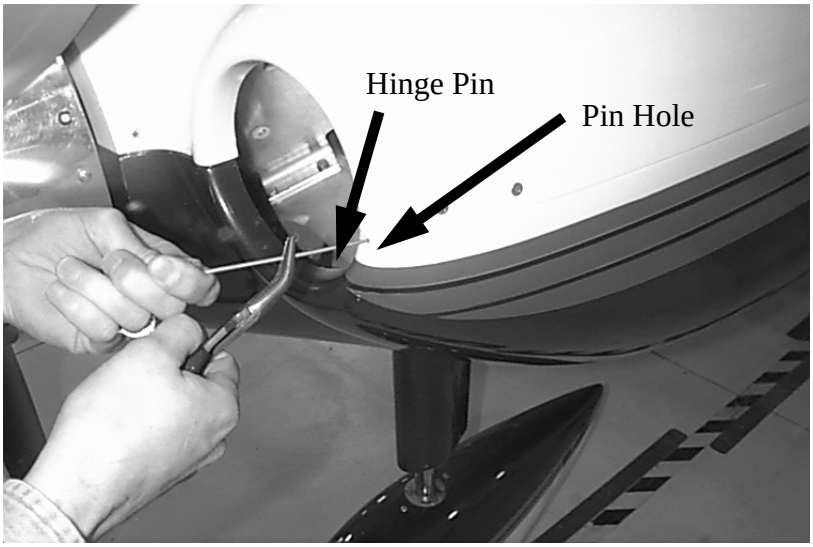

Figure 71 – 16 Installing Hinge Pins

c. **Maintenance of Cowling –** The only cowling maintenance item is to repair the fire resistant paint on the inside of the upper and lower cowl that may have been worn off due to the rubbing of the baffling on the cowl.

The fire protective paint used is Ocean Fire Retardants #477 two-part epoxy. When the paint is exposed to flame, it puffs up to form a very dense, insulating, spongy foam crust, which retards heat penetration to the underlying surface.

Use the following repair method to repaint worn areas inside the cowling:

1. Ensure the surface of the damaged area is dry and free from oil or grease.
2. Mix eight parts #477A with 1 part #477B by volume. Allow 60 minutes for the paint to activate at a room temperature of 70° F.
3. Using a brush, apply two coats of paint to the damaged area 23 to 27 mils thick. Allow 60 minutes of drying time between coats.
4. Cure times at 70° F are:
   Dry to recoat – 60 minutes
   Dry to touch – 4 hours
   Dry to handle – 24 hours
   Through cure – 7 days

71-6. Engine Mount Removal, Inspection, and installation

a. **Removal of the Engine Mount** (see Figure 71 - 17)

1. With the engine removed per paragraph 71-4 and the tail of the aircraft fully secured using a jack stand and undertail tie strap, remove the nose gear and nose gear fairing per isntructions in Chapter 32).
2. Remove all additional clamps and tie straps used to attach wiring, cables, and hoses to the engine mount.
3. Remove the bolts attaching the upper and lower engine mount to the firewall.

CAUTION

Use caution when removing the bolts from the firewall to prevent bushings and washers on the inside of the firewall from falling out and getting lost.

4. Remove the engine mount from the firewall.

b. **Inspection of the Engine Mount**

Inspect all engine mount tubes and welds using bright light and a 10X (ten power) magnifying glass for signs of cracking, chafing, rusting, and pitting. Do not weld any part of the engine mount. If a crack is found, replacement of the engine mount is required. Small undamaged areas where corrosion protection has chafed off may be protected by removing all visible surface rust with a wire brush and applying a quality high temperature rust preventative paint.

c. **Installation of the Engine Mount**

1. Position the engine mount onto the firewall with the upper mounting pad tubes protruding through the firewall and into the backside bushings.
2. Install four engine mount bolts and washers through the front of the engine mounting pad tubes. To ensure the bolts do not “shank out”, one additional AN960-716 washer (for a total of two washers) may be used under the bolt head to attain a minimum protrusion of two threads and a maximum protrusion of 3.5 threads through the nut after torquing. Thread protusion includes the chamfer at the end of the bolt. Make sure the upper engine mount firewall bushings do not fall out while inserting the bolts. When correctly installed, bushings slide inside of the engine mounting pad tubes.
3. Install engine mount washers and nuts at the backside of the firewall. One washer at each lower bolt and two washers at each upper bolt. Make sure that the ground straps are installed between the two washers at the upper mounting bolt locations.
4. Torque the engine mount bolts 55 to 60 ft.-lbs.
5. Install in reverse order as disassembled all clamps and tie straps used to fasten down wiring, cables, and hoses which will chafe, rub, or interfere with surounding objects during engine operation.

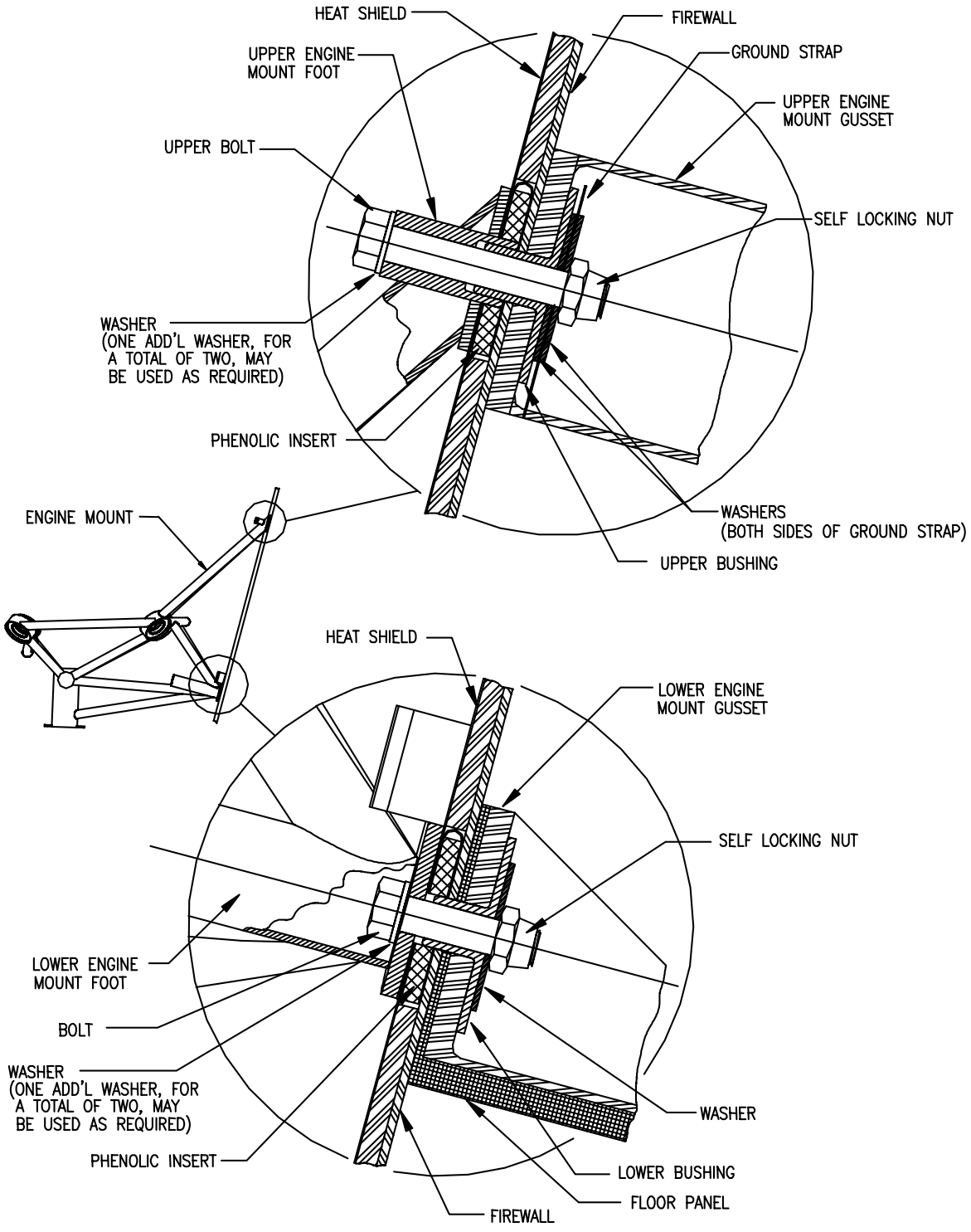

Figure 71 - 17 Engine Mount to Fuselage Attachment

71-7. Engine Air Filter Removal and Replacement (see Figure 71 - 18)

The engine air filter element is a disposable unit that must be replaced annually or every 100 hours. The filter element must also be replaced if it's face is covered by dust or debris by 50 % or greater. The Air filter frame and grill assembly are Bracket Aircraft Company, Inc. Part No.BA-9010. Air filter element is Brackett Aircraft Company, Inc. Part No. BA-9005.

a. Remove the upper cowling from the aircraft per instructions in paragraph 71-5.
b. Loosen the two large ducting clamps connecting the air filter housing to the induction tube.
c. Remove two bolts and washers connecting the air filter frame to two brace brackets, respectively.
d. Remove the bolt and washer connecting the air filter frame to the loop clamp around the warm air induction hose.
e. Remove the air filter housing from the engine.
f. Remove four screws securing the grill to the air filter frame.
g. Remove the air filter element and discard.
h. Wipe down the inside of the air filter frame to remove any dirt, debris, or excess filter wettant. WD-40 or similar solvent may be used to aid in removal of excess filter wettant.
i. Install the new filter element in reverse order as disassembled. Hand tighten screws securing air filter frame grill snug.
j. Ensure that clearance between the face of the filter frame and the alternator is at least 1/2 inch. Torque two filter frame to brace bracket bolts 30 to 36 in.-lbs.
k. Torque filter frame to loop clamp bolt 30 to 36 in.-lbs.

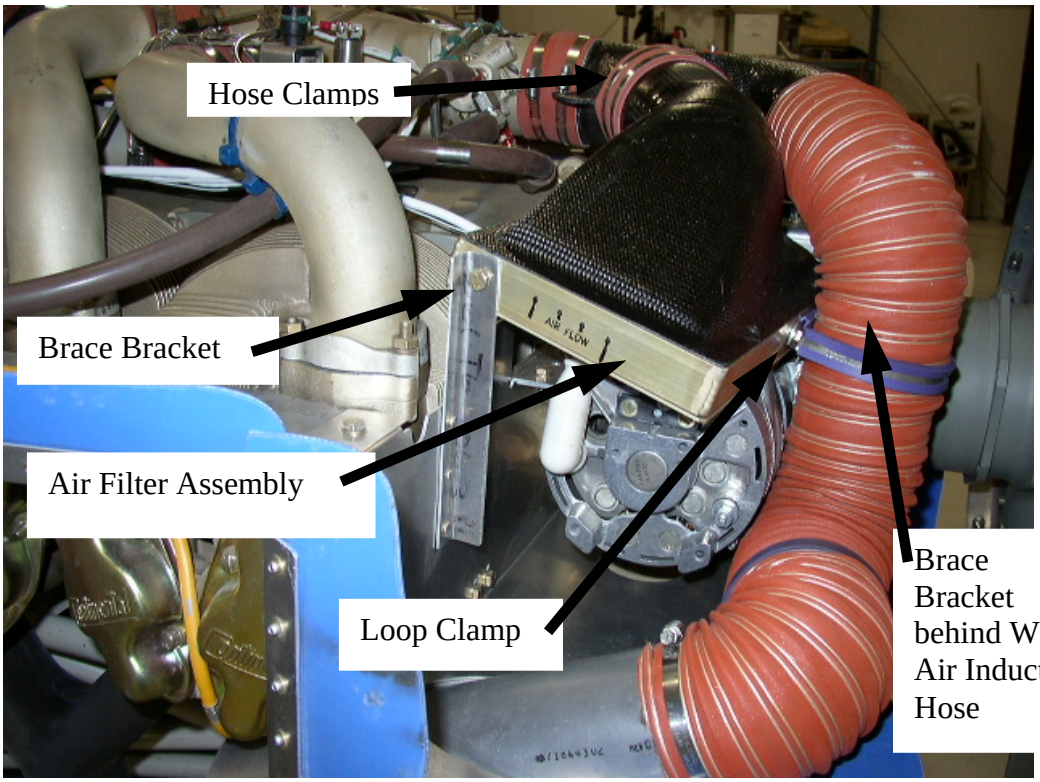

Figure 71 - 18 Engine Air Filter

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71-8. Engine Fuel Drain Tree Removal, Inspection, and Installation

a. **Engine Fuel Drain Tree Removal** (see Figure 71 - 19)

1. Remove the upper cowling from the aircraft per instructions in paragraph 71-5.
2. Using suitable wrenches, remove the fuel drain tree hard line fittings from the left and right side of the engine. Use caution not to back out fittings in the engine cylinders during removal.
3. Using suitable wrenches, remove the left and right side drain hoses from the sniffle valve fitting (see Figure 71 - 20).
4. Remove the engine oil pan retaining screws at the fuel line hose clamp locations, and remove hose clamps.
5. Detach the fuel line retaining clamps at the aft end of the engine mount, and remove the hose clamps.
6. Remove all tie straps that will prevent the fuel drain tree hoses and hard line from being removed.
7. Remove the fuel drain tree hoses and hard lines.
8. Using suitable wrenches, remove the fuel drain hoses from the fuel drain tree hard lines.
9. Remove the sniffle valve to drain tube flange retaining nut and washer, and remove the sniffle valve (see Figure 71 - 20).

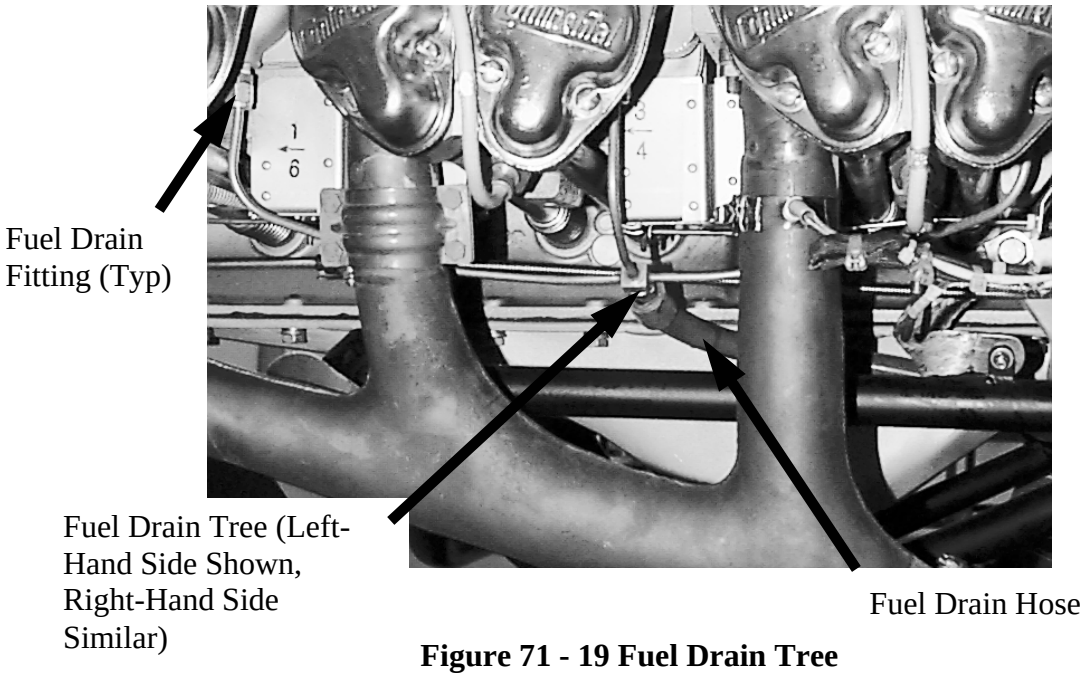

Figure 71 - 19 Fuel Drain Tree

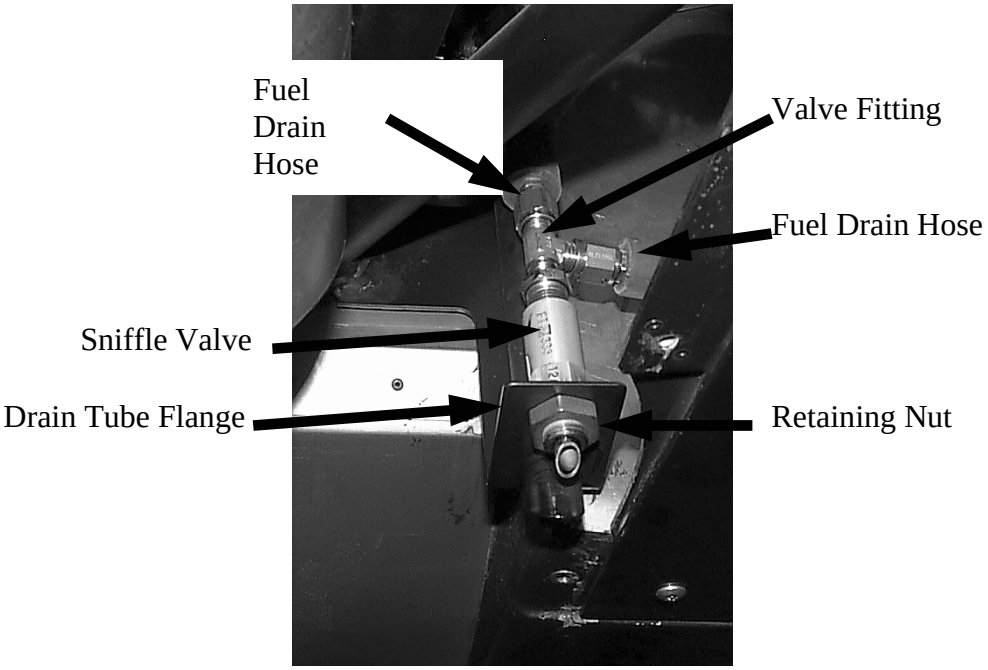

Figure 71 - 20 Fuel Drain Hoses and Valves

b. **Inspection of Engine Fuel Drain Tree**

Inspect all fuel drain tree hoses for burning, cracking, deterioration, and chafing. Inspect all fuel drain tree hose fitting ends for cracking and stripped or cross-threaded threads. Inspect fuel drain tree hard lines for vibration cracks, crushed tubes, and bending deformities. Inspect fuel drain tree hard line fitting ends for stripped or cross-threaded threads.

If in doubt about a possible crack, use a 10X (ten power) magnifying glass to thoroughly inspect the suspicious area.

Inspect the sniffle valve for a jammed or blocked internal check ball by blowing through the inlet side of the valve. Air should freely pass the check ball and exit the outlet side of the valve. Blowing air through the outlet side of the valve or applying vacuum to the inlet side of the valve should seat the check ball and prevent air from passing through the valve.

Replace all defective parts found in any of the fuel drain tree hoses, hard lines, or sniffle valve components.

c. **Installation of Engine Fuel Drain Tree**

1. Install the sniffle valve into the drain tube flange, and install the retaining nut. Torque the retaining nut 240 to 280 in.-lbs. (see Figure 71 - 20).
2. Using suitable wrenches, attach the drain hoses to the fuel drain tree hard line fittings. Torque fittings 135 to 150 in.-lbs. (see Figure 71 - 20).
3. Connect the fuel drain tree hard line fittings to the engine cylinder heads. Use caution not to turn the fittings in the engine cylinders during installation (see Figure 71 - 19).
4. Route the drain hose in reverse order as disassembled. Torque fitting 135 to 150 in.-lbs.
5. Connect the fuel drain hose fittings to the sniffle valve fitting ends. Torque the hose fittings 135 to 150 in.-lbs.

6. Install the fuel drain hose retaining clamps over the fuel drain hoses at the oil pan attachment screw locations. Attach the hose clamps to the oil pan using the oil pan retaining screws. Torque the oil pan retaining screws 155 to 175 in.-lbs.
7. Install the fuel drain hose retaining clamps over the fuel drain hoses at the aft engine mount locations using hose clamp bolts, nuts, and washers in reverse order as disassembled. Torque hose clamp retaining bolts 30 to 36 in.-lbs.
8. Install new tie straps at all locations noted prior to removing the fuel drain tree hoses and hard lines.

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71-9. Engine Baffling

a. The engine baffling directs airflow over the engine for proper cooling and consists of thin aluminum with silicone sheet seals. Baffling should be checked at each inspection for cracks, excessive gaps, and security of mounting to engine.

During routine inspections, the gap between the forward left and right side baffling and cowling skin just below the air inlet locations should be checked. A minimum of 0.30 in. must exist between the edge of the aluminum baffling and the lower cowling skin at these locations. If adjustment is necessary, pull back the baffling seal, and trim away the baffling material as needed. Leave no less than 3/16 in. of material between the center of the rivets used to attach the baffling seal and the edge of the baffling (see Figure 71 - 21).

