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			22-00-00		CONT.	22-00-01		CONT.
CHAPTER 22 TAB			915	APR 22/01	01	49	FEB 10/91	11
			916	AUG 22/01	01	50	MAY 01/84	05
AUTOFLIGHT			917	APR 22/01	01	51	FEB 10/91	13
			918	APR 22/01	01	52	FEB 10/91	13
EFFECTIVE PAGES			919	APR 22/01	01	53	DEC 22/01	04
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NUMBER OF PAGES			921	APR 22/01	01	55	FEB 10/91	08
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22-CONTENTS			22-00-01			22-00-02		
1	APR 22/07	SAS	1	NOV 10/91	01	1	DEC 22/01	01
2	APR 22/01	SAS	2	APR 22/01	01	2	DEC 22/01	01
3	AUG 10/98	SAS	3	FEB 01/83	01	3	DEC 22/01	21
4	DEC 22/08	SAS	4	AUG 01/82	01	4	DEC 22/01	12
5	DEC 22/02	SAS	5	AUG 01/82	01	5	DEC 22/01	16
6	DEC 22/07	SAS	6	AUG 01/82	01	6	DEC 22/01	19
7	APR 22/01	SAS	7	DEC 22/01	01	7	DEC 22/01	08
8	APR 22/06	SAS	8	FEB 01/83	02	8	FEB 10/92	09
9	DEC 22/06	SAS	9	DEC 22/01	01	9	DEC 22/99	01
10	AUG 22/04	SAS	10	DEC 22/01	01	10	DEC 22/01	06
11	APR 22/06	SAS	11	AUG 22/01	01	11	NOV 10/89	07
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			13	DEC 22/01	03	13	AUG 10/89	07
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2	NOV 01/86	02	16	NOV 10/88	06	16	DEC 22/99	20
3	DEC 22/01	15	17	FEB 10/89	11	17	AUG 10/90	06
4	DEC 22/01	11	18	AUG 10/91	03	18	AUG 10/90	04
5	FEB 10/93	07	19	FEB 10/89	03	19	MAY 10/88	03
6	DEC 22/99	19	20	AUG 10/89	09	20	MAY 10/88	03
7	MAY 01/87	01	21	NOV 01/83	03			
8	NOV 10/89	19	22	FEB 10/89	03	22-00-02		
9	DEC 22/01	18	23	AUG 10/88	04	201	AUG 22/99	01A
10	APR 22/02	21	24	FEB 10/89	05	202	APR 22/02	18A
11	AUG 22/05	11	25	MAY 01/84	01	203	AUG 22/99	03A
12	APR 22/02	04	26	FEB 10/89	04	204	AUG 22/99	03A
13	APR 22/02	05	27	AUG 01/82	01	205	DEC 22/99	03A
14	AUG 22/05	07	28	FEB 01/83	02	206	DEC 22/08	11A
15	APR 22/03	01	29	AUG 10/92	04	207	FEB 10/97	41A
16	APR 22/02	01	30	AUG 10/88	04	208	DEC 22/05	06A
17	APR 22/02	01	31	DEC 22/01	08	209	DEC 22/05	19A
18	AUG 22/05	06	32	DEC 22/01	06	210	DEC 22/05	16A
			33	DEC 22/01	03	211	DEC 22/05	01A
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903	APR 22/01	01	37	DEC 22/01	13	215	DEC 22/05	24A
904	APR 22/01	01	38	DEC 22/01	08	216	DEC 22/05	01A
905	APR 22/01	01	39	FEB 10/91	10	217	DEC 22/05	24A
906	APR 22/01	01	40	AUG 10/88	07	218	DEC 22/01	16A
907	APR 22/01	01	41	FEB 10/91	09	219	APR 22/01	30A
908	APR 22/01	01	42	FEB 10/91	12	220	DEC 22/00	29A
909	APR 22/01	01	43	AUG 10/89	10	221	DEC 22/05	24A
910	APR 22/01	01	44	FEB 10/91	21	222	DEC 22/05	30A
911	APR 22/01	01	45	FEB 10/91	19	223	DEC 22/05	25A
912	APR 22/01	01	46	FEB 10/91	20	224	DEC 22/05	21A
913	APR 22/01	01	47	FEB 10/91	15	225	DEC 22/05	24A
914	APR 22/01	01	48	MAY 01/84	07	226	DEC 22/05	25A

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231	DEC 22/05	23A
232	DEC 22/05	24A
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234	DEC 22/05	21A
235	DEC 22/05	21A
236	DEC 22/05	20A
237	DEC 22/05	22A
238	DEC 22/05	19A
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3	DEC 22/01	16
4	AUG 10/88	04
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6	DEC 22/01	19
7	APR 22/01	19
8	DEC 22/01	07
9	AUG 22/99	08
10	AUG 22/99	05
11	NOV 10/90	13
12	DEC 22/99	20
13	DEC 22/01	17
14	DEC 22/01	07
15	DEC 22/01	06
16	DEC 22/01	20
17	AUG 22/99	22
18	NOV 10/90	05
19	FEB 10/92	32
20	DEC 22/01	13
21	NOV 10/91	29
22	NOV 10/91	29
23	AUG 22/00	11
24	DEC 22/99	14
25	AUG 22/99	14
26	DEC 22/99	26
27	DEC 22/99	26
28	DEC 22/02	19
29	APR 22/01	22
30	AUG 22/01	24
31	AUG 22/99	26
32	AUG 22/99	22
33	AUG 22/99	27
34	DEC 10/98	45
35	DEC 22/01	14
36	DEC 22/01	13
37	APR 22/02	12
38	AUG 22/99	02
39	DEC 22/01	08
40	DEC 22/08	20
41	DEC 22/08	20
42	AUG 10/97	35
43	DEC 22/08	25
44	DEC 22/08	20