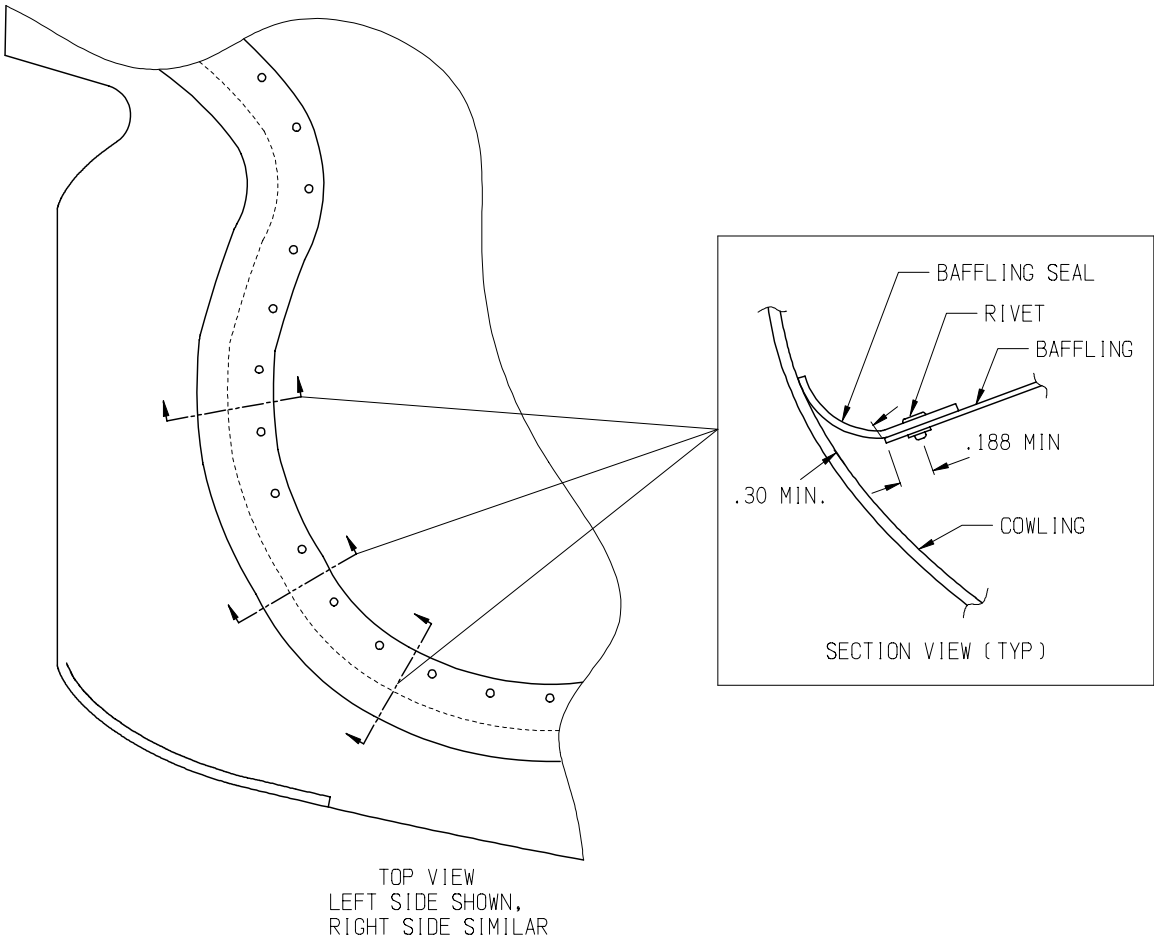

Figure 71 - 21 Baffling at Cowling Below Air Inlet

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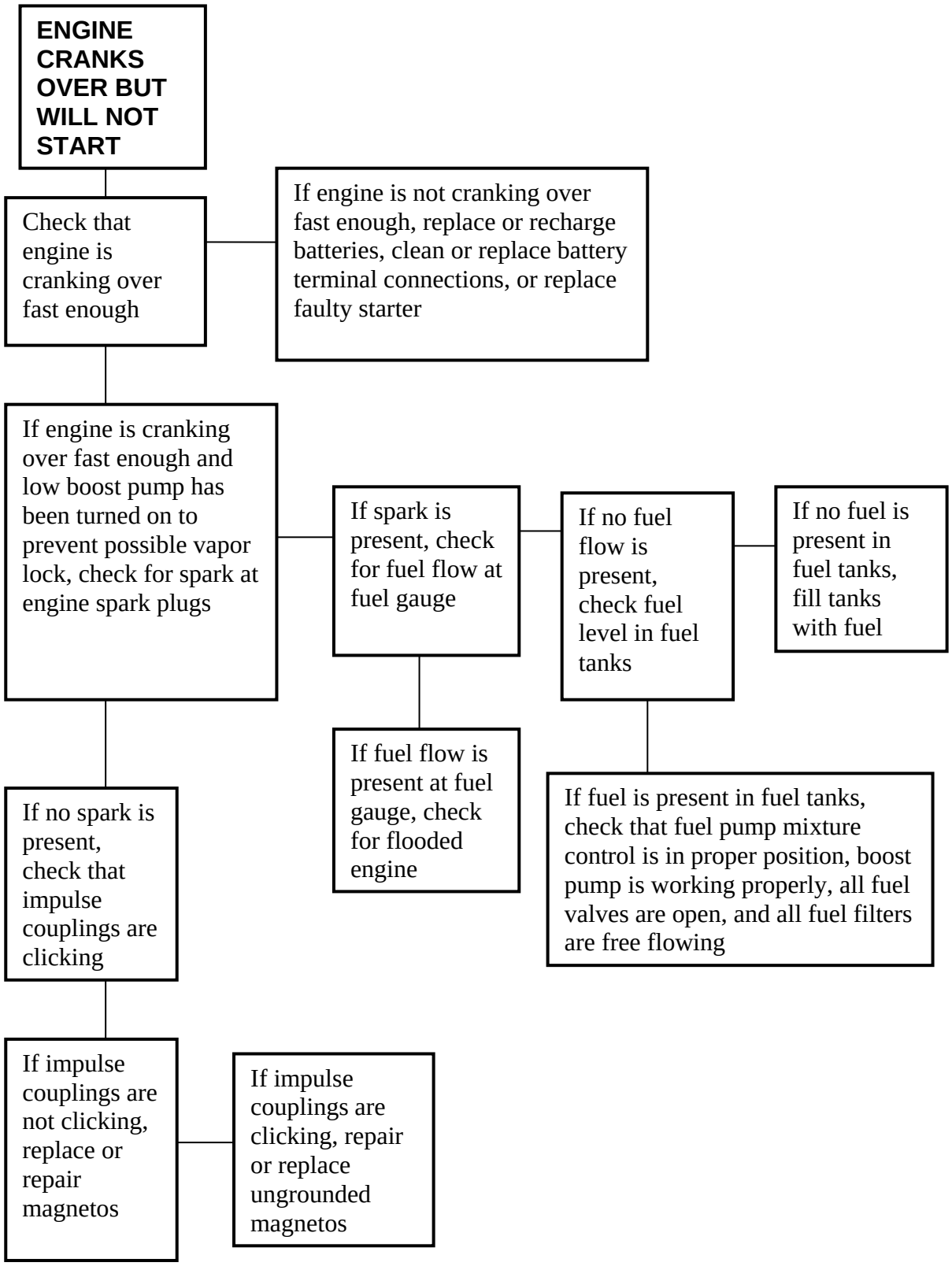
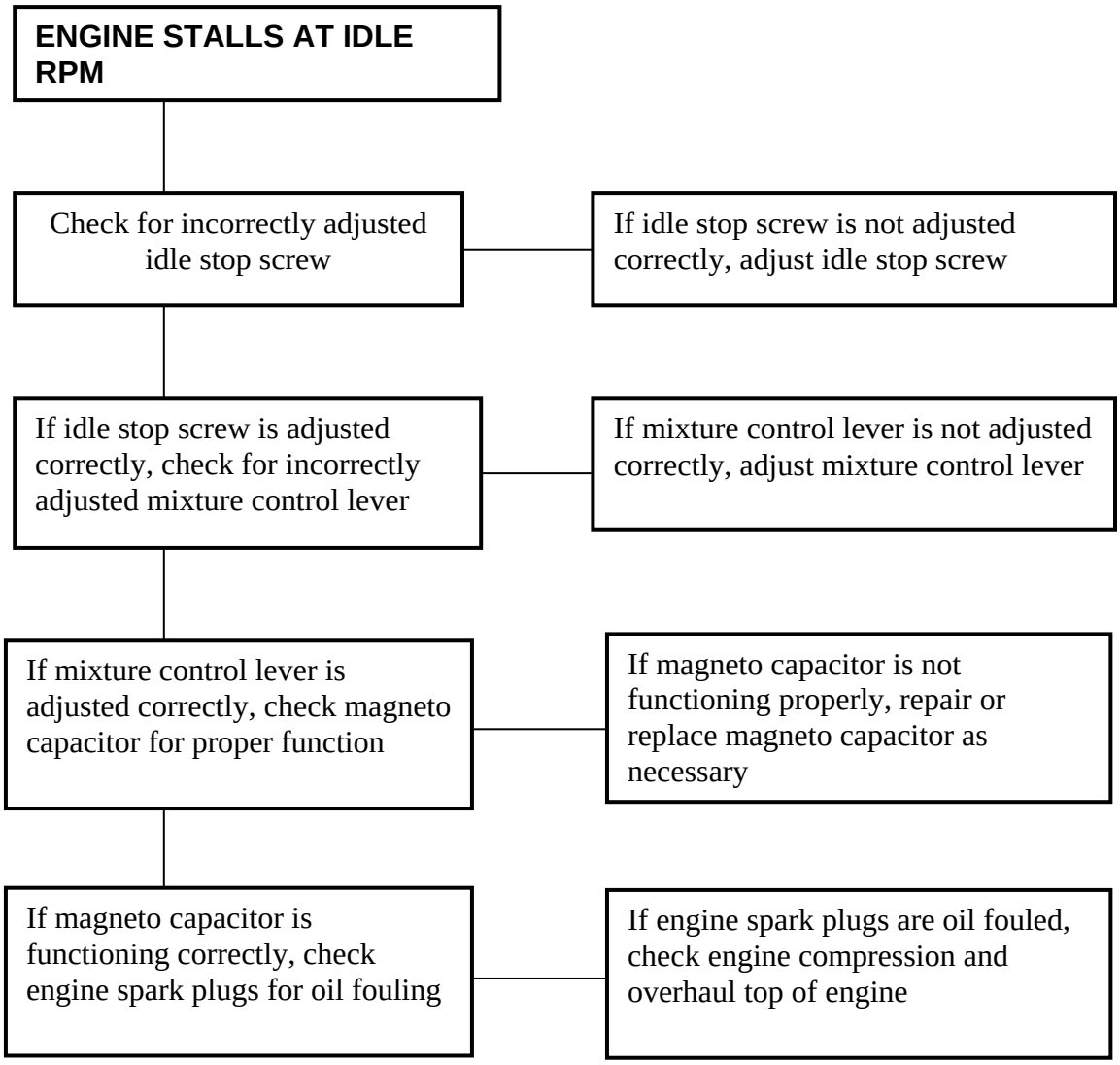
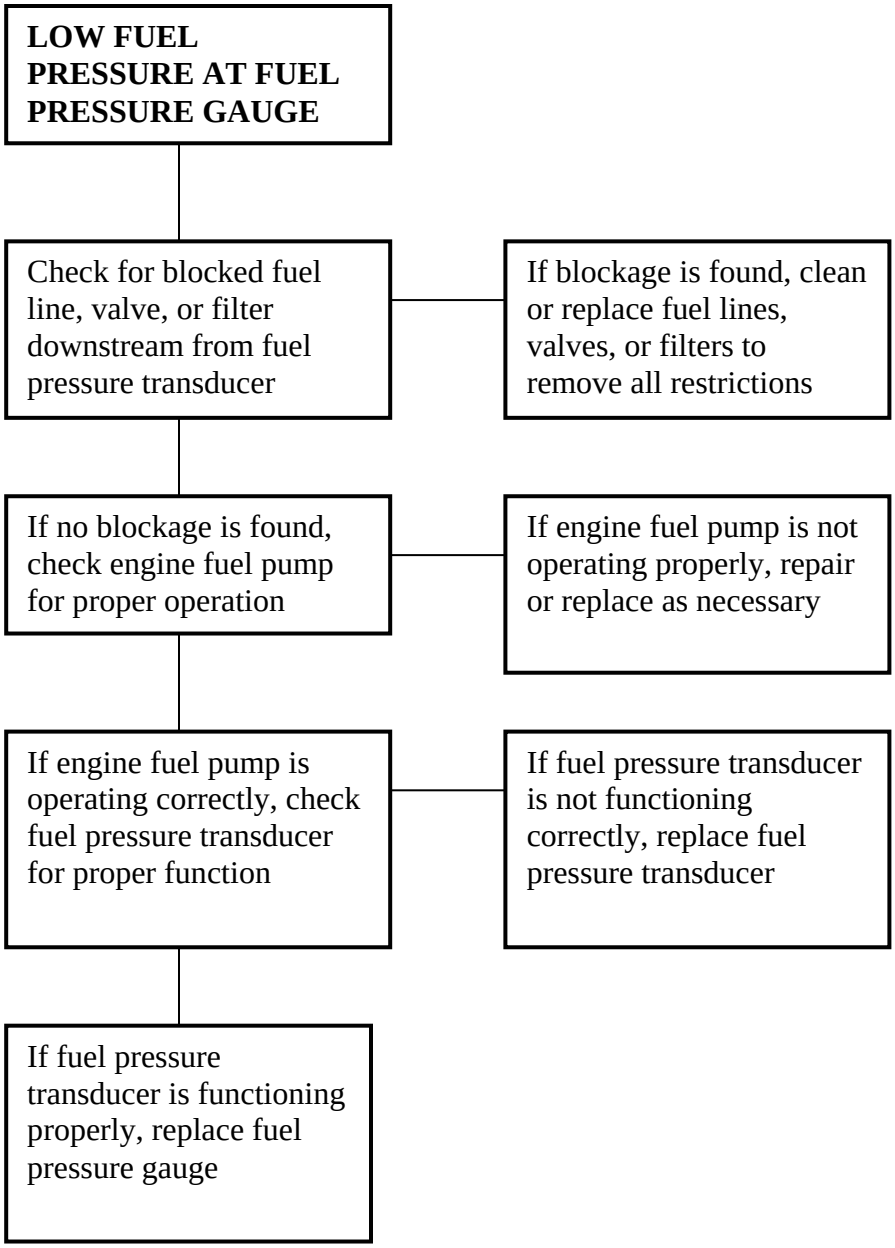
71-10. Engine Troubleshooting

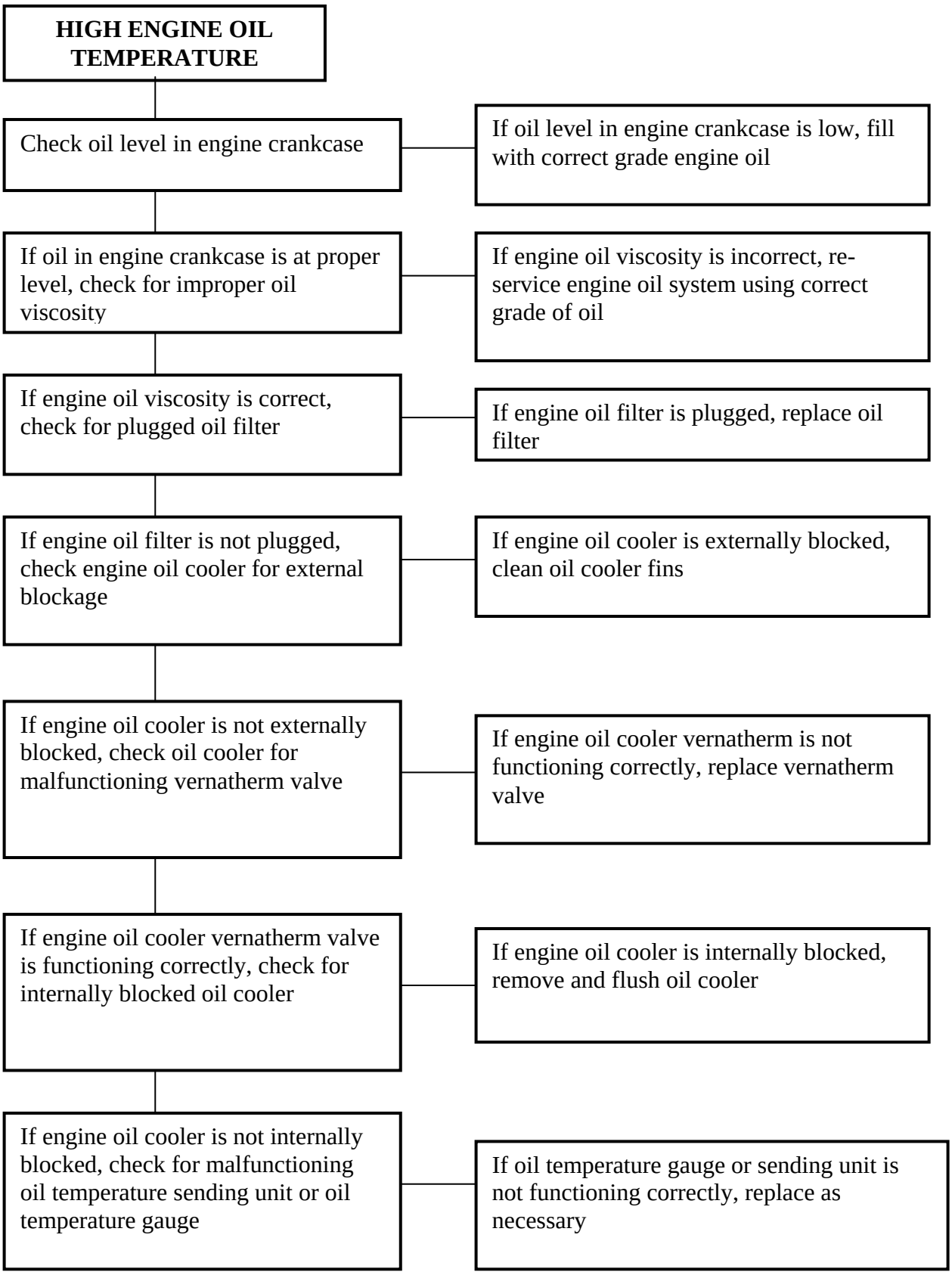
The following diagnostic tests are basic guidelines to follow when diagnosing engine starting and running problems. More detailed information on the engine's internal components can be found in the Teledyne Continental Motors Permold Series Maintenance Manual FORM NO. X30634A or Overhaul Manual FORM NO. X30568A.

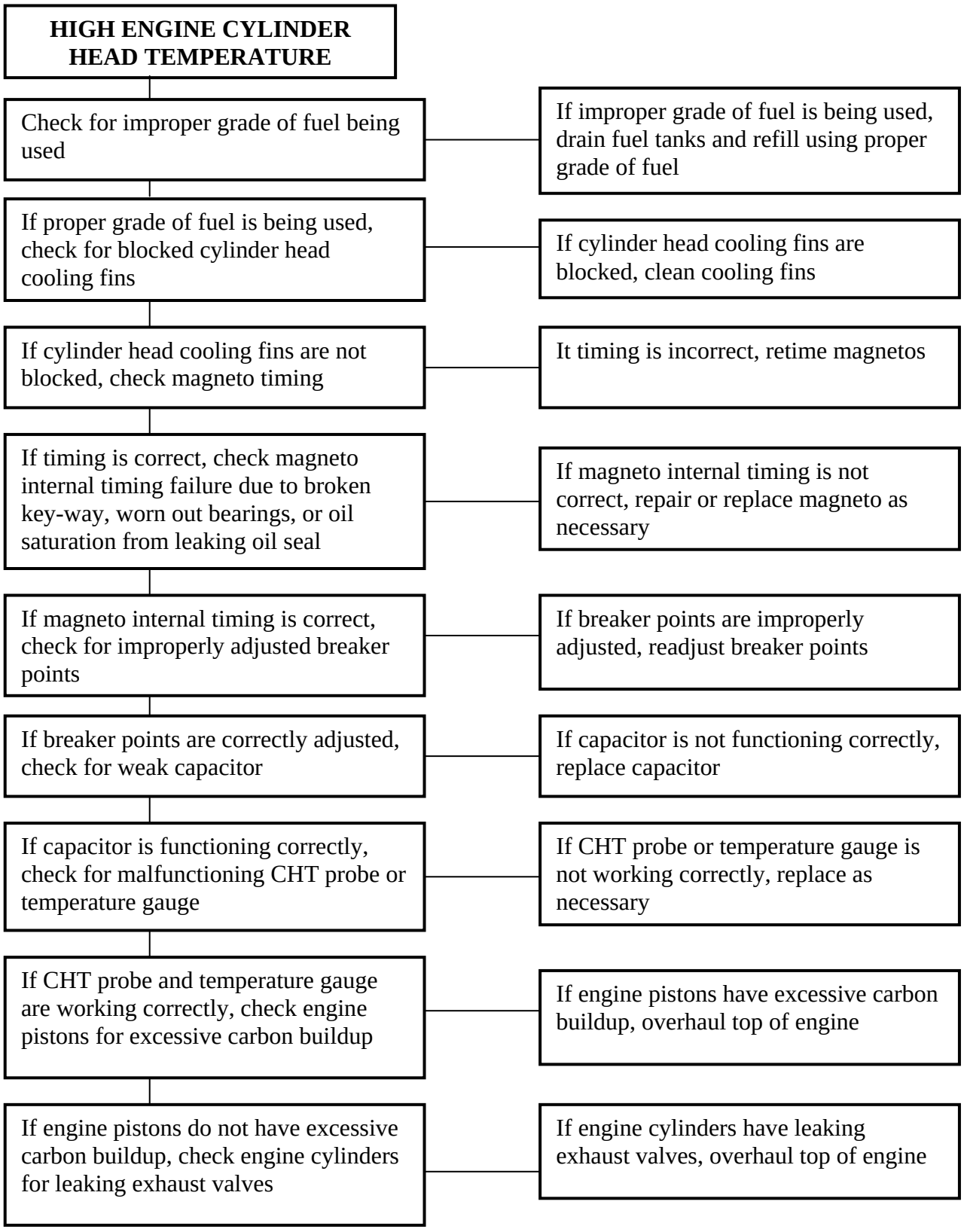

ENGINE CRANKS OVER BUT WILL NOT START
Check that engine is cranking over fast enough
If engine is not cranking over fast enough, replace or recharge batteries, clean or replace battery terminal connections, or replace faulty starter
If engine is cranking over fast enough and low boost pump has been turned on to prevent possible vapor lock, check for spark at engine spark plugs
If spark is present, check for fuel flow at fuel gauge
If no fuel flow is present, check fuel level in fuel tanks
If no fuel is present in fuel tanks, fill tanks with fuel
If fuel flow is present at fuel gauge, check for flooded engine
If fuel is present in fuel tanks, check that fuel pump mixture control is in proper position, boost pump is working properly, all fuel valves are open, and all fuel filters are free flowing
If no spark is present, check that impulse couplings are clicking
If impulse couplings are not clicking, replace or repair magnetos
If impulse couplings are clicking, repair or replace ungrounded magnetos