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506	AUG 22/05	06
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511	AUG 10/95	23
512	AUG 10/95	16
513	DEC 22/01	05
514	AUG 22/01	12
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2	NOV 10/88	04
3	DEC 22/99	02
4	DEC 22/06	12
5	DEC 22/01	17
6	DEC 22/01	02
7	MAY 10/92	06
8	MAY 10/92	14
9	DEC 22/01	04
10	DEC 22/01	14
11	DEC 22/01	18
12	DEC 22/01	23
13	DEC 22/01	20
14	DEC 22/01	01
15	AUG 01/84	02
16	AUG 22/07	03
17	NOV 10/90	07
18	MAY 10/89	08
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21	DEC 22/01	09
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405	AUG 22/01	01
406	AUG 22/01	01
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402	MAY 01/84	01
403	MAY 10/95	01
404	MAY 10/90	01
22-12-00		
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3	AUG 22/04	09
4	AUG 22/05	03
5	AUG 22/01	10
6	AUG 10/88	03
7	FEB 10/91	19
8	NOV 10/90	03
9	FEB 10/89	09
10	FEB 10/89	12
11	MAY 10/90	07
12	NOV 10/90	14
13	AUG 22/01	06
14	FEB 10/92	18
15	DEC 22/01	21
16	DEC 22/01	01
17	NOV 10/88	04
18	FEB 10/92	26
19	DEC 22/01	29
20	DEC 22/01	22
21	DEC 22/01	22
22	AUG 22/05	15
23	AUG 22/05	11
24	DEC 22/01	15

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27	AUG 22/05	15	407	MAY 10/90	01	45	AUG 22/07	12
28	AUG 22/05	15	408	MAY 10/90	01	46	AUG 22/07	17
29	AUG 22/05	16						
30	DEC 22/01	14	22-12-04			22-13-00		
31	AUG 22/05	19	401	FEB 10/91	16	101	AUG 22/01	02
32	AUG 22/05	18	402	MAY 10/90	27	102	MAY 10/94	02
33	AUG 22/05	15	403	FEB 10/91	24			
34	DEC 22/01	08	404	APR 22/06	16	22-13-01		
35	AUG 22/05	15	405	APR 22/06	06	401	AUG 10/93	01
36	DEC 22/01	13	406	APR 22/01	16	402	AUG 10/93	03
37	AUG 22/05	19	407	APR 22/01	18	403	FEB 10/90	08
38	APR 22/99	18	408	BLANK		404	AUG 22/01	14
39	NOV 10/90	23				405	APR 22/01	06
40	AUG 22/05	18	22-13-00			406	MAY 10/97	15
41	DEC 22/01	25	1	NOV 01/86	02	407	AUG 10/93	07
42	DEC 22/01	24	2	NOV 10/88	04	408	AUG 10/93	07
43	AUG 10/96	42	3	DEC 22/01	04	409	AUG 10/93	07
44	AUG 10/96	37	4	DEC 22/01	06	410	AUG 10/93	07
			5	AUG 22/07	04	411	AUG 10/93	07
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102	MAY 10/94	10	8	DEC 22/01	06	22-13-02		
			9	AUG 22/07	06	401	AUG 10/96	01
22-12-01			10	AUG 22/07	06	402	FEB 10/91	01
401	AUG 22/04	02	11	AUG 22/07	07	403	FEB 10/91	02
402	AUG 10/95	08	12	AUG 22/07	10	404	APR 22/06	01
403	NOV 10/89	26	13	NOV 10/90	02	405	FEB 10/91	02
404	APR 22/01	02	14	AUG 22/07	03	406	FEB 10/91	01
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410	APR 22/01	12	20	AUG 22/07	11	402	APR 22/06	01
411	AUG 22/07	06	21	FEB 10/92	21	403	FEB 10/91	01
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404	APR 22/06	01	32	AUG 22/07	14	406	FEB 10/91	18
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11	DEC 22/01	17	31	DEC 22/01	22	405	AUG 22/01	02
12	DEC 22/01	04	32	DEC 22/01	16	406	MAY 10/95	04
13	DEC 22/01	05	33	DEC 22/01	11	407	MAY 10/95	04
14	DEC 22/01	10	34	DEC 22/01	15	408	NOV 10/95	04
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4	NOV 10/89	06	516	FEB 10/93	13	406	APR 22/06	07
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9	NOV 10/93	12	521	APR 10/98	15	403	APR 22/00	02
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25	AUG 10/98	11				11	DEC 22/01	02
26	AUG 10/92	09				12	DEC 22/01	02

R = REVISED, A = ADDED OR D = DELETED
F = FOLDOUT PAGE

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13	APR 22/99	16
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7	DEC 22/03	17
8	DEC 22/03	01
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11	APR 22/00	07
12	APR 22/00	13
13	APR 22/00	01
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16	APR 22/02	11
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13	DEC 22/99	19
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23	AUG 22/05	23
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38	AUG 22/05	21
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40	AUG 22/05	24
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42	AUG 22/05	22
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102	AUG 10/90	07
22-32-00		
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502	AUG 10/98	08
503	AUG 10/98	02
504	AUG 10/98	17
505	AUG 22/99	20
506	AUG 10/98	18
507	APR 22/07	20
508	AUG 10/98	26
509	AUG 22/07	24
510	AUG 22/07	16
511	AUG 22/07	15
512	AUG 22/07	12
513	AUG 10/98	08
514	BLANK	
22-32-01		
401	APR 22/01	01
402	AUG 10/94	14
403	AUG 22/07	22
404	AUG 22/01	19
405	APR 22/01	19
406	APR 22/01	17
407	APR 22/01	10
408	BLANK	
22-32-02		
401	APR 22/00	01
402	APR 22/99	01
403	APR 22/00	01
404	APR 22/00	01
22-32-03		
401	DEC 22/01	03
402	AUG 22/99	01
403	DEC 22/01	01
404	DEC 22/00	01

Scandinavian Airlines System

PAGE	DATE	CODE	PAGE	DATE	CODE	PAGE	DATE	CODE
22-32-04			22-41-00		CONT.	22-41-00		CONT.
201	AUG 22/01	05	5	AUG 22/99	01	63	MAY 10/96	10
202	APR 22/02	09	6	APR 22/07	01	64	BLANK	
203	AUG 22/01	13	7	AUG 22/99	01			
204	APR 22/02	15	8	AUG 22/99	01	22-41-00		
205	MAY 10/97	17	9	AUG 22/99	01	101	FEB 10/94	02
206	MAY 10/97	15	10	AUG 22/99	01	102	MAY 10/95	03
207	MAY 10/97	15	11	AUG 22/99	01	103	NOV 10/93	03
208	MAY 10/93	17	12	AUG 22/99	01	104	BLANK	
209	APR 22/02	10	13	AUG 22/99	01			
210	APR 22/06	01	14	AUG 22/99	23	22-41-00		
211	MAY 10/97	15	15	DEC 22/01	02	501	DEC 22/00	01
212	APR 22/02	13	16	DEC 22/01	01	502	DEC 22/00	22
213	APR 22/05	12	17	DEC 22/01	01	503	APR 22/99	03
214	AUG 22/01	23	18	AUG 22/99	01	504	NOV 10/97	30
215	APR 22/99	17	19	AUG 22/99	01	505	AUG 22/01	03
216	AUG 22/01	08	20	AUG 22/99	05	506	AUG 22/01	25
217	APR 22/99	12	21	DEC 22/04	19			
218	APR 22/99	18	22	DEC 22/01	01	22-41-01		
219	APR 22/05	14	23	MAY 10/96	01	401	AUG 10/90	01
220	APR 22/02	18	24	APR 22/08	02	402	DEC 22/01	01
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22-32-07			28	MAY 10/96	01			
401	AUG 22/07	15	29	MAY 10/96	02			
402	AUG 22/01	15	30	APR 22/06	05			
403	AUG 10/97	14	31	MAY 10/96	06			
404	FEB 10/93	11	32	MAY 10/96	02			
405	AUG 10/97	13	33	MAY 10/96	01			
406	NOV 10/93	11	34	DEC 22/99	06			
407	AUG 10/97	13	35	APR 22/01	07			
408	AUG 22/01	14	36	MAY 10/96	05			
409	DEC 10/98	15	37	DEC 22/99	08			
410	DEC 10/98	15	38	MAY 10/96	01			
411	AUG 22/07	13	39	MAY 10/96	03			
412	AUG 22/07	05	40	MAY 10/96	01			
413	DEC 22/03	05	41	MAY 10/96	01			
414	DEC 22/03	05	42	MAY 10/96	01			
415	AUG 22/07	06	43	MAY 10/96	01			
416	AUG 22/07	07	44	MAY 10/96	01			
417	AUG 22/07	06	45	DEC 22/99	01			
418	DEC 22/03	05	46	DEC 22/01	01			
419	DEC 22/03	05	47	MAY 10/96	06			
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			49	MAY 10/96	12			
22-34-00			50	MAY 10/96	14			
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2	AUG 10/96	01	52	MAY 10/96	19			
			53	APR 22/01	20			
22-34-00			54	DEC 22/01	13			
101	FEB 10/95	01	55	MAY 10/96	23			
102	FEB 10/95	01	56	APR 22/03	13			
			57	DEC 22/01	02			
22-41-00			58	MAY 10/96	08			
1	MAY 10/96	02	59	DEC 22/01	02			
2	MAY 01/85	04	60	AUG 22/00	12			
3	AUG 22/99	01	61	DEC 22/01	02			
4	AUG 22/99	01	62	DEC 22/01	08			