ENGINE STALLS AT IDLE RPM
Check for incorrectly adjusted idle stop screw
If idle stop screw is not adjusted correctly, adjust idle stop screw
If idle stop screw is adjusted correctly, check for incorrectly adjusted mixture control lever
If mixture control lever is not adjusted correctly, adjust mixture control lever
If mixture control lever is adjusted correctly, check magneto capacitor for proper function
If magneto capacitor is not functioning properly, repair or replace magneto capacitor as necessary
If magneto capacitor is functioning correctly, check engine spark plugs for oil fouling
If engine spark plugs are oil fouled, check engine compression and overhaul top of engine

ENGINE MISFIRES DURING HIGH RPM
Check for blocked fuel line, valve, or filter
If blockage is found, clean or replace fuel lines, valves, and filters to remove all restrictions
If no blockage is found, check engine fuel pump and pump shaft operation
If engine fuel pump operation is inadequate, repair or replace fuel pump or pump shaft
If engine fuel pump and pump shaft operation is adequate, check magneto timing
If timing is incorrect, retime magnetos
If timing is correct, check magneto internal timing failure due to broken key-way, worn out bearings, or oil saturation from leaking oil seal
If magneto internal timing is not correct, repair or replace magneto as necessary
If magneto internal timing is correct, check for improperly adjusted breaker points
If breaker points are improperly adjusted, readjust breaker points
If breaker points are correctly adjusted, check for weak capacitor
If capacitor is not functioning correctly, replace capacitor
If capacitor is functioning correctly, check engine for broken valve spring
If broken valve spring is found, replace valve spring
If no broken valve springs are found, check for burned or sticking valve
If sticking or burnt valve is found, overhaul top end of engine
If no sticking or burnt valve is found, replace worn or sticking hydraulic tappet or worn camshaft


LOW FUEL PRESSURE AT FUEL PRESSURE GAUGE
Check for blocked fuel line, valve, or filter downstream from fuel pressure transducer
If blockage is found, clean or replace fuel lines, valves, or filters to remove all restrictions
If no blockage is found, check engine fuel pump for proper operation
If engine fuel pump is not operating properly, repair or replace as necessary
If engine fuel pump is operating correctly, check fuel pressure transducer for proper function
If fuel pressure transducer is not functioning correctly, replace fuel pressure transducer
If fuel pressure transducer is functioning properly, replace fuel pressure gauge


HIGH ENGINE OIL TEMPERATURE
Check oil level in engine crankcase
If oil level in engine crankcase is low, fill with correct grade engine oil
If oil in engine crankcase is at proper level, check for improper oil viscosity
If engine oil viscosity is incorrect, re-service engine oil system using correct grade of oil
If engine oil viscosity is correct, check for plugged oil filter
If engine oil filter is plugged, replace oil filter
If engine oil filter is not plugged, check engine oil cooler for external blockage
If engine oil cooler is externally blocked, clean oil cooler fins
If engine oil cooler is not externally blocked, check oil cooler for malfunctioning vernatherm valve
If engine oil cooler vernatherm is not functioning correctly, replace vernatherm valve
If engine oil cooler vernatherm valve is functioning correctly, check for internally blocked oil cooler
If engine oil cooler is internally blocked, remove and flush oil cooler
If engine oil cooler is not internally blocked, check for malfunctioning oil temperature sending unit or oil temperature gauge
If oil temperature gauge or sending unit is not functioning correctly, replace as necessary


HIGH ENGINE CYLINDER HEAD TEMPERATURE
Check for improper grade of fuel being used
If improper grade of fuel is being used, drain fuel tanks and refill using proper grade of fuel
If proper grade of fuel is being used, check for blocked cylinder head cooling fins
If cylinder head cooling fins are blocked, clean cooling fins
If cylinder head cooling fins are not blocked, check magneto timing
It timing is incorrect, retime magnetos
If timing is correct, check magneto internal timing failure due to broken key-way, worn out bearings, or oil saturation from leaking oil seal
If magneto internal timing is not correct, repair or replace magneto as necessary
If magneto internal timing is correct, check for improperly adjusted breaker points
If breaker points are improperly adjusted, readjust breaker points
If breaker points are correctly adjusted, check for weak capacitor
If capacitor is not functioning correctly, replace capacitor
If capacitor is functioning correctly, check for malfunctioning CHT probe or temperature gauge
If CHT probe or temperature gauge is not working correctly, replace as necessary
If CHT probe and temperature gauge are working correctly, check engine pistons for excessive carbon buildup
If engine pistons have excessive carbon buildup, overhaul top of engine
If engine pistons do not have excessive carbon buildup, check engine cylinders for leaking exhaust valves
If engine cylinders have leaking exhaust valves, overhaul top of engine

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CHAPTER 73

ENGINE FUEL SYSTEM

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General

NOTE: See Teledyne Continental Motors Permold Series Maintenance Manual Model IO-550-N Form Number X30634A for specific maintenance information. Contact Teledyne Continental Motors to ensure you have the latest revision to their manual. It is the responsibility of the owner or operator of the airplane to maintain the revision schedule for all manuals related to the Teledyne Continental Motors Permold Series Model IO-550-N.

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CHAPTER 74

IGNITION

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Chapter 74

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74-2	Ignition Switch	74-30-00 / Page 1

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74-1. GENERAL

See the Teledyne Continental Motors Permold Series Maintenance Manual Model IO-550-N Form Number X30634A for specific maintenance information of the ignition system. Contact Teledyne Continental Motors to ensure you have the latest revision to their manual. It is the responsibility of the owner or operator of the airplane to maintain the revision schedule for all manuals related to the Teledyne Continental Motors Permold Series Model IO-550-N.

74-2. IGNITION SWITCH

a. **Description –** The ignition switch is located on the lower left side of the instrument panel and is a standard aircraft switch that provides on and off electrical control of the magnetos and includes the start position. The switch is key-operated and mounts from the rear of the master switch panel.

b. **Removal and Installation**

1. Remove the left knee bolster panel per instructions in Chapter 25 to gain access to the rear of the master switch panel.
2. Remove the metal collar around the front side of the switch, and pull the switch out through the rear of the panel.
3. Remove and label the seven wires on the back of the switch.
4. Installation is a direct reversal of removal. Install the wires as labeled.

c. **Troubleshooting**

1. If there is no magneto drop or no shutoff, verify the ground wire at the back of the switch is intact.
2. If the starter will not engage, verify that there is power to the switch on the “B” terminal (marked on the back of the switch).

CAUTION

With the ignition switch removed or disconnected, the magnetos will be “hot”. With impulse couplings on both magnetos, personal injury may result from turning the propeller with the spark plug wires loose. Additionally, the engine may “catch” if the spark plugs are installed.

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CHAPTER
76

ENGINE CONTROLS

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Chapter 76

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76-4	Fuel Mixture	76-20-00 / Page 1
76-5	Induction Heat Cable	76-20-00 / Page 8

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76-1 GENERAL DESCRIPTION

This document discusses the description, operation, removal, installation, adjustment, testing, maintenance, and inspection of the IO-550-N engine fuel mixture and throttle control systems.

NOTE

Any cosmetic shrink wrap exterior coating on control cables is not required; it may be removed if it is damaged.

NOTE

If any hoses, wires, or lines are touching the engine induction tubes,oil radiator, or engine mount apply a dab of high temperature silicone between them.

76-2 TOOLS & EQUIPMENT

The mechanic should be equipped with a complete set of the necessary tools that include the following:

a. Wrenches – ¼ in. through 1 ¼ in.
b. Common and Phillips Head Screwdrivers
c. Pliers – Common, Diagonal Cutters, Needle Nose, Duck Bill, Snap Ring, Safety Wire
d. Ratchets – ¼ in., $^{3}/_{8}$ in., ½ in. drive
e. Sockets – ¼ in. drive $^{5}/_{32}$ in. through ½ in., $^{3}/_{8}$ in. drive $^{3}/_{8}$ in. through 1 in., ½ in. drive $^{7}/_{16}$ in. through 1 ¼ in.
f. Sockets (Deepwell) –$^{1}/_{2}$ in. drive, $^{7}/_{16}$ in. through 1 in.
g. Feeler Gauges
h. Leather or Soft Plastic Mallet
i. Torque Wrenches* 0-500 in.-lbs. and 0-100 ft.-lbs.
j. Micrometers*
k. Slide Hammer
l. Pullers
m. Thickness Gauges
n. Vernier Calipers*
o. Small Hole Gauges

* Must be currently calibrated, and the calibration must be traceable to the National Bureau of Standards.

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76-3 THROTTLE DESCRIPTION, OPERATION, REMOVAL, AND INSTALLATION

A throttle plate located inside the intake manifold controls combustion air during engine operation. A linear motion push-pull control cable is directly connected to the throttle plate lever arm (see Figure 76 - 1 or Figure 76 - 2). During engine operation, the pilot adjusts the incoming air quantity by either pulling out or pushing in on the control cable. One of two types of locking mechanism (friction or vernier) is built into the push-pull cable to prevent the cable from moving after the desired position has been set. With the friction mechanism, rotating the thumbwheel clockwise increases the force required to move the throttle (increases friction). Rotating the thumbwheel counterclockwise decreases the force required to move the throttle (decreases friction).

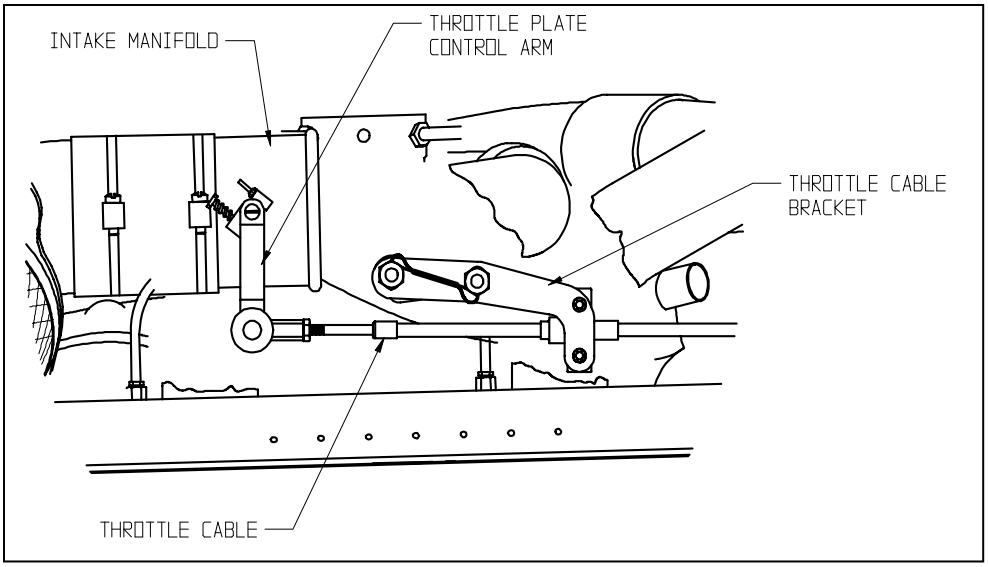


Figure 76 - 1 Throttle Cable Assembly (S/N 42001 to 42500)

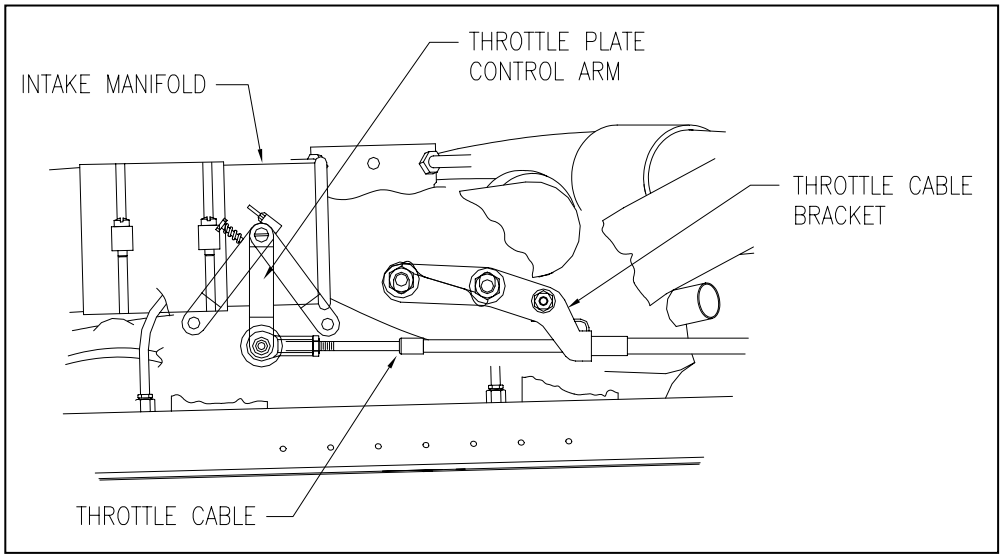


Figure 76 - 2 Throttle Cable Assembly (S/N 42501 and on)

a. **Throttle Cable Removal**

1. Remove the throttle control cable from the throttle plate control arm. Remove the bolt, nut, washers, and cotter pin (see Figure 76 - 3).

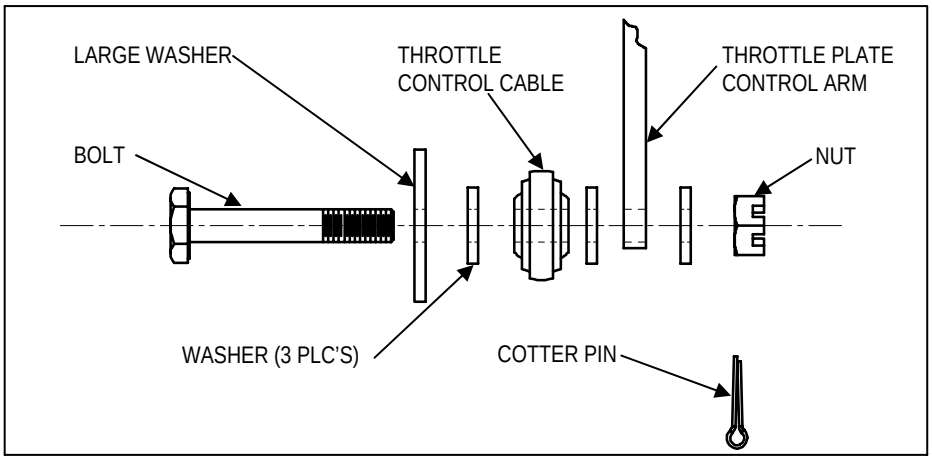


Figure 76 - 3 Throttle Cable

2. Remove the throttle control cable housing from the throttle cable bracket. For S/N 42001 to 42500 remove the bolts, nuts, clamp, spacer, and washers (see Figure 76 - 4). For S/N 42501 and on remove the bolt, washer, self locking nut, and retaining tang (see Figure 76 - 5).

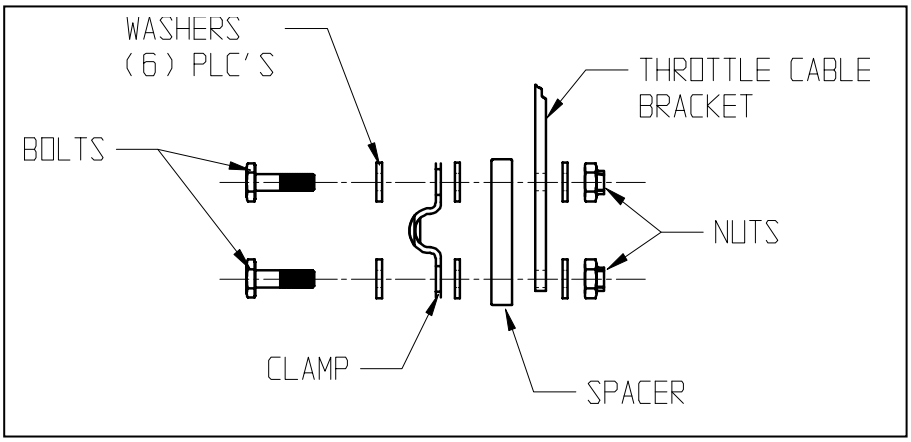


Figure 76 - 4 Throttle Cable Attachment to Bracket (S/N 42001 to 42500)

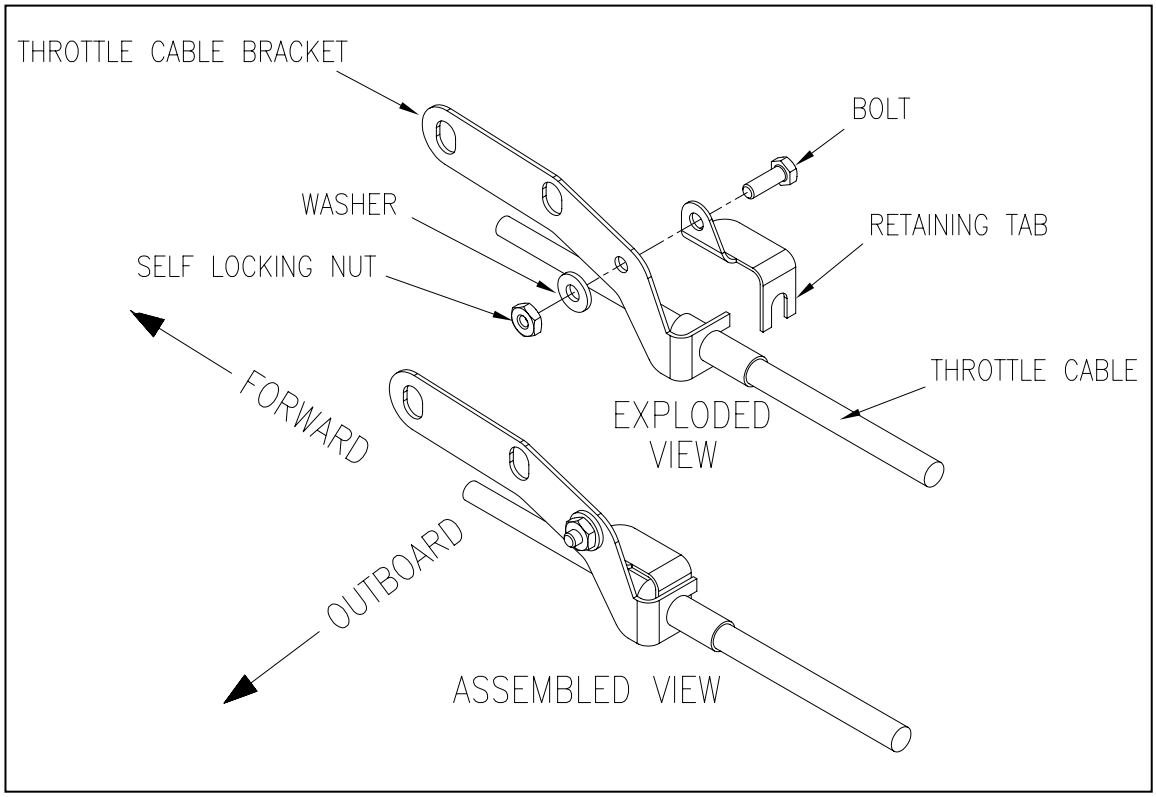
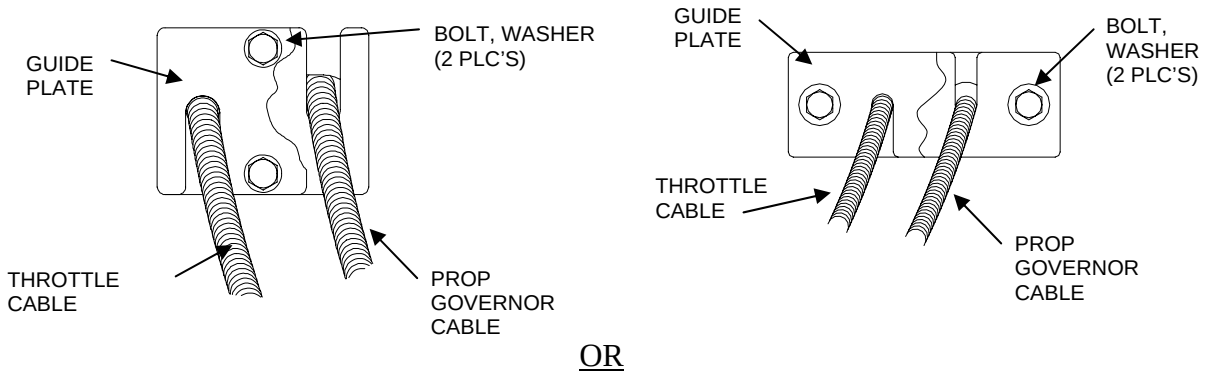


Figure 76 - 5 Throttle Cable Attachment to Bracket (S/N 42501 and on)

3. Remove the cable guide plates from the firewall heat shield. Remove the screws, guide plates, and washers (see Figure 76 - 6).