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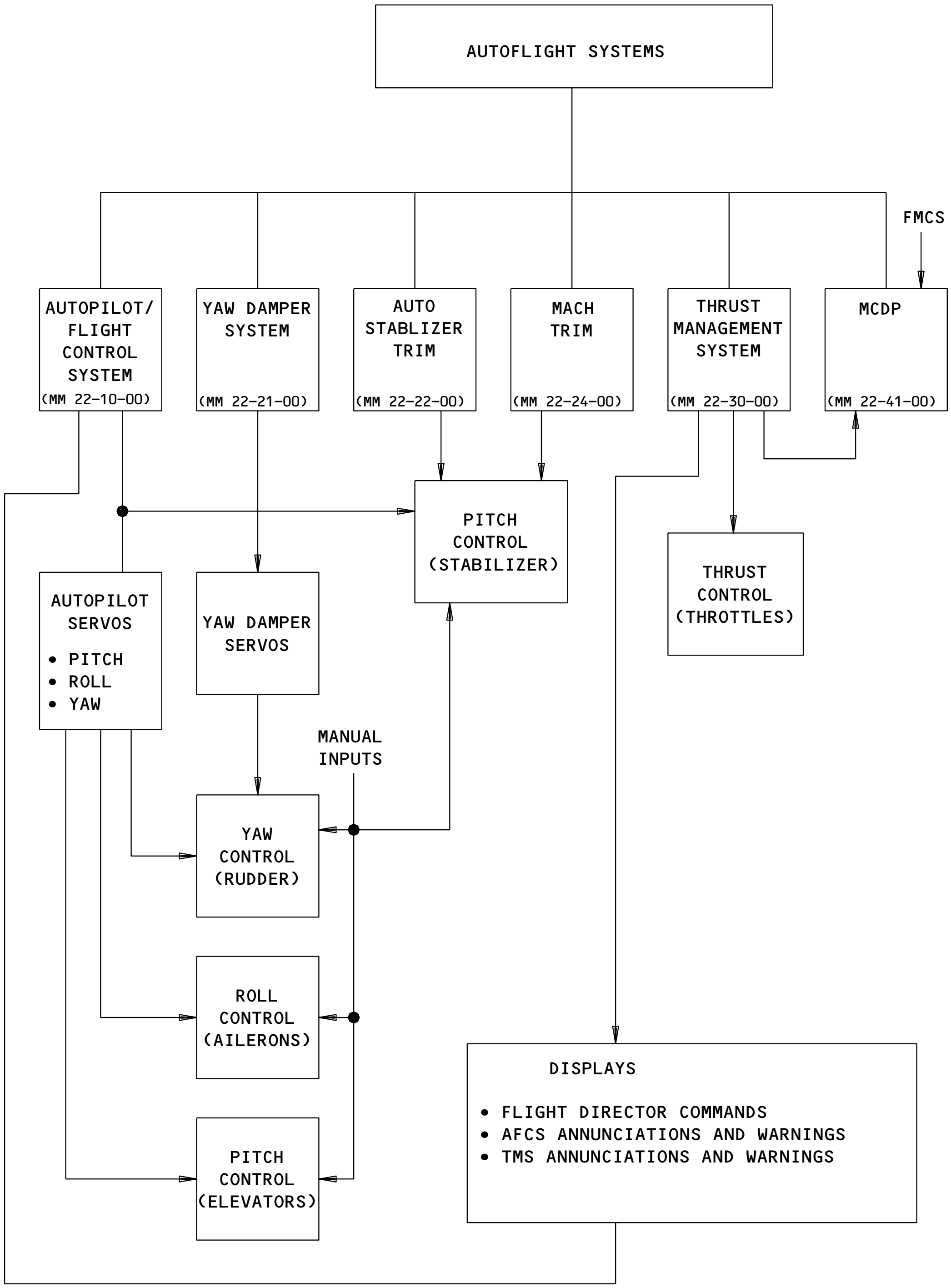
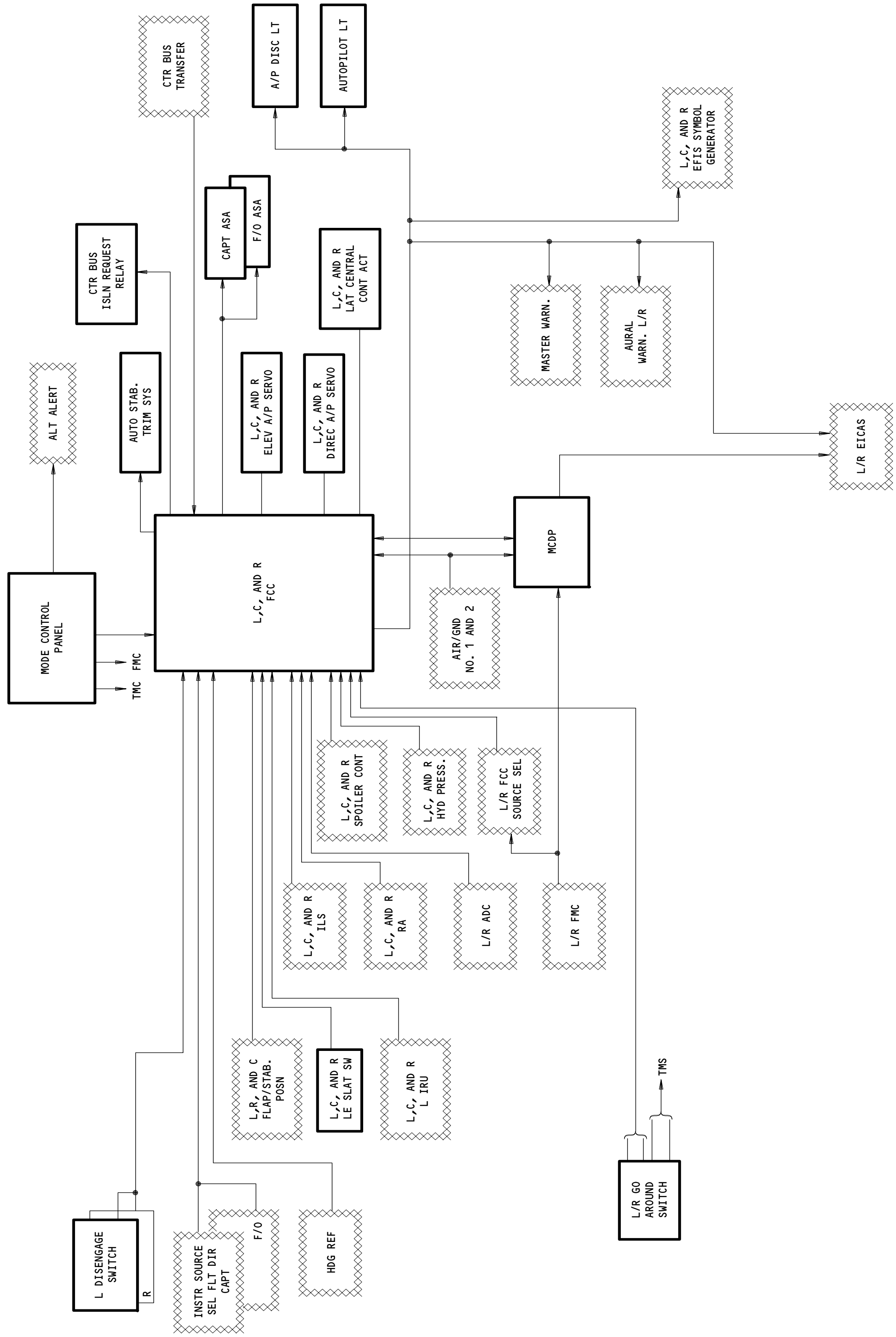
AUTOFLIGHT - GENERAL - DESCRIPTION AND OPERATION