OR

Figure 76 - 6 Throttle Cable Guide Plate

4. With the GPS and cover plate removed, loosen and remove two or four panel screws, depending on the cover plate installed. Pull the panel out from the instrument console until the throttle control cable backing nut can be accessed. Loosen the backing nut, and

gently pull the throttle control cable out of the panel (see Figure 76 - 12, Figure 76 - 13, Figure 76 - 14, Figure 76 - 15, or Figure 76 - 16).

b. **Throttle Cable Installation**

1. Install the throttle control cable through the panel cover. Slide the backing nut and washer over the cable housing and torque the backing nut 240 to 280 in.-lbs.
2. Install the panel cover assembly into the instrument console, and install two or four panel screws, depending on cover plate installed. Torque the panel screws 12 to 15 in.-lbs. (see Figure 76 - 12, Figure 76 - 13, Figure 76 - 14, Figure 76 - 15, or Figure 76 - 16).
3. Route the cable through the firewall bulkhead. Install the GPS and cover plates in reverse order as disassembled.
4. Install the throttle control cable to the throttle plate control arm. Install the bolt, washers, new cotter pin, and nut in reverse order as disassembled. Torque nut 30 to 36 in.-lbs. (see Figure 76 - 3).
5. Install the throttle cable housing to the throttle cable bracket. For S/N 42001 to 42500 install the bolts, washers, clamp, spacer, and nuts in reverse order as disassembled. For S/N 42501 and on insert the retaining tab tangs into the cable retaining slot with tangs on the forward side of the bracket; then install the bolt, washer and self locking nut. Torque nut(s) 30 to 36 in.-lbs. (see Figure 76 - 4 or Figure 76 - 5).
6. Install the cable guide plates, screws, and washers in reverse order as disassembled. Torque screws 30 to 36 in.-lbs. (see Figure 76 - 6).

c. **Throttle Cable Adjustment and Testing**

1. Remove the throttle cable from the throttle plate control arm (see Figure 76 - 3), and push the cable into the full open condition.
2. Loosen the jam nut on the eyelet end of the throttle control cable.
3. Move the throttle plate control arm into the full open position, and check that the arm is fully seated against its stop.
4. Keeping a minimum of 5 threads engagement of the throttle control cable into the rod end of the cable eyelet, adjust the cable eyelet until the backside of the throttle control cable knob is $^{1}/_{8}$ to $^{1}/_{4}$ in. from the cable panel nut (see Figure 76 - 17) and the bolt can be reinserted through the rod eye and throttle plate lever arm. In this position and with the bolt inserted, push in on the throttle control knob. A spring effect should be felt while pushing in on the cable knob the distance shown in Figure 76 - 17 due to flexing in the cable housing.
5. Install the throttle control cable to the throttle plate control arm. Install the bolt, washers, and nut in reverse order as disassembled. Torque nut 30 to 36 in.-lbs. (see Figure 76 - 3).
6. Push then pull the throttle control cable fully in and out. No binding should be felt throughout its full range of motion.
7. Check the throttle plate control arm motion while operating the cable.
8. Check that all outside entities such as wiring and hoses are clear of the arm motion.
9. Check that the cable-locking mechanism is functioning properly by not allowing cable movement unless the thumb wheel is loosened.

d. **Throttle System Inspection and Maintenance**

1. Check that no binding is present during push-pull cable operation. Replace the throttle control cable if binding is felt with the cable disconnected from the throttle plate control

arm. If binding is present only with the cable connected to the throttle plate control arm, throttle plate service is required.

2. Check that the throttle plate control arm is fully seated in the full open position with proper clearance between the control knob and cable panel nut. If required, adjust the fuel throttle control cable per section 76-4 of this document.
3. Check bolt torque at the cable to throttle cable bracket location (see Figure 76 - 4). If nuts are loose or backed off, replace both bolts and nuts. Torque nuts 30 to 36 in.-lbs.
4. The throttle control cable is a self-contained unit. No servicing of the cable is required.

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76-4 FUEL MIXTURE DESCRIPTION, OPERATION, REMOVAL, INSTALLATION

Fuel is delivered to the engine by a mechanical fuel pump mounted directly to the engine. The quantity or mixture of fuel is controlled by a linear motion push-pull cable attached directly to the fuel pump's mixture control arm (see Figure 76 - 7 or Figure 76 - 8). During engine operation, the pilot adjusts the fuel mixture by either pulling out or pushing in on the control cable. A locking mechanism is built into the push-pull cable to prevent the cable from moving after the desired position has been set. To deactivate the locking mechanism the pilot presses a button located on the end on the cable knob allowing movement. Fine or precise adjust to the fuel pump mixture control arm can be accomplished by rotating the control knob either clockwise or counterclockwise resulting in small and controlled cable movement.

For S/N 42003 to 42044 a full rich spring is connected between the fuel pump mixture control arm and the fuel mixture cable bracket. If the fuel mixture cable were to break or become disconnected, the full rich spring will pull the mixture control arm into the full rich position preventing engine stall. For S/N 42045 and on, the full rich spring is not used.

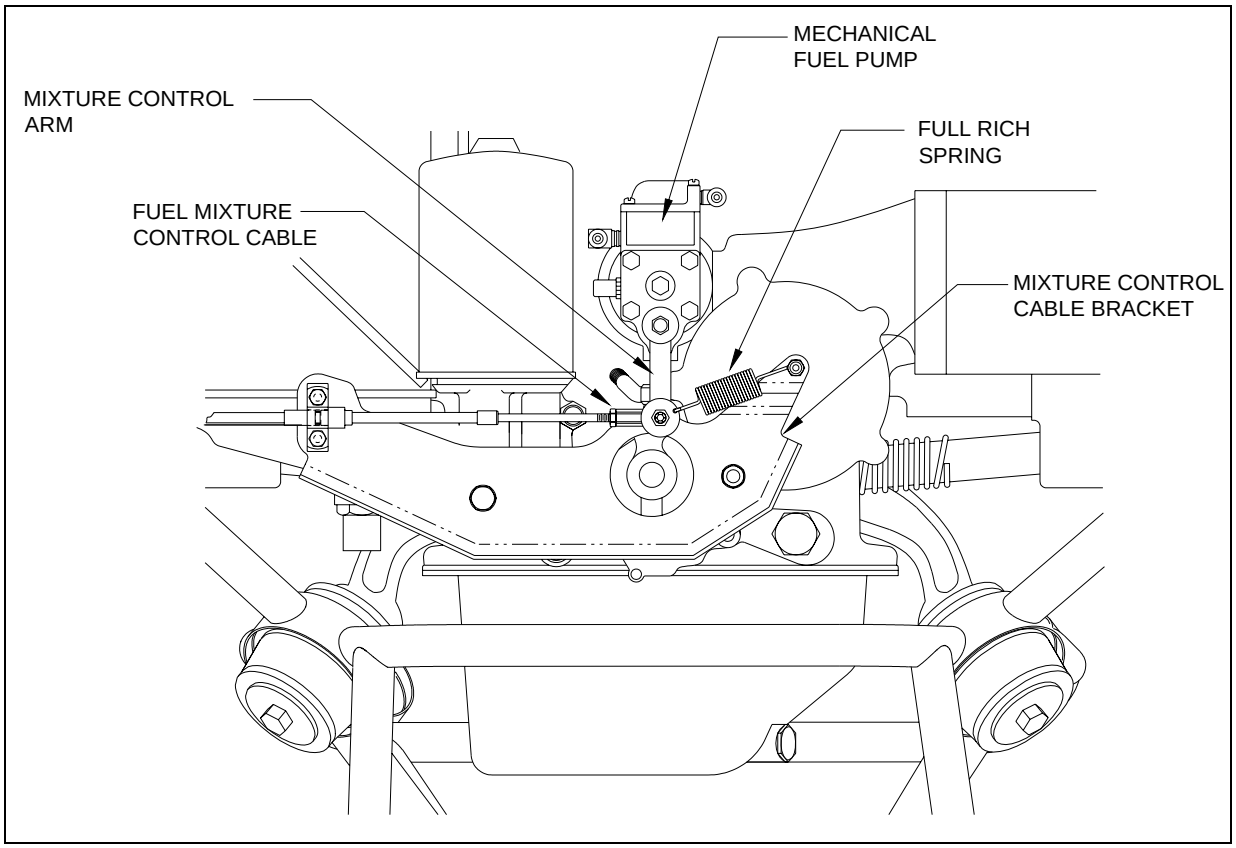

Figure 76 - 7 Fuel Mixture Control (S/N 42003 T0 42044)

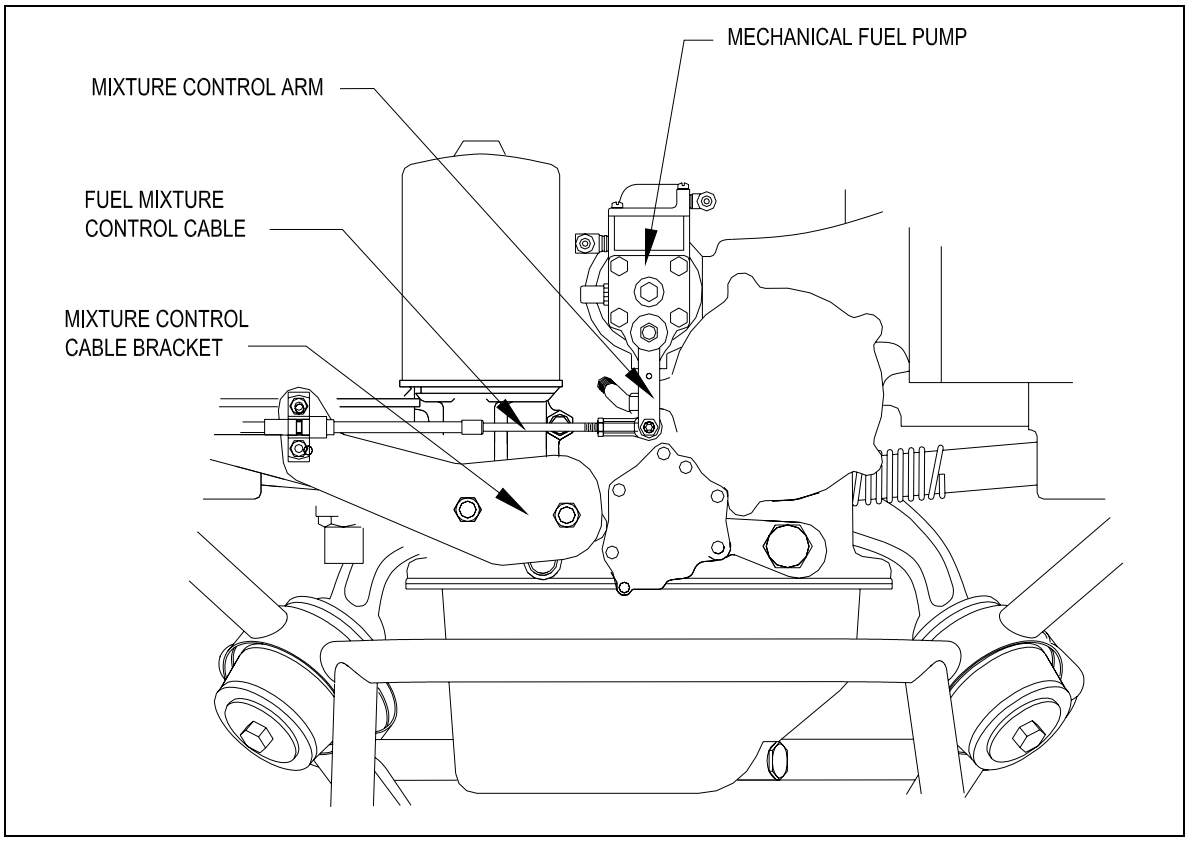

Figure 76 - 8 Fuel Mixture Control (S/N 42045 and on)

a. **Fuel Mixture Control Removal**

1. Remove the fuel mixture control cable from the fuel pump mixture control arm. Remove the bolt, nut, washers, full rich spring (S/N 42003 to 42044), and cotter pin (see Figure 76 - 9).

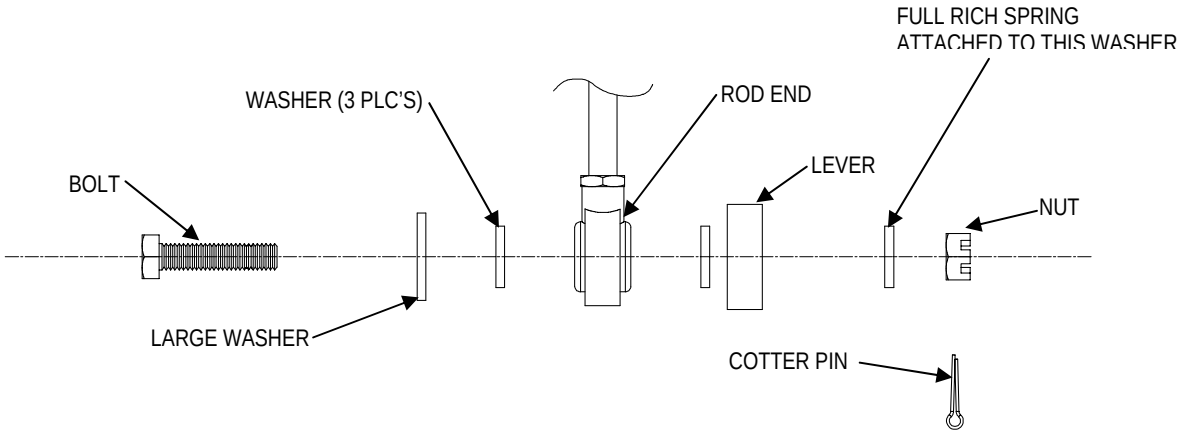

Figure 76 - 9 Fuel Mixture Control Assembly

2. Remove the mixture control cable housing from the fuel mixture control bracket by removing the bolts, nuts, clamp, spacers, and washers (see Figure 76 - 10).

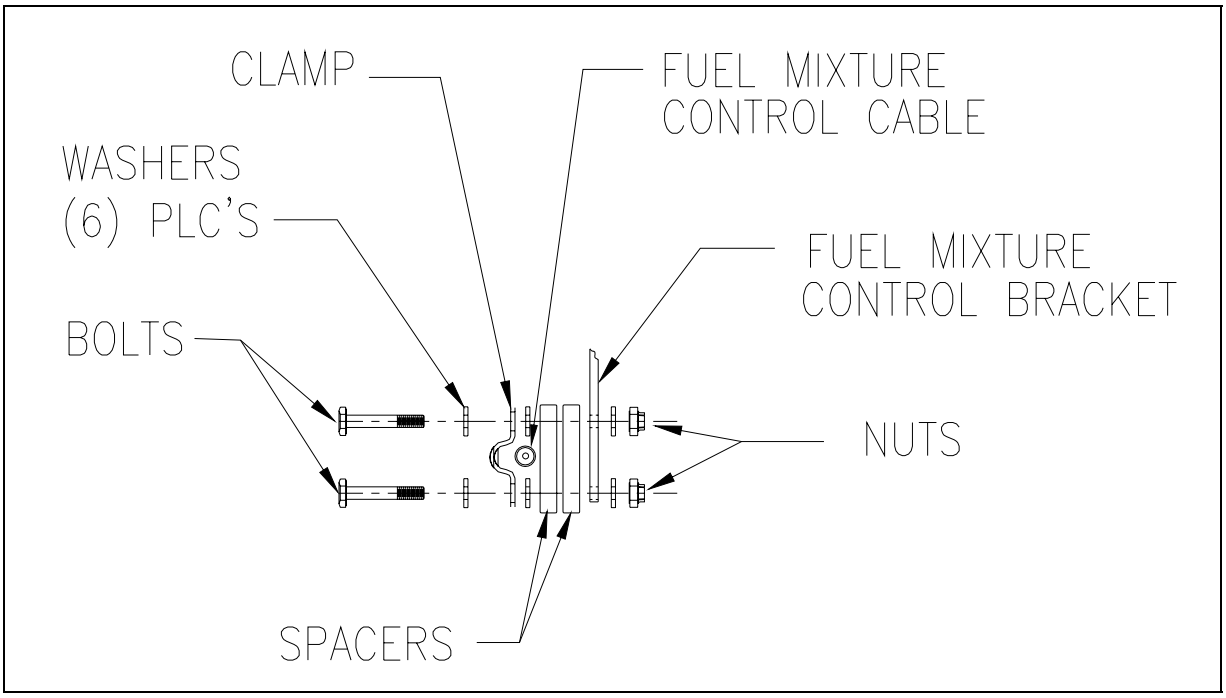

Figure 76 - 10 Fuel Mixture Control Assembly

3. Remove the cable guide plates from the firewall heat shield by removing the screws, guide plates, and washers (see Figure 76 - 11).

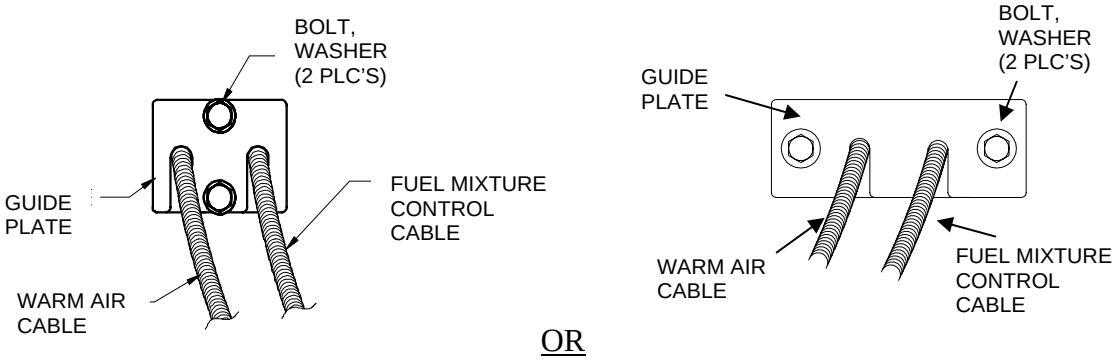

Figure 76 - 11 Fuel Mixture Control Guide Plate

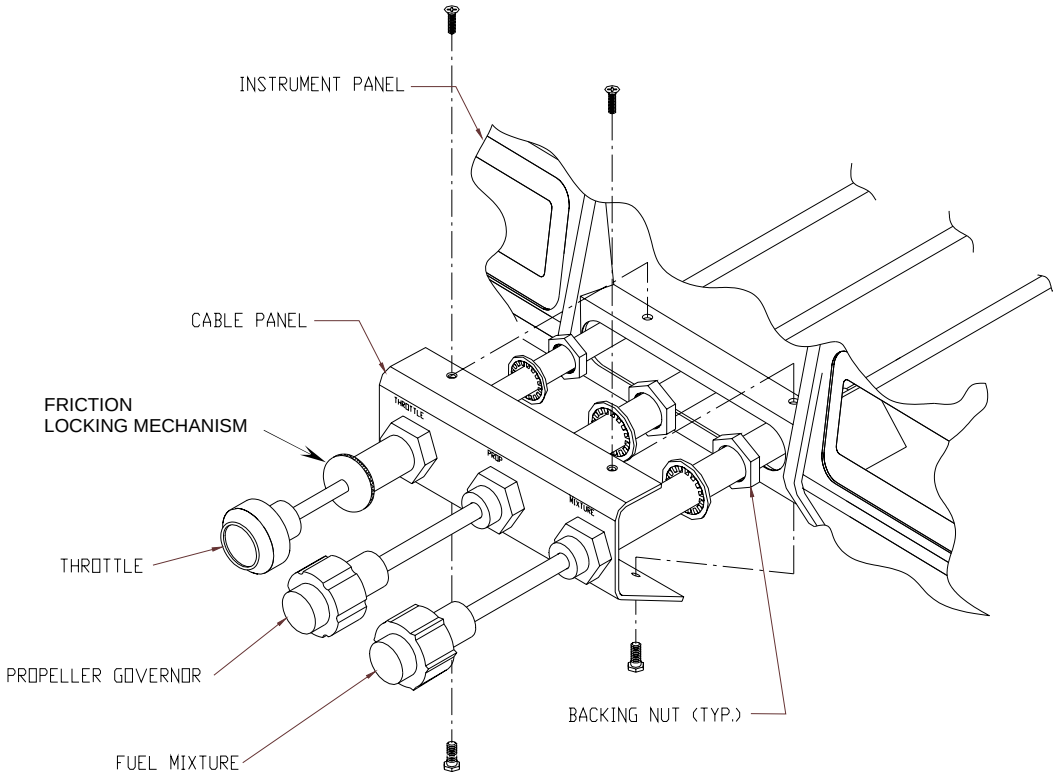

Figure 76 - 12 Fuel Mixture Control Panel Type 1

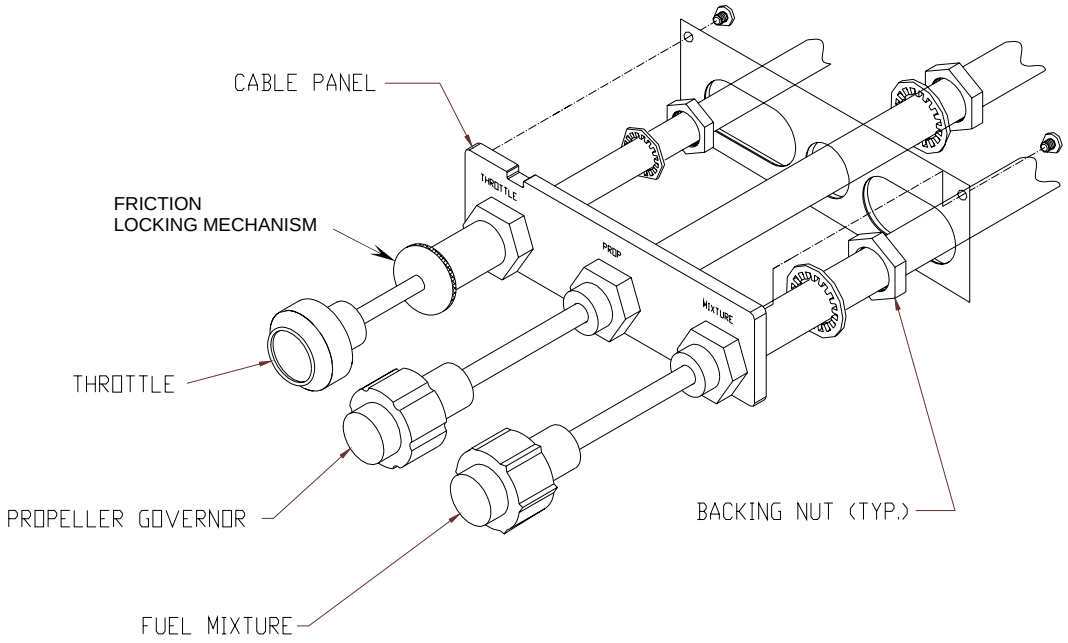

Figure 76 - 13 Fuel Mixture Control Panel Type 2

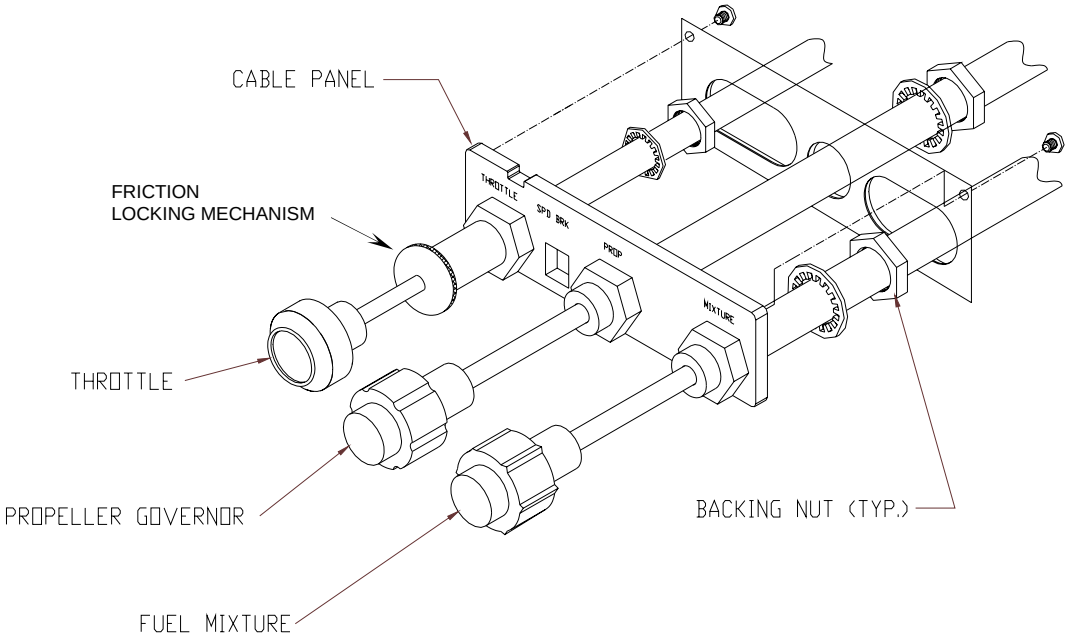

Figure 76 - 14 Fuel Mixture Control Panel Type 3

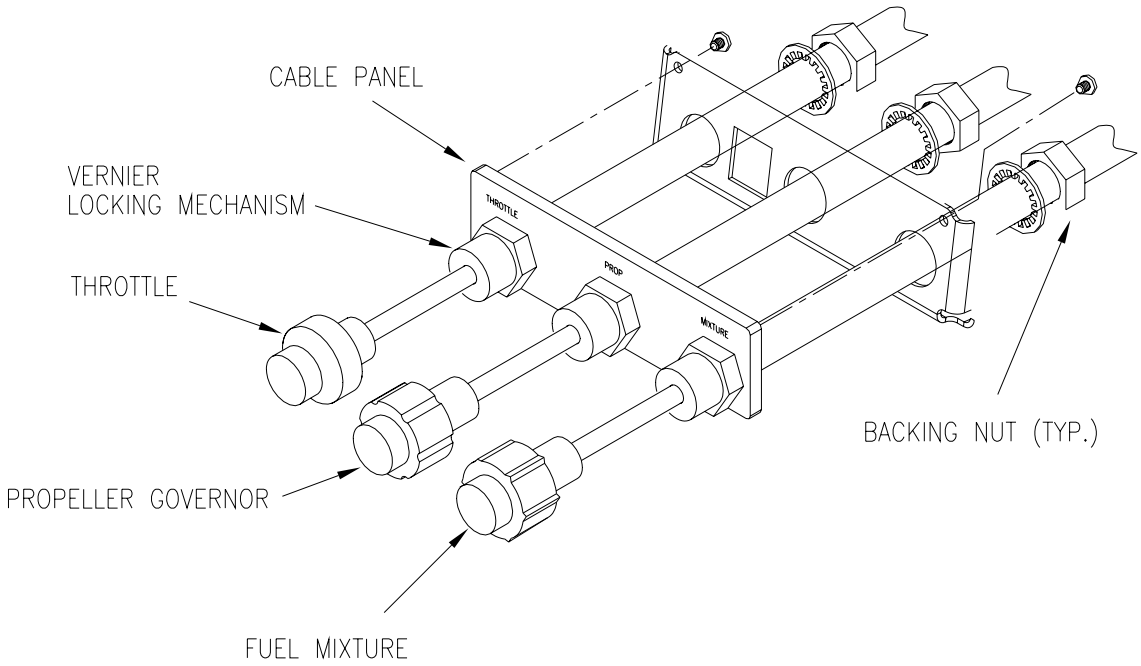

Figure 76 - 15 Fuel Mixture Control Panel Type 4

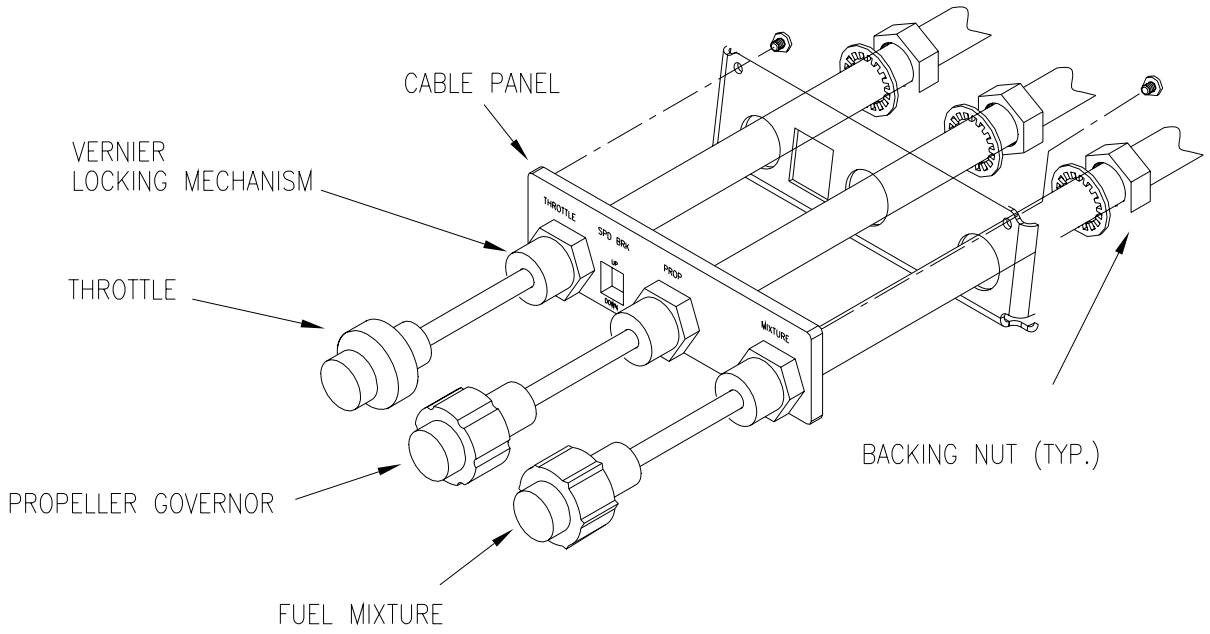

Figure 76 - 16 Fuel Mixture Control Panel Type 5

b. **Fuel Mixture Cable Installation**

1. Install the fuel mixture control cable through the panel cover by sliding the backing nut and washer over the cable housing. Torque the backing nut 240 to 280 in.-lbs.
2. Install the panel cover assembly into the instrument console, and install two or four panel screws depending on type of cover plate installed. Torque the panel screws 12 to 15 in.-lbs. (see Figure 76 - 12, Figure 76 - 13, Figure 76 - 14, Figure 76 - 15, or Figure 76 - 16).
3. Route the cable through the firewall bulkhead.
4. Install the GPS and cover plates in reverse order as disassembled.
5. Install the fuel mixture control cable to the engine driven fuel pump by installing the bolt, washers, new cotter pin, and nut in reverse order as disassembled. Torque nut 30 to 36 in.-lbs. (see Figure 76 - 9).
6. Install the fuel mixture control cable housing to the fuel mixture bracket by installing bolts, washers, clamp, spacer, and nuts in reverse order as disassembled. Torque nuts 30 to 36 in.-lbs.
7. With a large bend radius in the fuel mixture control cable between the firewall heat shield and the fuel mixture bracket, install the cable guide plates, screws, and washers. Torque screws 30 to 36 in.-lbs. (see Figure 76 - 11).
8. For S/N 42003 to 42044, reattach the full rich spring to the fuel pump mixture control arm washer. For S/N 42045 and on, there is no full rich spring.

c. **Fuel Mixture Adjustment and Testing**

1. Remove the fuel mixture control cable from the fuel pump mixture control arm (see Figure 76 - 9), and push the cable knob into the full rich position.
2. Loosen the jam nut at the eyelet end of the fuel mixture control cable.
3. Move the mixture control arm into the full rich position, and check that the arm is fully seated against its stop.

NOTE

The full rich spring will hold the arm in the fully seated position.

4. Keeping a minimum of 5 threads engagement of the fuel mixture control cable into the rod end of the cable eyelet, adjust the cable eyelet until the backside of the fuel mixture control cable knob is $^{1}/_{8}$ to $^{1}/_{4}$ in. from the cable panel nut (see Figure 76 - 17) and the bolt can be reinserted through the rod eye and mixture control lever. In this position and with the bolt inserted, push in on the mixture control knob. A spring effect should be felt while pushing in on the cable knob the distance shown in Figure 76 - 17 due to flexing in the cable housing.

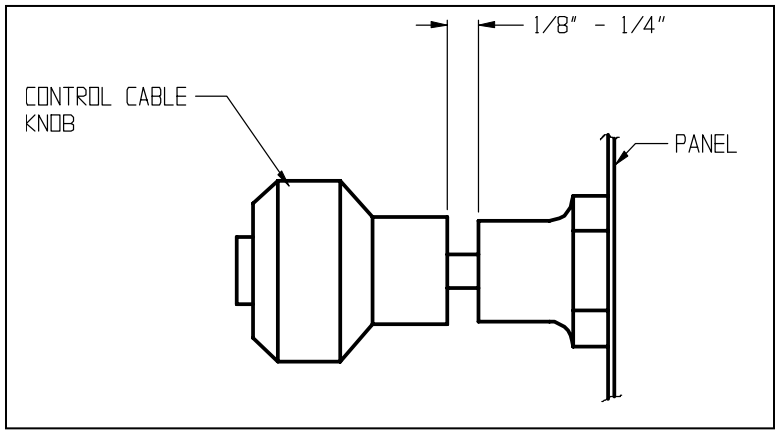

Figure 76 - 17 Proper Control Position

5. Install the fuel mixture control cable to the fuel pump mixture arm by installing the bolt, washers, cotter pin, and nut in reverse order as disassembled. Torque nut 30 in.-lbs., then apply additional torque, if necessary, until the slot of the nut aligns with the hole in the bolt (see Figure 76 - 9).
6. Push then pull the fuel mixture control cable fully in and out. No binding should be felt throughout its full range of motion.
7. Check the fuel pump mixture control arm motion while operating the cable. For S/N 42501 and on, ensure the angle between the control arm and cable (measured upward from the centerline of the cable end to the centerline of the control arm) is 170º max. when the control arm is in the idle position.
8. Check that all outside entities such as engine components, wiring, and hoses are clear of arm motion. For S/N 42501 and on, check that clearance is 0.125" min. except between the arm at full rich position and the air conditioning compressor drive pulley and belt which must be 0.25" min.; the existing #10 AN970-3 washer between the mixture control cable and the arm may be replaced with a #10 MS27183-9 washer as required.
9. Check that the cable-locking mechanism is functioning properly by not allowing cable movement unless the knob button is pressed.
10. Rotate the fuel mixture control cable knob fully clockwise and counterclockwise. Check that the fuel pump mixture control arm is moving in a smooth fashion while rotating the knob.

d. **Fuel Mixture System Inspection and Maintenance**

1. Check that no binding is present during push-pull and clockwise/counterclockwise cable operation. Replace the fuel mixture control cable if binding is felt with the cable disconnected from the fuel pump mixture control arm. If binding is present only with the cable connected to the fuel pump mixture control arm, fuel pump service is required.
2. Check that the fuel pump mixture control arm is fully seated in the full rich position with proper clearance between the control knob and cable panel nut. If required, adjust the fuel mixture control cable per 76-4 of this document.
3. Check wear between the full rich spring hook-end and the fuel pump mixture control arm. Replace both the full rich spring and the fuel pump mixture control arm if wear is present.
4. Check wear between the full rich spring hook-end and fuel mixture bracket shoulder bolt. Replace both the full rich spring and fuel mixture bracket shoulder bolt if wear is present.
5. Check the bolt torque at the cable to fuel mixture bracket location (see Figure 76 - 10). If nuts are loose or backed off, replace both bolts and nuts. Torque nuts 30 to 36 in.-lbs.
6. The fuel mixture control cable is a self-contained unit. No servicing of the cable is required

76-5 INDUCTION HEAT CABLE

One butterfly valve located inside the engine induction tube directs either fresh, mixed, or hot air into the engine. A linear motion push-pull control cable is directly connected to the butterfly valve (see Figure 76 - 18). As the cable moves, the butterfly valve opens. Mixing of the air is accomplished when the valve is positioned somewhere between the fully closed and fully open positions.

During engine operation the pilot adjusts the incoming air temperature by either pulling out or pushing in on the control cable. A locking mechanism is built into the push-pull cable to prevent the cable from moving after the desired position has been set. To deactivate the locking mechanism, the pilot presses a button located on the end on the cable knob allowing movement.

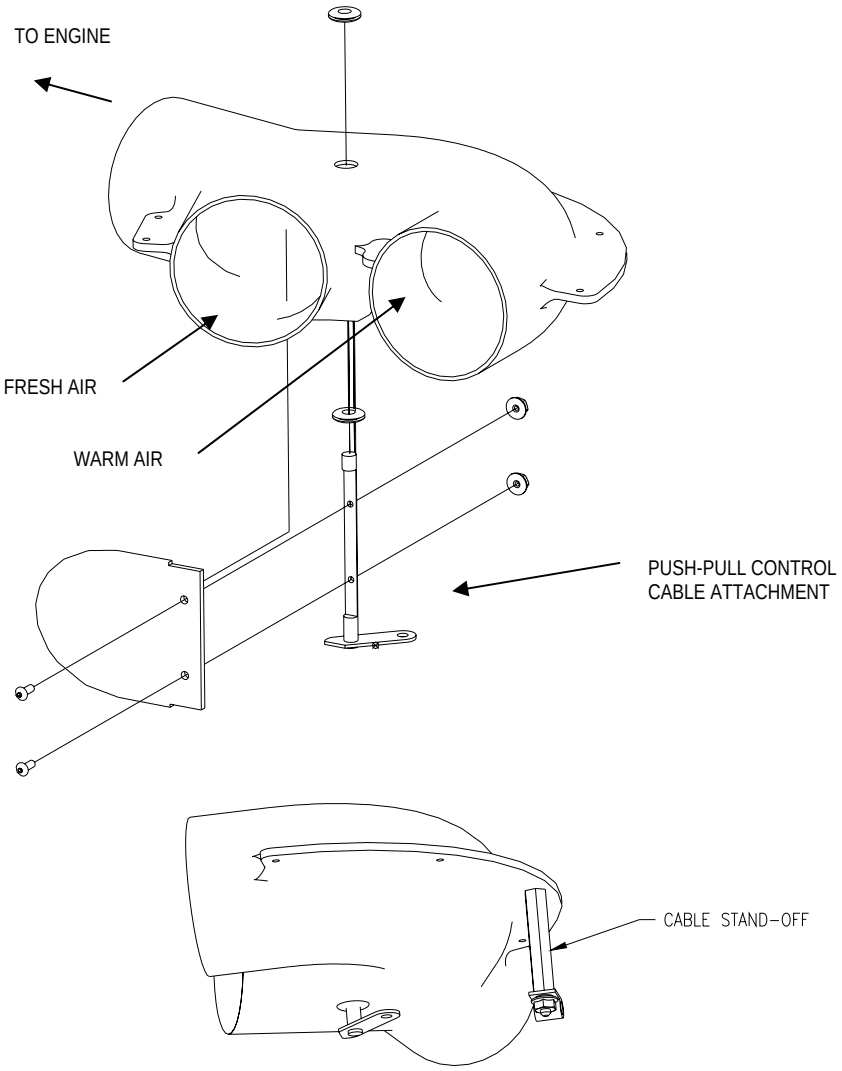

Figure 76 - 18 Induction Heat

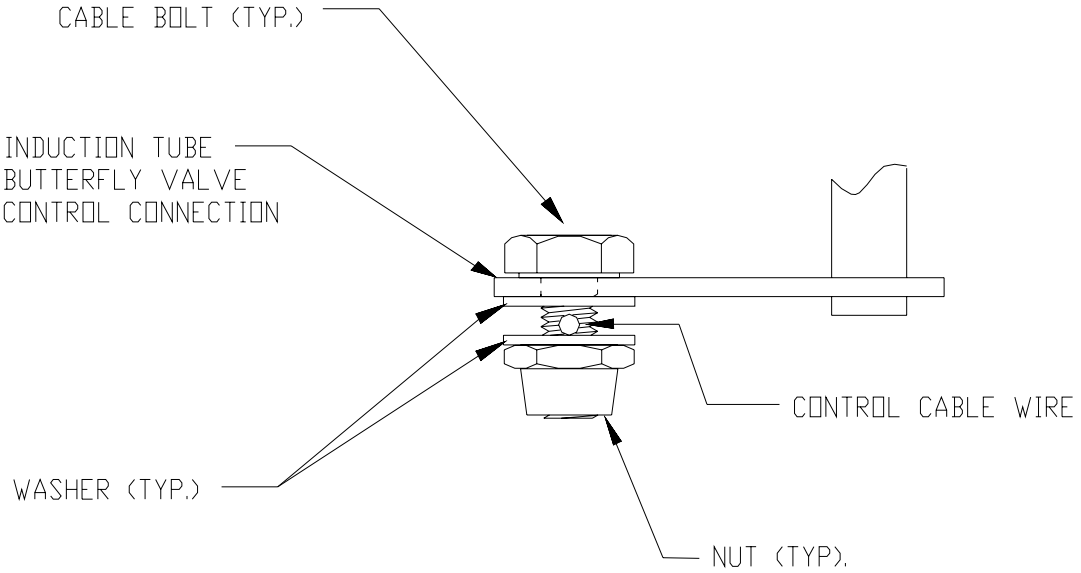

Figure 76 - 19 Induction Heat Control Cable Connection to Valve

a. **Induction Heat Cable Removal**

1. Remove the induction heat control cable from the induction tube butterfly valve lever. Loosen the bolt, nut, and washers until the cable can be pulled out from the bolt (see Figure 76 - 19).
2. Pull the induction heat control cable out from cable-stand-off and away from the induction tube.
3. Remove the three clamps, bolts, washers, and nuts attaching the control cable to the baffling, and pull the cable through the rubber grommet in the rear baffling component (See Figure 76 - 20).
4. Remove the cable guide plates from the firewall heat shield by removing the screws, guide plates, and washers (see Figure 76 - 21).
5. Working from behind the left side knee bolster panel, loosen the control cable nut until completely disengaged from the cable threads (see Figure 76 - 22). Pull the cable out from the knee bolster panel.