1. General
   A. Autoflight Systems (Fig. 1)
      (1) The Autopilot/Flight Director System provides automatic control of the ailerons, elevator, stabilizer, and rudder control systems for flying selected flight modes. It provides pitch and roll flight director commands, system warnings, and mode annunciations. Rudder directional control is provided during the landing rollout.
      (2) The Yaw Damper System provides automatic yaw damping to the rudder to minimize oscillations due to Dutch roll and gust induced side slip. It also provides turn coordination.
      (3) The Automatic Stabilizer Trim System uses flight control computer commands to the Stabilizer Trim Aileron Lockout Module to provide trim commands to the stabilizer. This system is only active when an autopilot is engaged.
      (4) The Mach Trim System improves the airplanes stability by trimming the stabilizer as airspeed changes. Electric, manual, or automatic trimming inhibits the Mach Trim System.
      (5) The Thrust Management System provides automatic thrust control of the engines based on selected modes, current conditions, and engine limits.
      (6) The Maintenance Monitoring System provides centralized recording of flight faults from the FCC's, TMC, and FMC's. It also allows maintenance personnel to perform programmed tests.
   B. Autopilot/Flight Director System (AFDS) (Fig. 2)
      (1) The AFDS consists of three separate autopilot systems that use a single mode control panel. Each autopilot channel consists of a Flight Control Computer (FCC) with associated servos and displays. Each FCC contains all the logic and signal handling circuitry for pitch, roll, and yaw axis control. The AFDS has autoland approach capability when two or three autopilot channels are engaged. Normal climb, cruise, descent, and approach are available with any one channel engaged. The airplane is not certified for autopilot takeoff. However the flight director is available for takeoff guidance.
      (2) The Autopilot/Flight Director System consists of the following subsystems:
         (a) Autopilot (Flight Control) (Ref 22-10-00) provides a general description and operation of the autopilot system. This section covers all the autopilot modes.



Autoflight Systems
Figure 1



Autopilot/Flight Director System
Figure 2

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(b) Autopilot/Flight Director Power (22-11-00) covers the general description of ac and dc power distribution to components common to the autopilot/flight director system. It also covers the component details and operation of the three flight control computers, Autoflight Control System (AFCS) mode control panel, autopilot disengage switches, and AFCS go-around switches.

(c) Autopilot/Flight Director Pitch Channel (22-12-00) covers the general description of the autopilot/flight director pitch channel. It gives the component details and operation of the Elevator Autopilot Servo (EAS) and EAS electrohydraulic servovalve and solenoid valves.

(d) Autopilot/Flight Director Roll and Yaw Channel (22-13-00) covers the general description of the autopilot/flight director roll and yaw channel. It gives the component details and operation of the Directional Autopilot Servos (DASs), Lateral Central Control Actuators (LCCAs), and DAS and LCCA electroyhydraulic servovalve and solenoid valves.

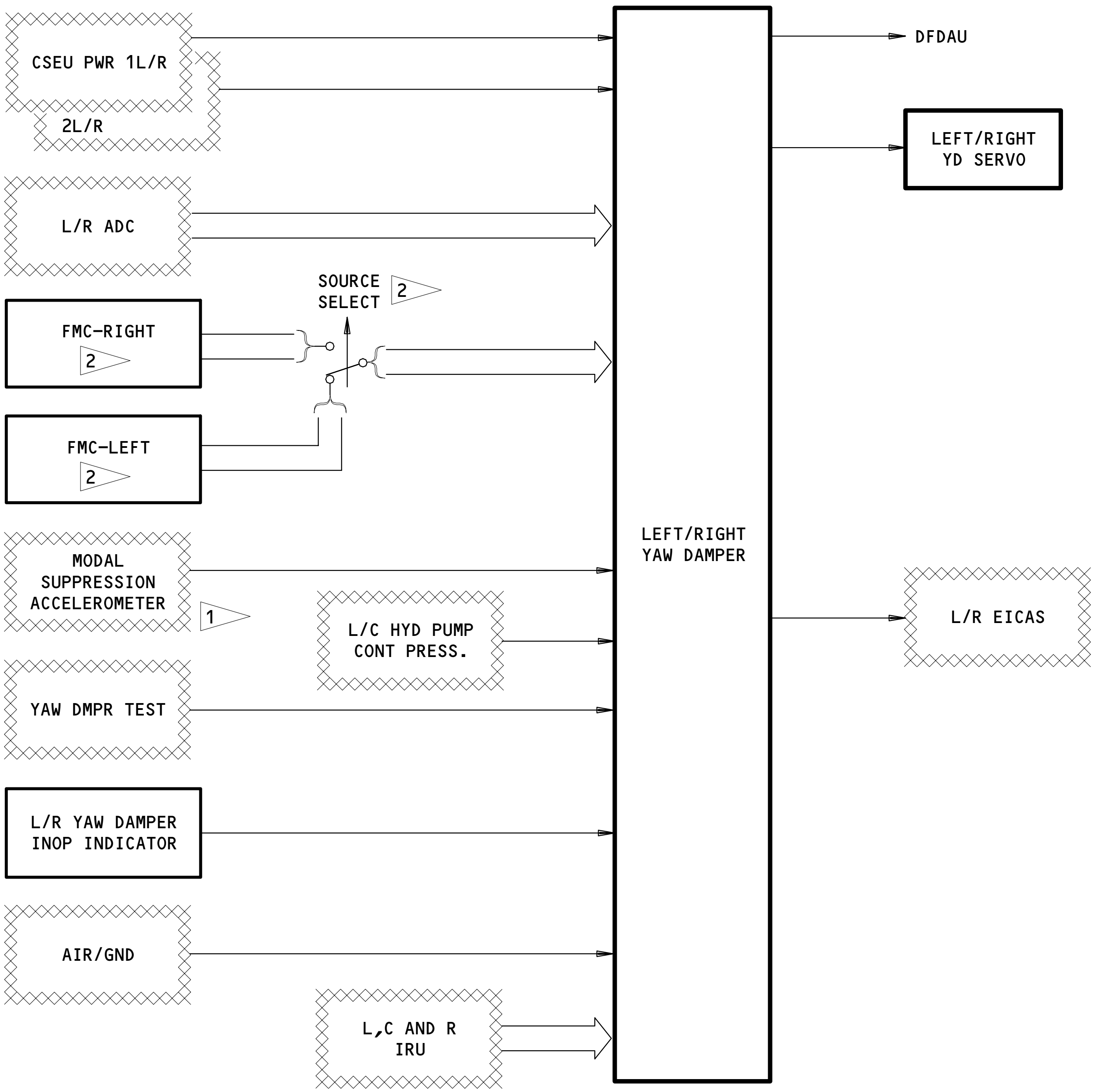
(e) Autopilot/Flight Director Warning and Annunciation (22-14-00) covers the description of the autopilot/flight director warning and annunciation. It covers the component details and operation of the autoland status annunciators, autopilot caution and disconnect lights, and Engine Indication and Crew Alerting System (EICAS) autopilot displays.