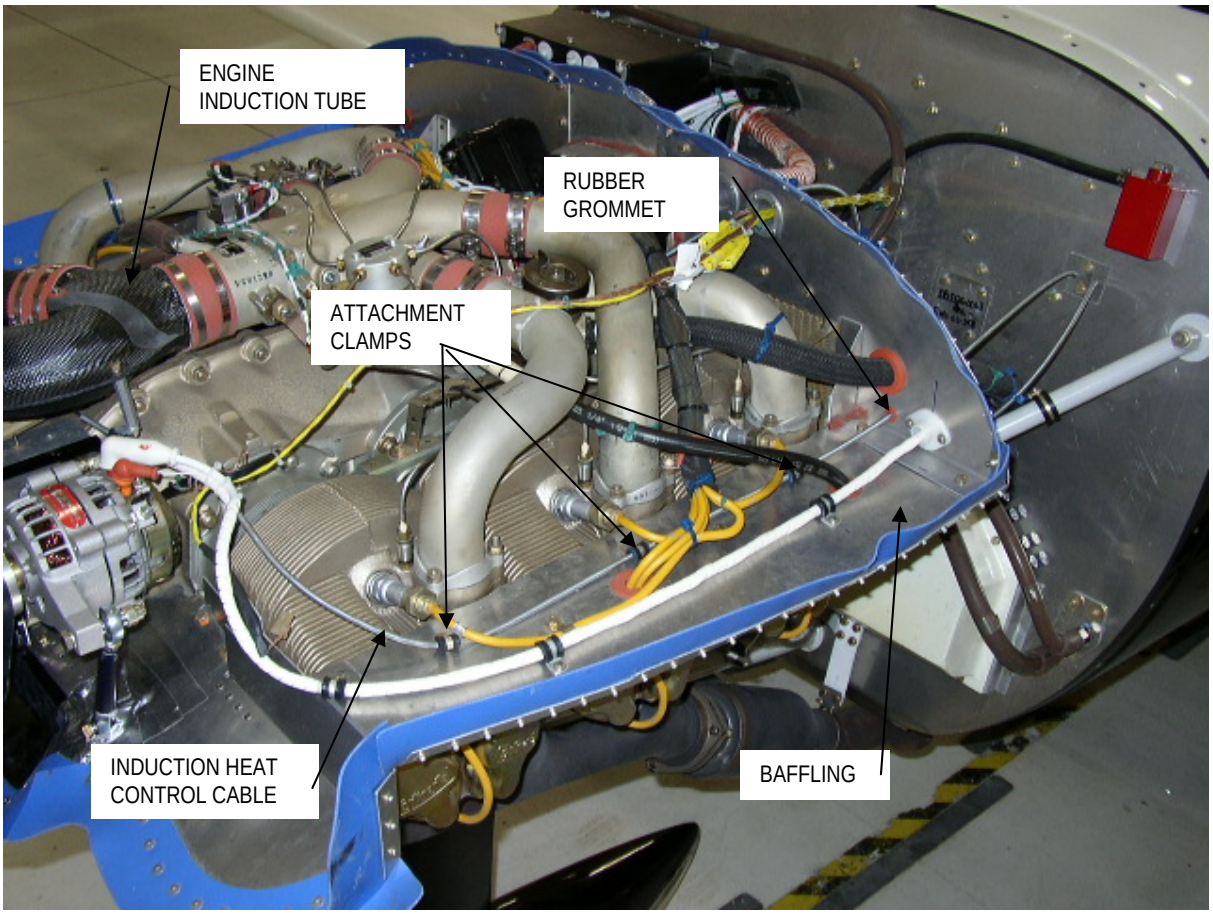

Figure 76 - 20 Induction Air Cable

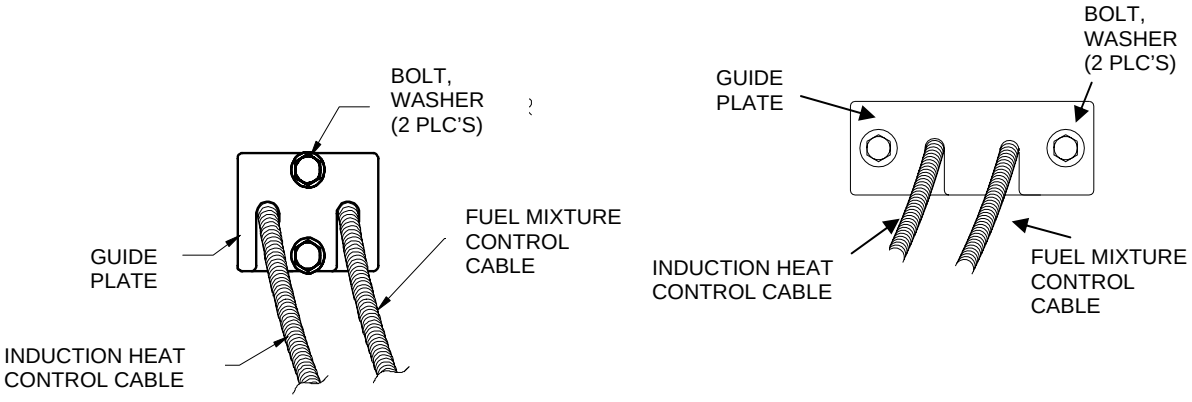

Figure 76 - 21 Induction Air Guide Plate

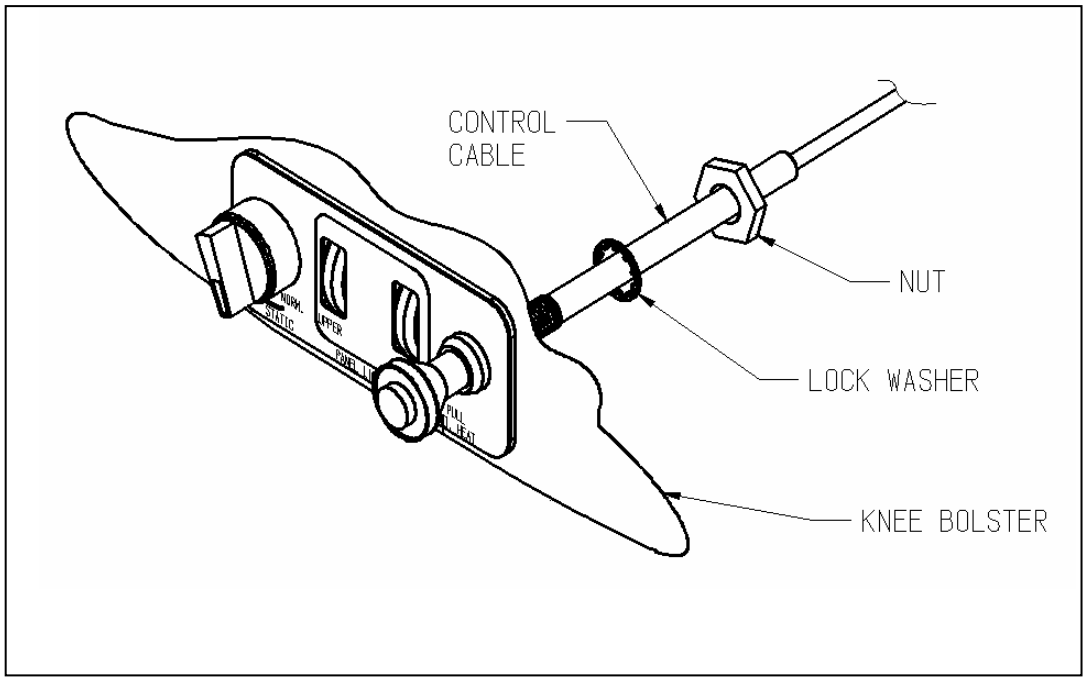

Figure 76 - 22 Induction Air Control Cable Panel Assembly

b. **Induction Heat Cable Installation**

1. Install the induction heat cable through the knee bolster panel by installing the lock washer and retaining nut over the cable housing, and screwing the nut onto the threaded part of the cable (see Figure 76 - 22). Torque retaining nut 55 to 60 in.-lbs.
2. Route the end of the control cable through the firewall, then through the rubber grommet on the rear baffling component (see Figure 76 - 20).
3. Reinstall the guide plates over the control cables passing through the firewall in reverse order as disassembled. Torque guide plate retaining screws and washers 30 to 36 in.-lbs.
4. Route the end of the cable through the cable stand-off, and slide the end of the cable wire through the hot air lever bolt. Hand tighten cable stand-off nut snug, do not over tighten.
5. With the control cable knob in the full fresh air position (pushed completely in), position the butterfly valve inside of the induction tube so that the hot air valve is completely closed. Verify that when the control cable is completely pushed up against the housing, the butterfly valve is in the correctly adjusted position. Torque the cable wire retaining nut 20 to 24 in.-lbs. while in this position.
6. Reinstall the control cable to the baffling retaining clamps, bolts, washers, and nuts in reverse order as disassembled (see Figure 76 - 20). Torque nuts 30 to 36 in.-lbs.

c. **Induction Heat Control Cable Maintenance**

The induction heat control cable is a maintenance-free component. No lubrication is required.

d. **Induction Heat Control Cable Inspection**

1. Check the control cable function by pressing in the cable locking button and moving the cable fully in and out. No binding should be felt during cable operation.

2. Check the butterfly valve position by pushing in fully on the control cable knob. The fresh air valve should be fully open while in this position.
3. Check the connection at the cable to the cable stand-off. Connection should be tight and not allow any cable movement.
4. Check the control cable wire to the hot air lever retaining bolt connection. Cable wire should be secure within the bolt with no slippage between the bolt and wire.

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CHAPTER
77

ENGINE INDICATION

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Chapter 77

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77-1 GENERAL

This chapter includes instructions on the maintenance, removal, and installation of equipment that provides a means of monitoring the operating conditions of the engine. Information from technical publications provided by the manufacturer's of the various pieces of engine monitoring equipment are not covered in this manual and should be consulted prior to any maintenance of the equipment. Information contained in this chapter provides the basic procedures for the removal and installation of the different pieces of the engine indicating system.

a. **Basic Option –** The instruments that display the status of the various engine systems are located on the engine instrument panel. This panel is to the left side of the flight instrument panel. The left and right fuel quantity gauge is at the top left of the panel and is discussed in Chapter 28. The dual ammeter gauge is at the middle left of the panel and is discussed in Chapter 24. The oil pressure and oil temperature gauge is located at the bottom left of the panel and is discussed in Chapter 79. The tachometer, manifold pressure, fuel flow, cylinder head, and exhaust gas temperature gauges are located on the right side of the engine instrument panel and are discussed in this chapter.

b. **Avidyne Option –** Aircraft equipped with the Avidyne Entegra EX5000 Multifunction Display (MFD) with EMAX total engine monitoring system will display engine operating conditions on this unit, as well as on the instruments in the flight instrument panel. The MFD is discussed in Chapter 34.

c. **Garmin G1000 Option –** Aircraft equipped with the Garmin G1000 system will display engine operating conditions on the MFD. Additional engine indicating gauges will not be present in the engine instrument panel.

CAUTION

Before any maintenance, removal, or installation of equipment is performed, make sure both the master and avionics switches are in the off position. Failure to do so could result in damage to these instruments.

NOTE

If any hoses, wires, or lines are touching the engine induction tubes, oil radiator, or engine mount apply a dab of high temperature silicone between them.

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77-2 TACHOMETER

a. **Removing the Tachometer**
   1. Remove the engine instrument panel per instructions in Chapter 25.
   2. Unplug the connector going into the back of the tachometer.
   3. Unscrew the four screws surrounding the tachometer.

b. **Installation of the Tachometer**
   1. Put the tachometer in its slot, and screw it into the engine instrument panel.
   2. Plug the connector into the back of the tachometer.
   3. Reinstall the engine instrument panel.

c. **Troubleshooting the Tachometer Transducer**
   1. The tachometer transducer is located in the right magneto.
   2. Verify that there is approximately 10VDC of excitation to the transducer with the master switch on and the tachometer connected to the aircraft wire harness.
   3. Verify ground is good.
   4. Check the continuity of the signal wire to the tachometer.
   5. If all these points check good, replace the transducer.

d. **Removing the Tachometer Transducer**
   1. The tachometer transducer is located in the right magneto.
   2. Loosen the magneto to gain access to the bottom of the magneto.
   3. Remove the transducer.
   4. Re-time the magneto.

e. **Installing the Tachometer Transducer**
   1. Loosen the right magneto.
   2. Install the transducer.
   3. Re-time the magneto.

77-3 MANIFOLD PRESSURE GAUGE

Depending upon the avionics equipment installed in the aircraft, there will either be a manifold pressure gauge or a manifold pressure/fuel flow gauge. For removal of the manifold pressure/fuel flow gauge see paragraph 77-4.

a. **Removing the Manifold Pressure Gauge**
   1. Remove the engine instrument panel per instructions in Chapter 25.
   2. Unplug the connector going into the back of the gauge.
   3. Unscrew the four screws surrounding the manifold pressure gauge.
   4. Remove the gauge.

b. **Installation of the Manifold Pressure Gauge**
   1. Put the gauge in its slot and screw it into the engine instrument panel.
   2. Plug the connector into the back of the gauge.
   3. Reinstall the engine instrument panel.

c. **Troubleshooting the Manifold Pressure Transducer**
   1. The manifold pressure transducer is located on the top of the engine alongside the throttle butterfly valve.
   2. Verify that there is approximately 5VDC of excitation to the transducer with the master switch on and the manifold pressure gauge connected to the aircraft wire harness.

3. Verify ground is good.
4. Check the continuity of the signal wire to the manifold pressure gauge.
5. If all these points check good, replace the manifold pressure transducer.

d. **Removing the Manifold Pressure Transducer**
1. The manifold pressure transducer is located on the top of the engine alongside the throttle butterfly valve.
2. Remove the connector.
3. Remove the transducer.

e. **Installing the Manifold Pressure Transducer**
1. Install the transducer.
2. Install the connector
3. Check for connector clearance with the throttle arm.

77-4 FUEL PRESSURE/FLOW GAUGE OR MANIFOLD PRESSURE/FUEL FLOW GAUGE

Depending upon the avionics equipment installed in the aircraft, there will either be a fuel pressure/flow gauge or a manifold pressure/fuel flow gauge. Either gauge will be installed in the same location on the engine instrument panel. For troubleshooting, removal, and installation of the manifold pressure transducer see paragraph 77-3

a. **Removing the Fuel Pressure/Flow Gauge or the Manifold Pressure/Fuel Flow Gauge**
1. Remove the engine instrument panel per instructions in Chapter 25.
2. Unplug the connector going into the back of the gauge.
3. Unscrew the four screws surrounding the gauge.
4. Remove the gauge.

b. **Installation of the Fuel Pressure/Flow Gauge or the Manifold Pressure/Fuel Flow Gauge**
1. Put the gauge in its slot and screw it into the engine instrument panel.
2. Plug the connector into the back of the gauge.
3. Reinstall the engine instrument panel.

c. **Troubleshooting the Fuel Pressure Transducer**
1. The fuel pressure transducer is located on the top of the engine on the right side.
2. Verify that there is approximately 5VDC (S/N 42001 to 42500) or 10VDC (S/N 42501 and on) of excitation to the transducer with the master switch on and the fuel flow gauge connected to the aircraft wire harness.
3. Verify ground is good.
4. Check the continuity of the signal wire to the fuel flow gauge.
5. If all these points check good, replace the fuel flow transducer.

d. **Removing the Fuel Pressure Transducer** (See Figure 77 - 1 or Figure 77 - 2)
1. With the upper cowling removed, disconnect the electrical connector from the top of the fuel flow transducer.
2. Disconnect the fuel line from the bottom of the fuel flow transducer.
3. Remove the tangential worm clamp used to secure the transducer to the upper bracket.
4. Remove the screws, washers, and nuts used to attach the upper and lower brackets together.

5. Loosen the two large clamps used to connect the upper and lower bracket retaining ears to the engine intake manifold runner-tube until the retaining ears on the upper bracket can be slid under the clamp.
6. Remove the upper bracket.
7. Remove the rubber vibration washer from the top of the fuel flow transducer.
8. Pull the fuel flow transducer up and out of the bottom bracket.

e. **Installing the Fuel Pressure Transducer** (See Figure 77 - 1 or Figure 77 - 2)

1. Install the fuel pressure transducer into the lower bracket.
2. Install the rubber vibration washer onto the upper part of the fuel flow transducer.
3. Install the upper bracket over the transducer. Slide the upper bracket retaining ears under the two large intake manifold runner-tube clamps. Position the clamps in the same location as before removal.
4. Reinstall the upper and lower bracket attachment screws, washers, and nuts in reverse order as disassembled. Hand tighten snug.
5. Reinstall the tangential worm clamp around the transducer and the upper bracket. Tighten the clamp until the transducer is completely contained.
6. Reconnect the fuel pressure hose to the bottom of the transducer. Apply Loctite 592 thread sealant to pipe threads on the fittings and the transducer. Assemble fittings to hand tight position, mark male and female fittings appropriately and add 1-1/2 turns to achieve proper angular orientation. Always loosen (back off) fitting within 1/2 turn from the ending torque position to achieve desired position.
7. Reconnect the electrical connections to the top of transducer.

f. **Other References** – See Chapter 28 for information on the fuel system.

77-5. FUEL FLOW TRANSDUCER

A fuel flow transducer transmits flow data to the Avidyne Entegra EX5000 Multi-function display or to the Garmin G1000 MFD. The transducer is located within the main fuel line on the upper right side of the engine.

a. **Removing the Fuel Flow Transducer**

1 Remove the fire sleeve.
2. Remove the electrical connector.
3. Remove the transducer.

b. **Installing the Fuel Flow Transducer**

1 Install the transducer.
2. Install the electrical connector.
3. Install the fire sleeve.

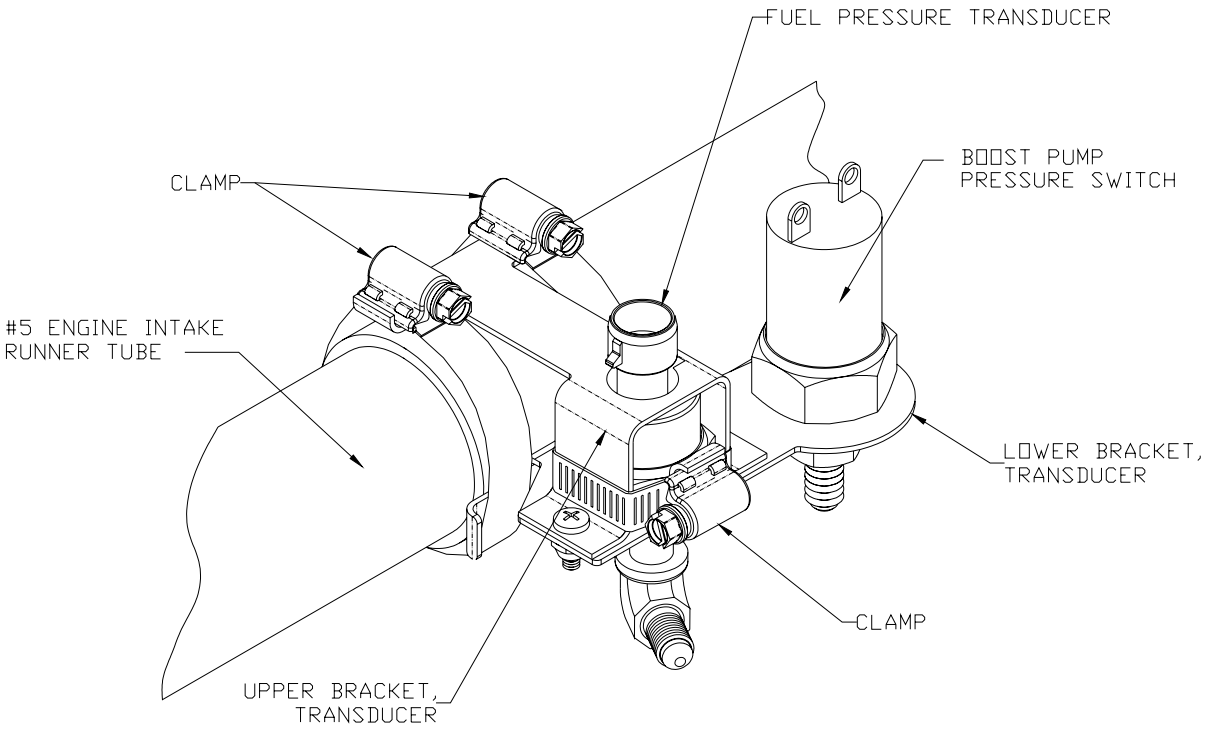

Figure 77 - 1 Fuel Pressure Transducer (Avidyne Option)

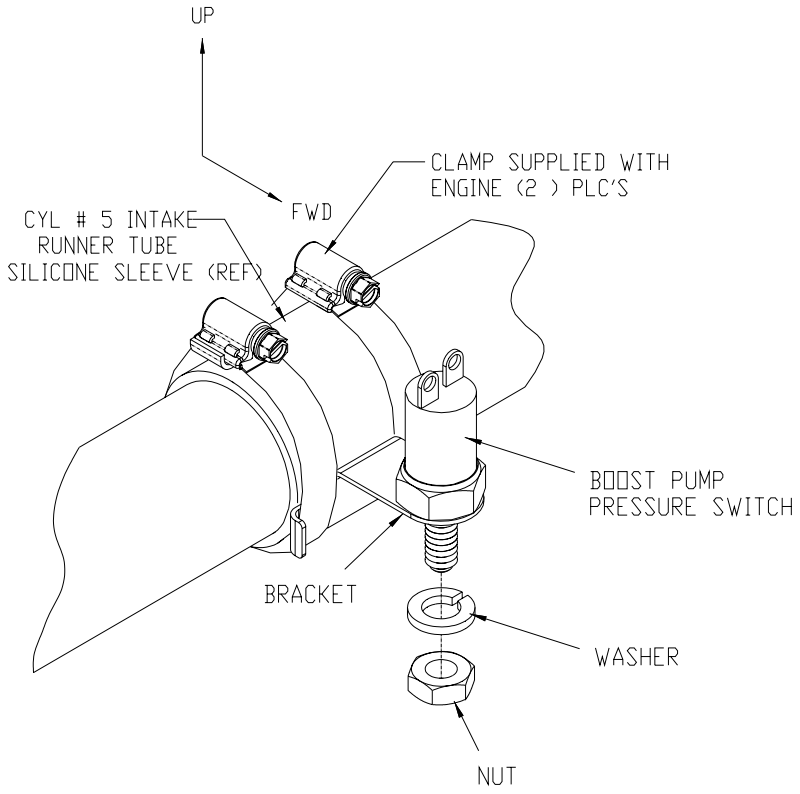

Figure 77 - 2 Fuel Pressure Switch (Garmin G1000 Option)

77-6. CYLINDER HEAD TEMPERATURE (CHT)

Aircraft equipped with the Avidyne EMAX Total Engine Monitoring System will have six cylinder head temperature (CHT) probes with an adapter probe attached to the probe in the #2 cylinder (see Figure 77 - 3). The adaptor probe reports to the CHT gauge in the engine instrument panel. If the aircraft is without EMAX there will be a single CHT probe inserted into the #2 cylinder which reports to the CHT gauge in the engine instrument panel. These probes are resistive thermal devices (RTD) and must have clean connections to work correctly. If a connection is resistive, the CHT reading will be high.

Aircraft equipped with the Garmin G1000 system will have six cylinder head temperature probes. The #2 cylinder adaptor probe will not be present. These probes are type J thermocouples.

a. **Removing the CHT Gauge (not present with Garmin G1000 option)**
   1. Remove the engine instrument panel (See Chapter 25 for instructions).
   2. Unplug the connector going into the back of the gauge.
   3. Unscrew the four screws surrounding the cylinder head temperature gauge.

b. **Installation of the CHT Gauge (not present with Garmin G1000 option)**
   1. Put the gauge in its slot, and screw it into the engine instrument panel.
   2. Plug the connector into the back of the gauge.
   3. Reinstall the engine instrument panel.

c. **Removing the CHT Probe(s)**
   1. With EMAX
      a) Remove the cowling per instructions in Chapter 76.
      b) The probes are attached with either integral tube nuts or bayonet attachment, except at the #2 cylinder. The probe at cylinder # 2 includes an additional adapter probe which will be either tube nut style or push and turn bayonet style (see Figure 77 - 3).
      c) Cut tie wraps and slide the protective sleeve from the wiring connections to the probe.
      d) Disconnect wiring from the probe.
      e) Unscrew the tube nut and remove the probe or push the probe up into the cylinder, turn it, and remove, depending upon the style of probe installed. For the #2 cylinder, remove the probe from the adaptor probe and then unscrew the adapter probe from the cylinder head.
   2. Without EMAX
      a) Remove the cowling per instructions in Chapter 76.
      b) Cut tie wraps and slide protective sleeve from the wiring connections to the probe.
      c) Disconnect wiring from the probe.
      d) Unscrew the tube nut and remove the probe or push the probe up into the cylinder, turn it, and remove, depending upon the style of probe installed.

d. **Installation of CHT Probe(s)**
   1. With EMAX
      a) Apply anti-seize compound to the CHT probe threads and to the CHT adaptor probe threads.
      b) For the #1, 3, 4, 5, and 6 cylinders, The probe is attached with either a tube nut or a bayonet clip. For bayonet style, push the probe up into the cylinder and turn it,

thus locking the probe into place. For tube nut style, tighten the tube nut tight plus ¼ turn.

c) For the #2 cylinder, insert the adaptor probe into the threaded hole in the bottom of the cylinder head and tighten tight plus ¼ turn. Then insert the CHT probe. If the CHT probe is attached by tube nut, tighten the tube nut tight plus ¼ turn. If the CHT probe is bayonet style, push the probe up into the cylinder and turn it to lock it into place.

d) Connect wiring to the probe(s). Slide the protective sleeve over the wiring connections and secure with tie wraps.

e) Install the cowling per instructions in Chapter 76.

2. Without EMAX

a) Apply anti-seize compound to the CHT probe threads and to the CHT adaptor probe threads.

b) The probe is attached with either a tube nut or a bayonet clip. For bayonet style, push the probe up into the cylinder and turn it, thus locking the probe into place. For tube nut style, tighten the tube nut tight plus ¼ turn.

c) Connect wiring to the probe. Slide the protective sleeve over the wiring connections and secure with tie wraps.

d) Install the cowling per instructions in Chapter 76.

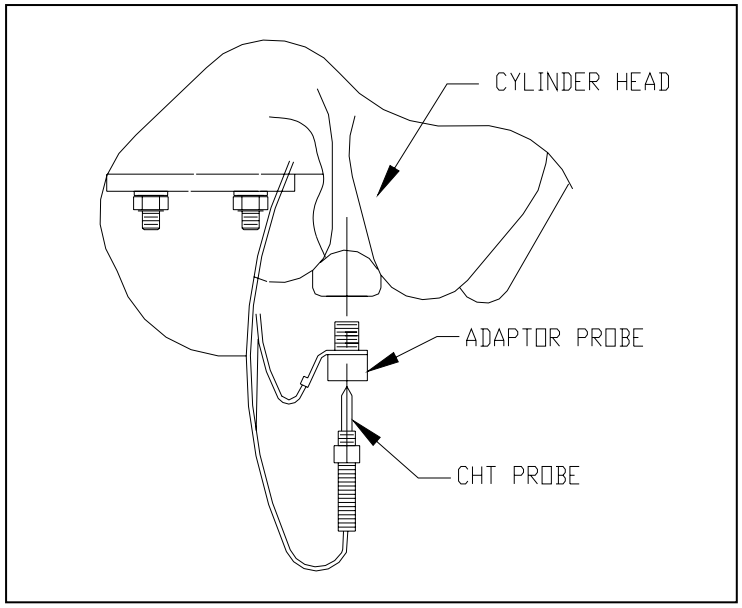

Figure 77 - 3 CHT Adaptor Probe

e. **Testing of the CHT Gauge**

NOTE

Testing of the cylinder head temperature gauge will require the use of a thermocouple oven, a reliable heat probe, and a power source independent of the airplane.

1. Plug the thermocouple oven into the power source.
2. Place the probe from the cylinder head temperature gauge into the top of the thermocouple oven heating element.
3. Place the standard probe into the bottom of the heating element.
4. Compare the temperatures.

f. **Calibration of the CHT Gauge**

1. Field calibration of the cylinder head temperature gauge is not available. For calibration of the cylinder head temperature gauge, see a qualified instrument technician.

77-7. EXHAUST GAS TEMPERATURE (EGT)

Aircraft equipped with the Avidyne EMAX Total Engine Monitoring System will have seven exhaust gas temperature probes. Six probes for the EMAX system and one probe for the gauge in the engine instrument panel. The exhaust stack for the #2 cylinder supports two of the seven probes. Aircraft equipped with the Garmin G1000 system will have six exhaust gas temperature probes.

a. **Removing the EGT Gauge**

1. Remove the engine instrument panel per instructions in Chapter 25.
2. Unplug the cable going into the back of the gauge.
3. Unscrew the four screws surrounding the exhaust gas temperature gauge.

b. **Installation of the EGT Gauge**

1. Put the gauge in its slot, and screw it into the engine instrument panel.
2. Plug the cable into the back of the gauge.
3. Reinstall the engine instrument panel.

c. **Removing the EGT Probe(s)**

1. Remove the cowling per instructions in Chapter 76.
2. The EGT probe is attached with a clamp to the exhaust stack.
3. Cut tie wraps and slide protective sleeve off probe wiring connection.
4. Disconnect wiring to the probe.
5. Loosen the clamp, and remove the entire clamp and probe assembly.
6. If necessary, remove wiring from the clamps at the baffling and baffling attachment bracket on the engine boss by removing the bolt, washers, clamp, and nut.

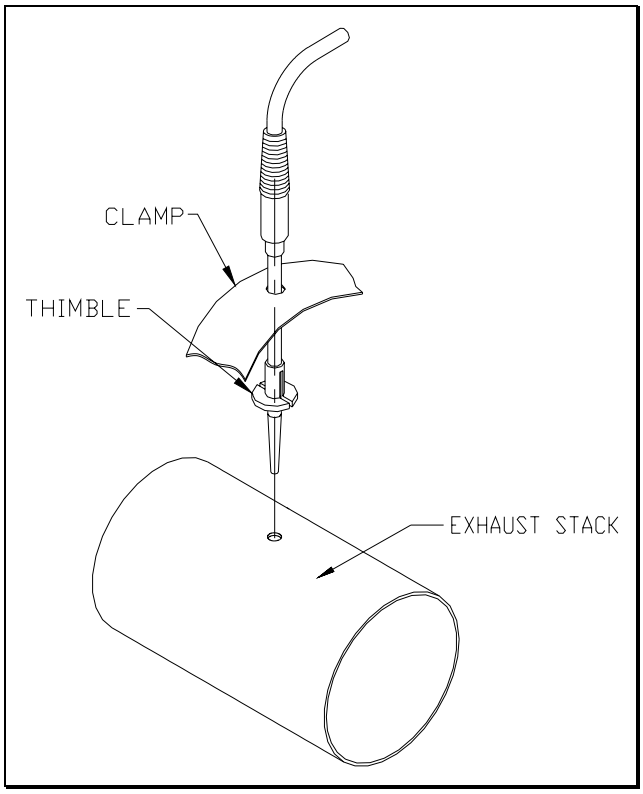

Figure 77 - 4 EGT Probe

d. **Installation of the EGT Probe(s)**

1. Insert the probe in the exhaust stack so that the tip of the probe is in the approximate center of the exhaust stack. Do not go over center.
2. Torque clamp 40 to 50 in.-lbs., and trim excess strap from clamp (not to exceed a 1 in. tab).
3. Connect wiring to probe and slide protective sleeve over the connections. Secure protective sleeve with tie wraps.
4. If wiring was removed from the clamps, install the bolt, washers, clamp, and nut. For clamps on the baffling, tighten the nut until the clamp is compressed and does not rotate. For clamps on the baffling attach bracket on the engine boss, torque bolt 72 to 84 in.-lbs.
5. Install the cowling per instructions in Chapter 76.

e. **Testing of the EGT Gauge**

NOTE

Testing of the exhaust gas temperature gauge will require the use of a thermocouple oven, a reliable heat probe, and a power source independent of the airplane.

1. Plug the thermocouple oven into the power source.
2. Place the probe from the exhaust gas temperature gauge into the top of the thermocouple oven heating element.
3. Place the standard probe into the bottom of the heating element.

4. Compare the temperatures.

f. **Calibration of the EGT Gauge**

1. Field calibration of the exhaust gas temperature gauge is not available. For calibration of the exhaust gas temperature gauge, take it to a qualified instrument technician.

77-8. OIL PRESSURE GAUGE

a. **Oil Pressure Transducer Removal**

1. Remove the electrical connection from the end of the oil pressure transducer.
2. Using a wrench, hold the AN912-3J bushing. Remove the oil pressure transducer from the oil cooler case. See Figure 77 - 5.

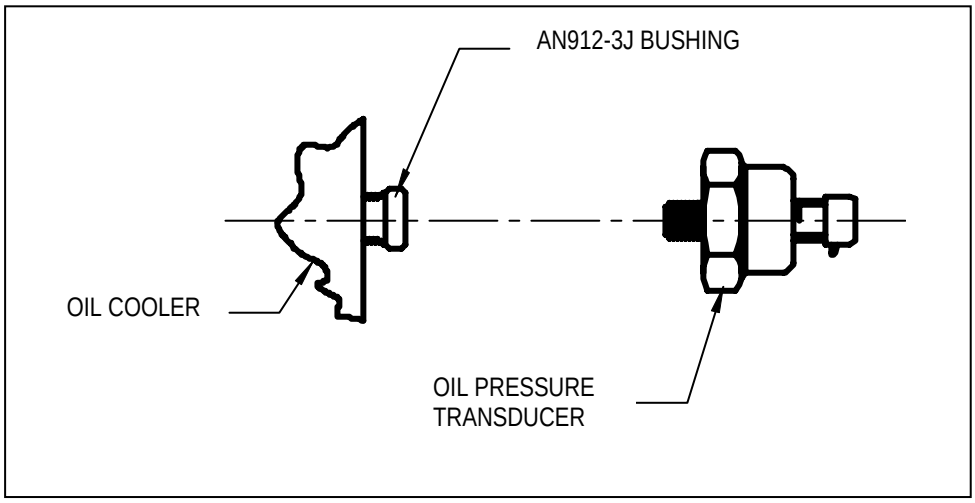

Figure 77 - 5 Oil Pressure Transducer

b. **Oil Pressure Transducer Installation**

1. Apply Teflon thread sealant (not tape) to the threads of the oil pressure transducer.
2. Install the oil pressure transducer into the AN912-3J fitting. Using a wrench, hold the AN912-3J bushing, and torque the oil pressure transducer 60 to 80 in.-lbs.
3. Attach the electrical connection to the end of the oil pressure transducer.

c. **Oil Pressure Transducer Troubleshooting**

1. The oil pressure transducer is located next to the Hobbs oil pressure switch under the oil cooler.
2. Verify that there is approximately 5VDC of excitation to the transducer with the master switch on and the oil pressure gauge connected to the aircraft wire harness.
3. Verify ground is good.
4. Check the continuity of the signal wire to the oil pressure gauge.
5. If all these points check good, replace the oil pressure transducer.

77-9. HOBBS SWITCH

a. See Chapter 31 for information on the Hobbs switch.

77-10. OIL TEMPERATURE

a. See Chapter 79 for information on the oil temperature system.

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CHAPTER 78

EXHAUST

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Chapter 78

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78-1. Exhaust System Description and Operation

This document describes operation, removal, and installation of the IO-550-N engine exhaust system. The exhaust system is a dual outlet configuration with baffle-style mufflers and canister-style heat exchanger.

The exhaust system is a dual outlet configuration consisting of a left and right side manifold, muffler, and tailpipe. A canister-style heat exchanger is housed over the right side muffler. Exhaust gasses produced by the engine are channeled through the manifold and muffler, then out the tailpipe. Mufflers are a non-glass packed baffle design fabricated to lower engine exhaust noise to the required dB(A) levels while still providing high flow characteristics.

Small pins welded within the right muffler, transfer heat into the heat exchanger chamber. Fresh air is channeled through the heat exchanger chamber where it is heated, then routed into the environmental control system for use as cabin floor heat and windshield defrosting (see Figure 78 - 1).

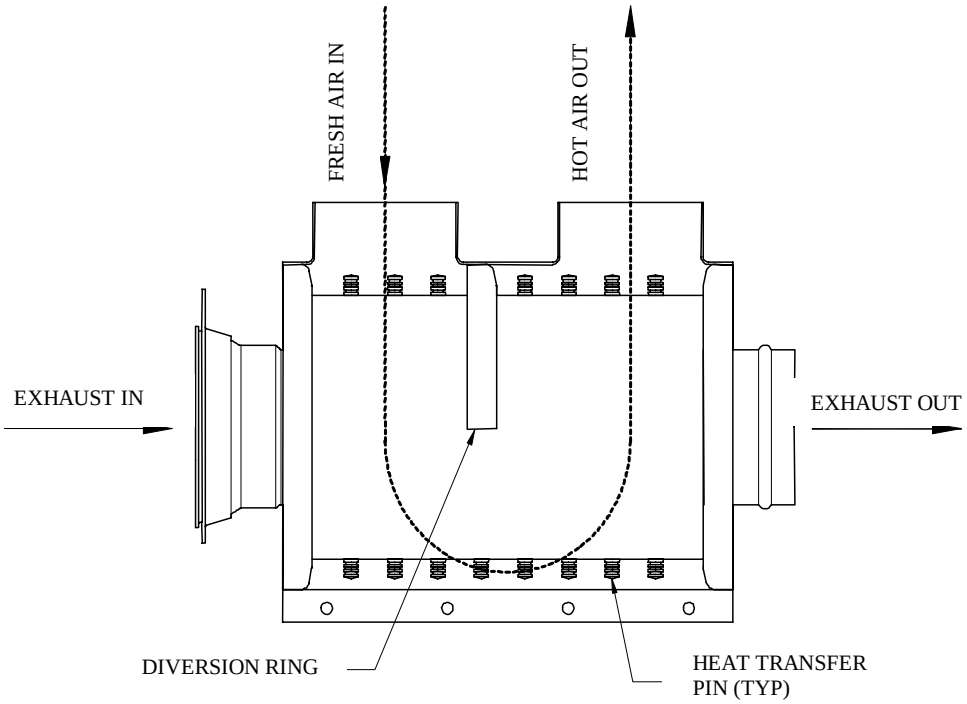

Figure 78 - 1 Exhaust System

Manifold to muffler connections are established using a conventional ball joint method. Spring pressure is used to seat the ball formation between the manifold and muffler flanges (see Figure 78 - 2).

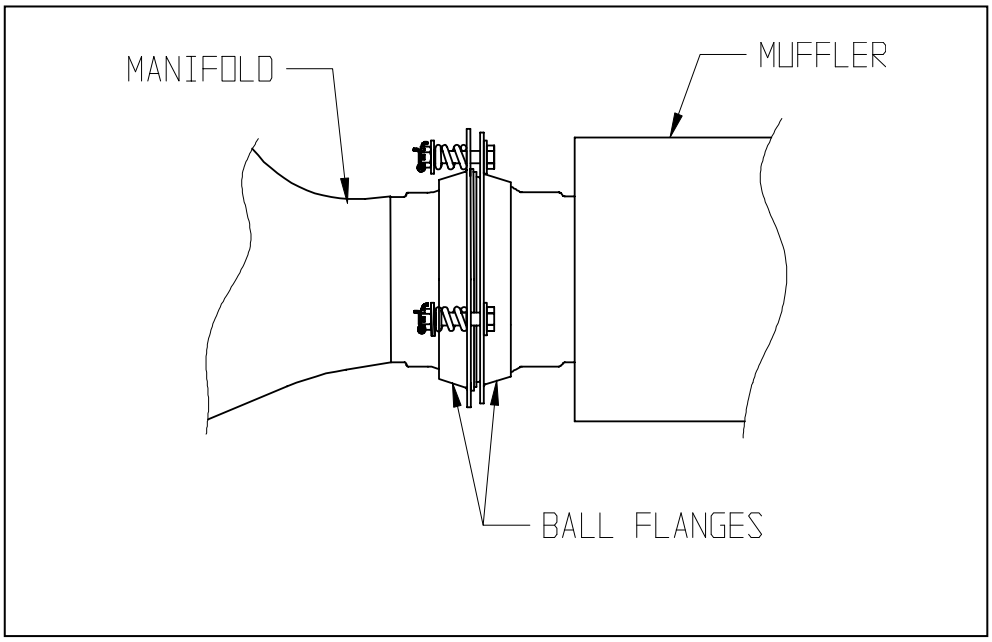

Figure 78 - 2 Manifold to Muffler Connection

Muffler to tailpipe connections are established using D-clamps. A formed ridge at both the muffler outlet pipe and receiving tailpipe end create an interlocking configuration when D-clamps are assembled over muffler to tailpipe connections (see Figure 78 - 3).

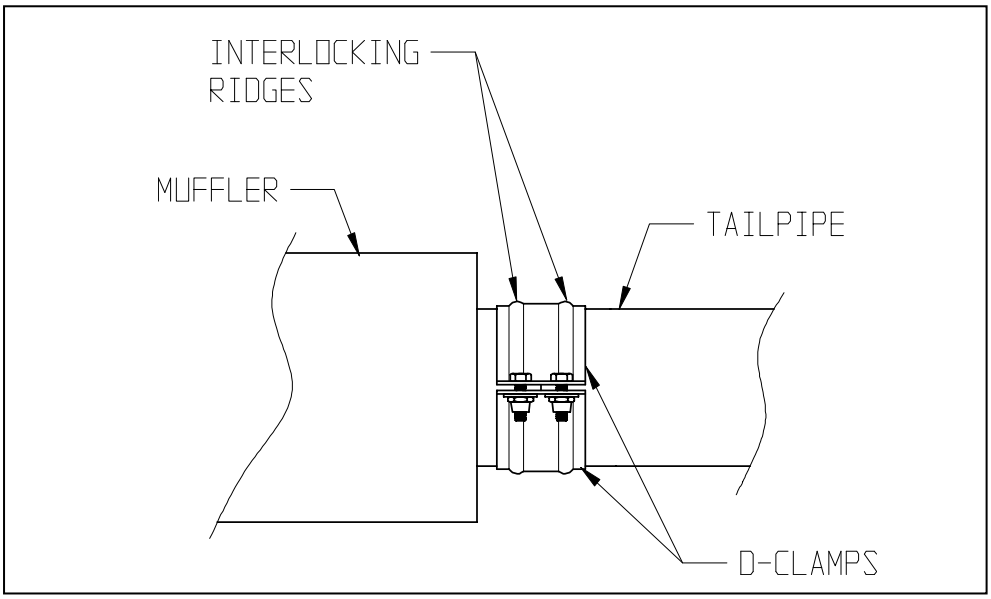

Figure 78 - 3 Muffler to Tailpipe Connection

Both the left and right side tailpipes are supported at each end by Teflon straps. These straps control vibration and movement that occurs during engine operation as well as support the weight at the exit end of each tailpipe (see Figure 78 - 4).