(f) Autopilot/Flight Director Interchannel Data (22-15-00) covers the general description and operation of cross channel data buses. Digital data is transmitted between the three flight control computers.

C. Yaw Damper System (Ref 22-21-00) (Fig. 3)

(1) 767-200;

the yaw damper system provides commands to the rudder that dampen undesirable side slip and provide coordination for normal turns. The system consists of a control panel with status indication, two yaw damper servos, and two yaw damper modules. The yaw damper modules receive data from the Air Data Computers (ADCs) and Inertial Reference Units (IRUs). The modules also receive on-off commands from the control panel. The modules sent status information to the control panel, and electrical control commands to the two yaw damper servos. The servos translate the electrical commands to mechanical outputs which drive the rudder system.



1 767-300 AIRPLANES
2 767-300 AIRPLANES WITH -125 AND SUBSEQUENT YDM

Yaw Damper System
Figure 3

(2) 767-300;
the yaw damper system provides commands to the rudder that dampen undesirable side slip and provide coordination for normal turns. The system consists of a control panel with status indication, two yaw damper servos, and two yaw damper modules. The yaw damper modules receive data from the Air Data Computers (ADCs), Inertial Reference Units (IRUs), Modal Suppression Accelerometer, and Flight Management Computers (FMCs) *[1]. The modules also receive on-off commands from the control panel. The modules send status information to the control panel, and electrical control commands to the two yaw damper servos. The servos translate the electrical commands to mechanical outputs which drive the rudder system.

*[1] 767-300 airplanes with -125 and subsequent yaw damper modules.

(3) 767-200;
this section covers the general description of the yaw damper system. It gives the component details and operation of the yaw damper panel, yaw damper modules, Yaw Damper Servos (YDSs), and YDS electrohydraulic servovalve and solenoid valves.

(4) 767-300;
this section covers the general description of the yaw damper system. It gives the component details and operation of the yaw damper panel, yaw damper modules, Yaw Damper Servos (YDSs), YDS electrohydraulic servovalve and solenoid valves, and modal suppression accelerometers.

D. Automatic Stabilizer Trim System (Fig. 4)

(1) The automatic stabilizer trim system (Ref 22-22-00) uses commands from the FCCs to trim the stabilizer. The stabilizer is trimmed to reduce elevator loads. The automatic trim mode is only active when the autopilot is engaged.

(2) This section covers the general description and operation of the automatic stabilizer trim system.

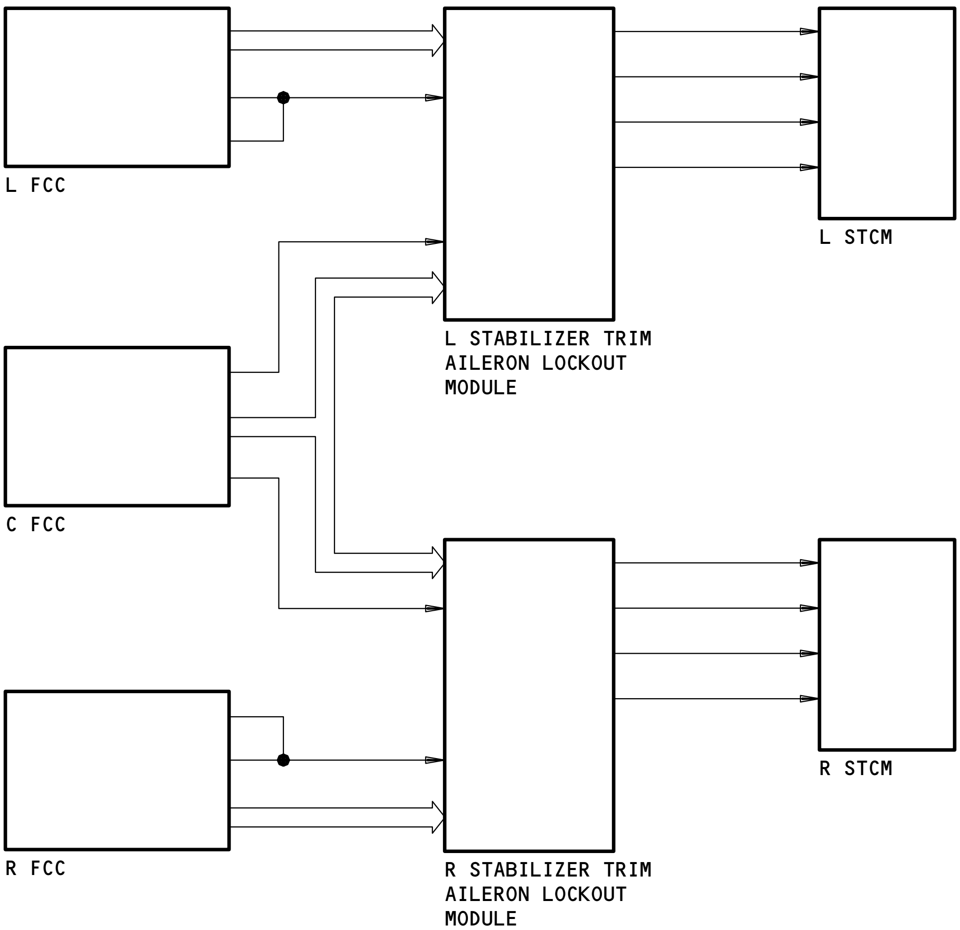
E. Mach Trim System (Fig. 5)

(1) The Mach Trim System (Ref 22-24-00) is a software mode of the Stabilizer Trim Aileron Lockout Module (SAM). It uses Air Data Computer Airspeed/Mach data to provide trimming commands to the stabilizer. This improves the airplanes speed stability as airspeed changes. The Mach Trim System is active when electric, manual, and automatic trimming commands are not present.

(2) This section covers the general description and operation of the Mach Trim System.

F. Thrust Management System (Fig. 6)

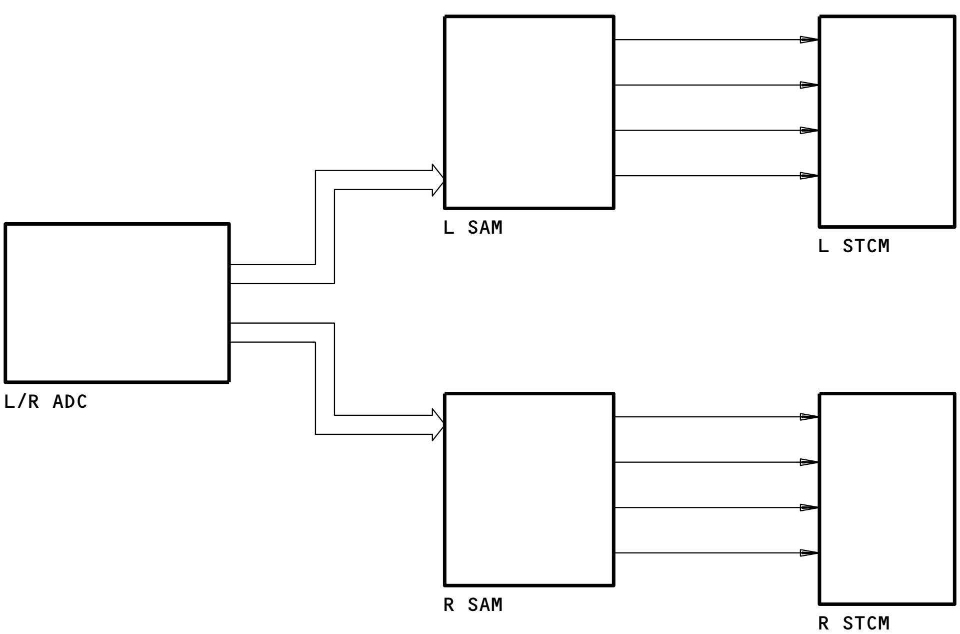
(1) The Thrust Management System (TMS) controls the engine thrust settings during takeoff, cruise and landing (full flight regime). The TMS functions include thrust limit computation, autothrottle control, and information display. Engine thrust limits are computed by the Thrust Management Computer (TMC) based on the selected mode on the Thrust Mode Select Panel (TMSP). The TMS can be operated independent of, or coordinated with the AFDS and FMCS.



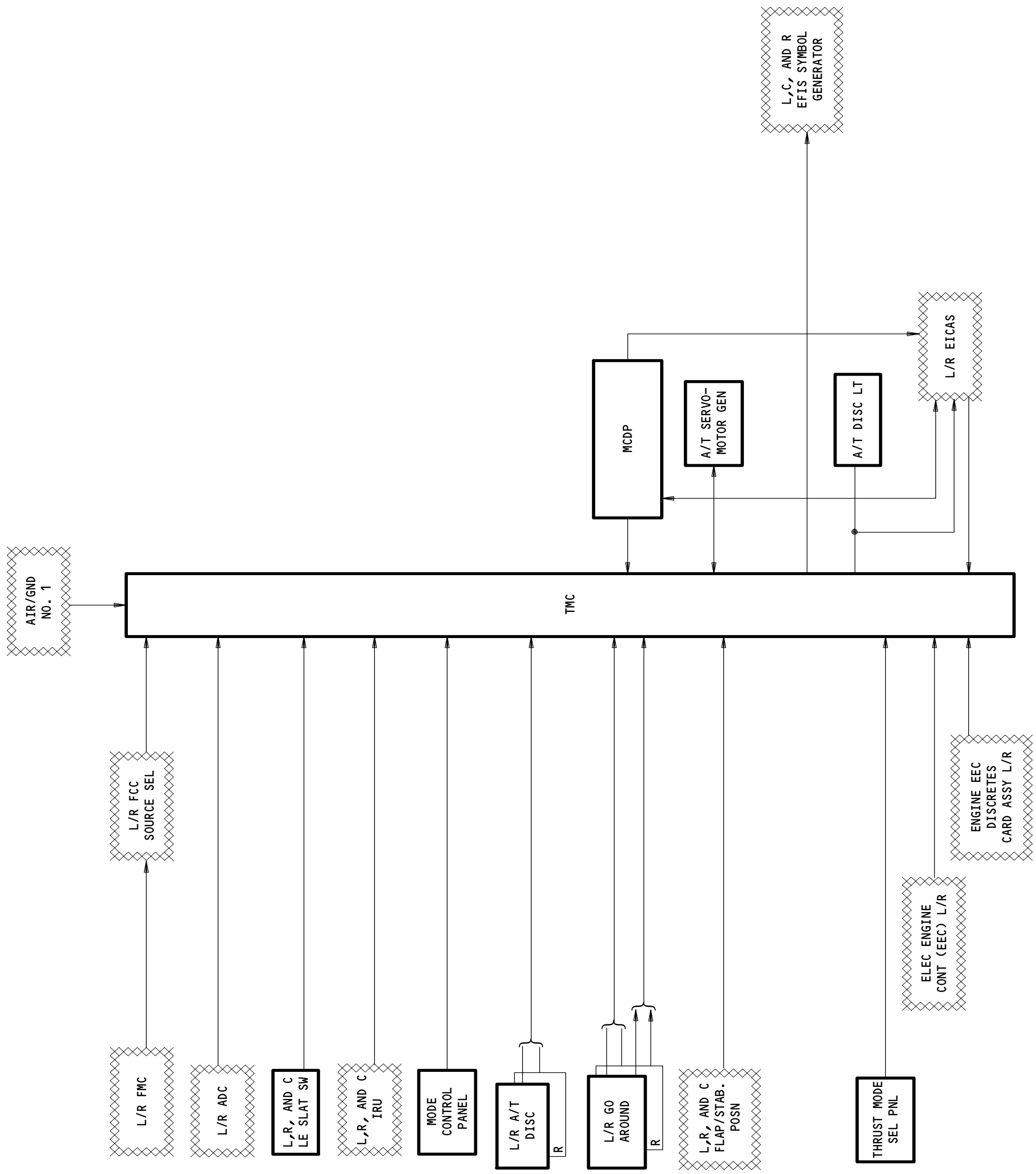
Automatic Stabilizer Trim System
Figure 4