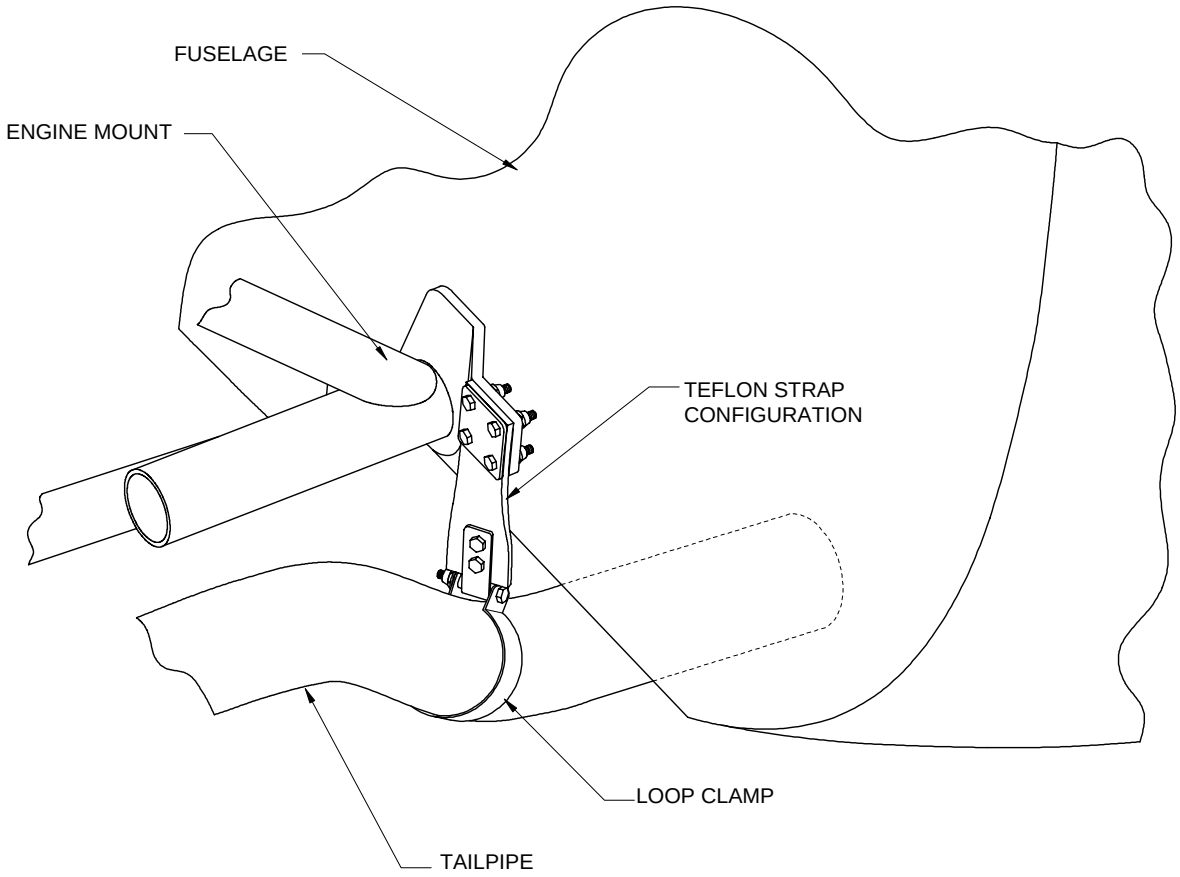

Figure 78 - 4 Tailpipes

The muffler assembly contains a tone cone (shown in Figure 78 - 5) that decreases exhaust noise.

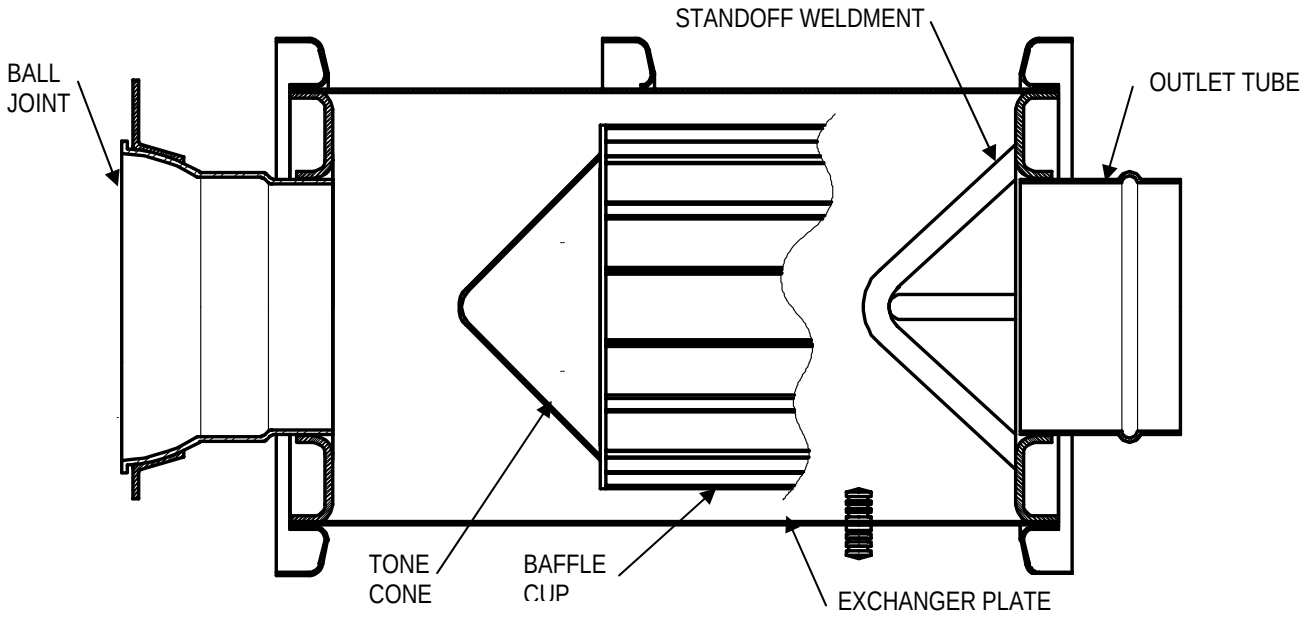

Figure 78 - 5 Muffler Assembly with Tone Cone

78-2. Tools & Equipment

The mechanic should be equipped with a complete set of the necessary tools that include the following:

a. Wrenches – ¼ in. through 1 ¼ in.
b. Common and Phillips Head Screwdrivers
c. Pliers – Common, Diagonal Cutters, Needle Nose, Duck Bill, Snap Ring, Safety Wire
d. Ratchets ¼ in., $^{3}/_{8}$ in., ½ in. drive
e. Sockets – ¼ in. drive $^{5}/_{32}$ in. through ½ in., $^{3}/_{8}$ in. drive $^{3}/_{8}$ in. through 1 in., ½ in. drive $^{7}/_{16}$ in. through 1¼ in.
f. Sockets (Deepwell) –$^{1}/_{2}$ in. drive, $^{7}/_{16}$ in. through 1 in.
g. Feeler Gauges
h. Leather or Soft Plastic Mallet
i. Torque Wrenches* 0-500 in.-lbs. and 0-100 ft.-lbs.
j. Micrometers*
k. Slide Hammer
l. Pullers
m. Thickness Gauges
n. Vernier Calipers*
o. Small Hole Gauges

* Must be currently calibrated, and the calibration must be traceable to the National Bureau of Standards.

WARNING

Mufflers must <u>never</u> be replaced with anything other than the OEM unit.

78-3. Exhaust System Removal

With the upper and lower cowling removed from the aircraft, remove the EGT probe from the left, rear exhaust manifold runner tube. Loosen the EGT clamp until the tail of the clamp is free, then carefully pull the EGT out of the exhaust runner tube (see Figure 78 - 6). If the optional digital engine scanner is installed, remove the EGT per instructions in Chapter 77.

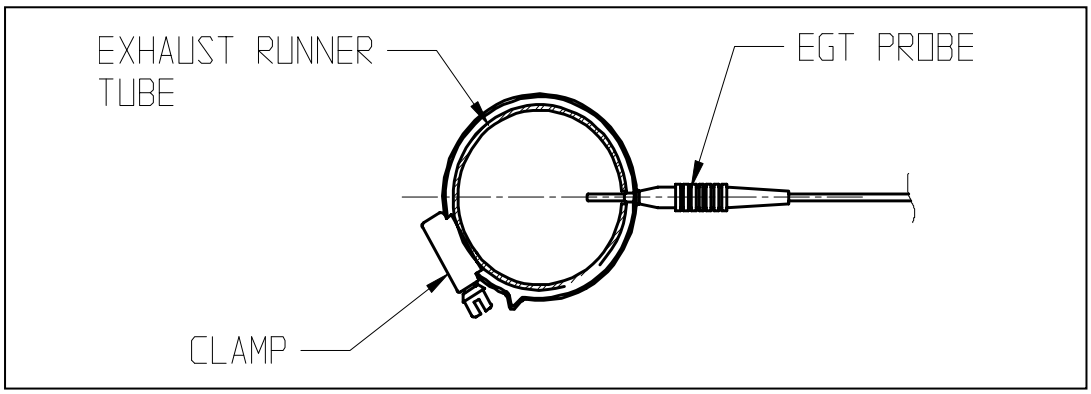

Figure 78 - 6 EGT Probe in Exhaust Runner Tube

a. Loosen clamps and carefully pull the ducting hoses away from the heat exchanger. If ducting is stuck, gently twist the ducting while pulling outward to prevent damage to ducting material (see Figure 78 - 7).

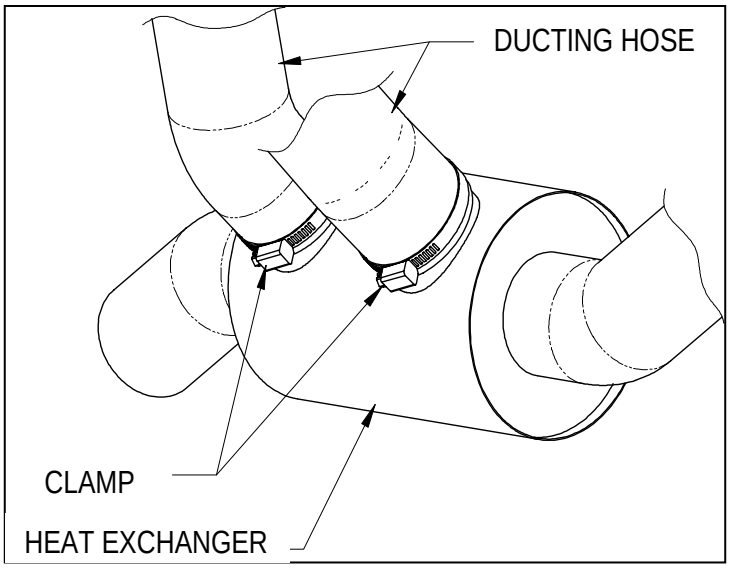

Figure 78 - 7 Heat Exchanger and Ducting

b. Remove the right and left side exhaust hanger clamp attachment bolt, nut, and washer. Pull the silicone straps out of the metal hanger clamps, and remove the clamps from the tailpipes.
c. Remove the nuts and washers attaching the exhaust manifold flange to the cylinder head at all flange locations, and remove the exhaust system from the engine (see Figure 78 - 4). Discard all flange gaskets.
d. Remove the cotter pins, bolts, springs, nuts, and washers connecting the exhaust manifold to the muffler from both the left and right side exhaust (see Figure 78 - 8). Discard all cotter pins.
e. Remove the bolts, nuts, and washers attaching the muffler to the tailpipe D-clamp, and remove the D-clamps from both the left and right side exhaust (see Figure 78 - 2).
f. Remove the expansion D-clamp bolts, nuts, and washers, and remove the D-clamps from both the left and right side middle of the exhaust manifold runners.
g. Remove the middle riser tube from the middle manifold runner tube (see Figure 78 - 9).
h. Remove the four sheet metal screws from the heat exchanger shroud, and remove the shroud. Note the position of the shroud inlet and outlet tubes before removing.

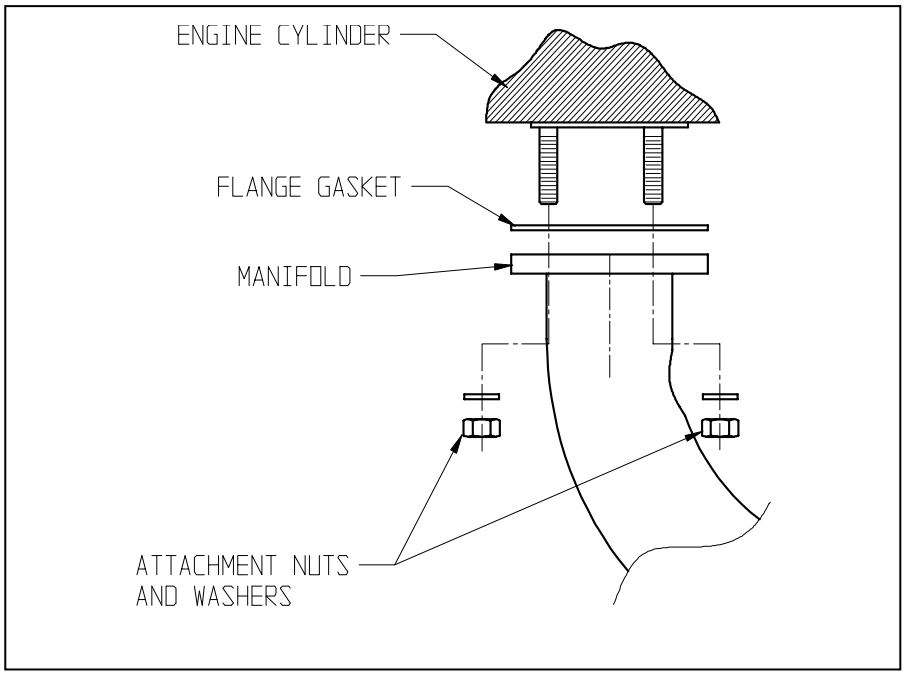

Figure 78 - 8 Manifold Attachment

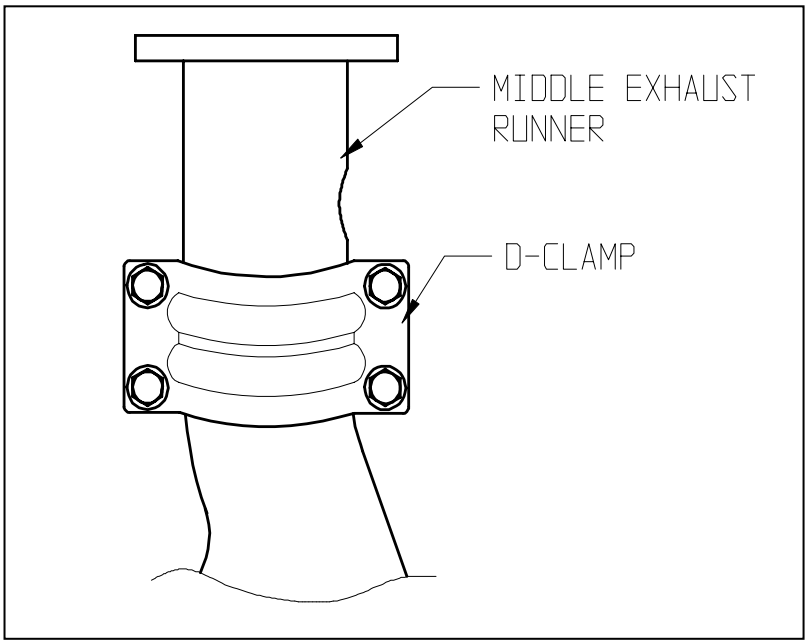

Figure 78 - 9 Exhaust Runner and D-Clamp

78-4. Exhaust System Cleaning

Cleaning the exhaust system is required before inspection. With the exhaust system completely disassembled, wash each component in a mineral spirit or Stoddard solvent solution, and wipe dry with a clean cloth. Light air pressure may also be used to dry parts.

CAUTION

Never use a wire brush, abrasive paper, or abrasive cleaners to remove dirt and carbon deposits from the exhaust system. These products will remove any visual evidence of exhaust leakage that would otherwise render the part unacceptable during inspection.

78-5. Exhaust System Inspection

The entire exhaust system must be thoroughly inspected for cracks, breaks, or burn-throughs that could lead to a possible engine fire, poor engine performance, or poor heat exchanger function. Inspect both the inside and outside exhaust manifold, muffler, heat exchanger, tone cone, and tailpipe locations under good lighting using a 10 X (power) magnifying glass and borescope (recommended) looking for the following imperfections:

a. Gray, red, or black gas stains at all welds, pins, bends, and seams that would indicate a crack or break.
b. Pitting, scaling, or rippling that would indicate burn-through from intense heat and metal expansion/contraction.
c. Chafing of metal at all clamp locations that would indicate a loose or defective connection.
d. Eroding, missing, or deformed heat exchanger transfer pins.
e. Bulging and/or distortion of surfaces that would indicate metal fatigue.
f. Eroding muffler inner baffle components.
g. Broken springs.
h. Bolts that show signs of chafing due to improper alignment.
i. Flange holes that show signs of chafing or elongation due to improper alignment.

Indication of exhaust leakage at the D-clamp connections may be normal due to regular use and clamp design. Determination of the seriousness of any such indications must be made during inspection of the exhaust system.

78-6. Exhaust System Installation

a. Install the left and right side riser tubes into the left and right middle manifold runner tubes. Position the D-clamps over the formed ridges on both the left and right side middle manifold runner and riser tubes. Install bolts, washers, and nuts finger tight in reverse order as disassembled. Do not tighten nuts. Nuts will be tightened after final positioning of exhaust system (see Figure 78 - 9).
b. Install the heat exchanger shroud on the center and over the right-side muffler. Install four sheet metal screws. Torque screws 30 to 36 in.-lbs.
c. Install the right-side heat exchanger/muffler assembly to the right-side exhaust manifold. Install springs, bolts, washers, and nuts finger-tight in reverse order as disassembled. Do

not tighten nuts. Nuts will be tightened after final positioning of the exhaust system (see Figure 78 - 2).

d. Install the left-side muffler to the left-side exhaust manifold. Install springs, bolts, washers, and nuts finger-tight in reverse order as disassembled. Do not tighten nuts. Nuts will be tightened after final positioning of the exhaust system (see Figure 78 - 2).
e. Install the right-side tailpipe to the right-side exhaust assembly. Position D-clamps over the formed ridges in both the muffler and tailpipe. Install bolts, washers, and nuts finger-tight in reverse order as disassembled. Do not tighten nuts. Nuts will be tightened after final positioning of the exhaust system (see Figure 78 - 3).
f. Install the left-side tailpipe to the left-side exhaust assembly. Position D-clamps over the formed ridges in both the muffler and tailpipe. Install bolts, washers, and nuts finger-tight in reverse order as disassembled. Do not tighten nuts. Nuts will be tightened after final positioning of exhaust system (see Figure 78 - 3).
g. Install the left and right-side exhaust assembly to the engine. Install a new flange gasket TCM P/N 652458 between the exhaust manifold flange and cylinder head at all locations. Install nuts and washers at all flange locations (see Figure 78 - 8) and wet torque nuts with 50W aviation oil 100 to 110 in.-lbs. Torque the D-clamps nuts at the left and right-side middle manifold runner tube to the riser tube locations 30 to 36 in.-lbs. (see Figure 78 - 9).
h. Install the metal band clamps over the left and right-side tailpipes. Position the clamps directly under the Teflon hanger straps. Connect the left and right-side metal band clamps to the silicone hanger straps with bolts, nuts, and washers in reverse order as disassembled leaving hanger nuts finger-tight. Place two washers on each bolt between the Teflon strap and the bracket to fill the gap. Position the left and right-side tailpipes so that the outlet end of the pipe is pointing down and away from the airplane fuselage. Ensure a minimum 0.38 in. clearance between the left and right side tailpipe and the face of the exhaust tunnel heat shield on the fuselage. Torque the left and right side hanger clamp nuts 30 to 36 in.-lbs. (see Figure 78 - 4).
i. Rotate the right-side muffler/heat exchanger inlet and outlet tubes to position noted during disassembly. Torque the left and right-side muffler to the tailpipe D-clamp nuts 30 to 36 in.-lbs. (see Figure 78 - 3).
j. Verify that the ball joint connections at the left and right-side manifold to muffler connections are properly aligned, then tighten the ball joint retaining nuts until the cotter pin can be inserted. Install a new cotter pin P/N MS24665-151 at all left and right-side locations per MS33540 (see Figure 78 - 2).
k. Install the EGT probe into the left rear manifold runner tube. Insert the clamp tail into the clamp barrel, and tighten the clamp making sure the EGT probe is fully seated against the runner tube (see Figure 78 - 6).
l. Connect the ducting tubes and clamps over the inlet and outlet heat exchanger tubes in reverse order as disassembled. Tighten ducting clamps (see Figure 78 - 7).

78-7. Exhaust System Testing

a. **Standard Testing**
   1. With the upper and lower cowling removed and the engine running at full operating temperature, check for exhaust leaks at all ball joint, D-clamp, EGT probe, and manifold flange gasket locations.

2. Check that all wires, hoses, and devices are sufficiently located away from all exhaust components. After the exhaust has been run tested, install the upper and lower cowling, and check for interference between any part of the exhaust system and cowling.

b. **Pressure Testing**

1. Every 1000 hours or every 5 years, whichever occurs first, the exhaust should be pressure tested using the following procedure.
   a) Remove the exhaust per instructions in section 78-3.
   b) Completely submerge the part under water with 15 psig ± 2 psig of air pressure inside the part.
   c) Look closely to determine if air bubbles are escaping.
   d) If leaks are found, mark the area with a marker, and return the parts for repair or install new parts.

CHAPTER 79

OIL

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General

NOTE: See Teledyne Continental Motors Permold Series Maintenance Manual Model IO-550-N Form Number X30634A for specific maintenance information. Contact Teledyne Continental Motors to ensure you have the latest revision to their manual. It is the responsibility of the owner or operator of the airplane to maintain the revision schedule for all manuals related to the Teledyne Continental Motors Permold Series Model IO-550-N.

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CHAPTER 80

STARTING

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General

NOTE: See Teledyne Continental Motors Permold Series Maintenance Manual Model IO-550-N Form Number X30634A for specific maintenance information. Contact Teledyne Continental Motors to ensure you have the latest revision to their manual. It is the responsibility of the owner or operator of the airplane to maintain the revision schedule for all manuals related to the Teledyne Continental Motors Permold Series Model IO-550-N.

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