(2) The thrust management system consists of the following subsystems:
(a) Thrust Management Power (Ref 22-31-00) covers the general description of the ac and dc power distribution and interface components common to the TMS. It covers the component detail and operation of the TMC and TMSP.
(b) Thrust Management System (Ref 22-32-00) covers the general description of the TMS. It covers the component details and operation of the autothrottle servomotor generator, disengage switches, leading edge slat switch, microswitch pack, and brake pack assembly.
(c) Thrust Management Warning and Annunciation (Ref 22-34-00) covers the general description and operation of the thrust management system warning and annunciation.

G. Maintenance Monitor (Ref 22-41-00) (Fig. 7)
(1) The maintenance monitor system consists of a Maintenance Control Display Panel (MCDP). It provides a centralized record of flight faults from the flight control computers, thrust management computer, flight management computers, and their related sensors. It allows maintenance crews to perform ground tests on the AFDS, TMS, and their related sensors without additional ground test equipment.
(2) This section covers the general description, component details, and operation of the MCDP.



Mach Trim System
Figure 5



Thrust Management System
Figure 6

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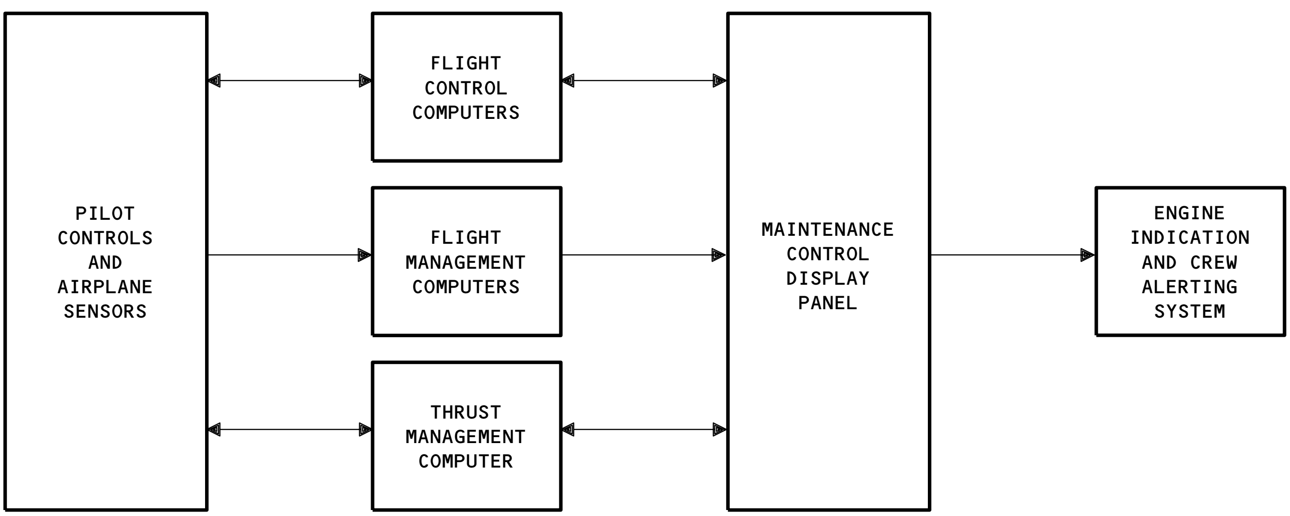
2. Abbreviations

A. The following list contains abbreviations commonly used in this chapter.

ACCRONYM	DEFINITION
A/D	Analog to Digital
ACMP	Alternating Current Motor Pump
ADC	Air Data Computer
ADCS	Air Data Computing System
ADF	Automatic Directing Finder
ADP	Air Driven Pump
AFCS	Autoflight Control System
AFDS	Autopilot/Flight Director System
AIDS	Airborne Integrated Data System



Maintenance Monitor System
Figure 7

ACCRONYM	DEFINITION
AIL	Aileron
ALT	Altitude or Altimeter
AOA or ALPHA	Angle of Attack
A/P	Autopilot
APP	Approach
APU	Auxiliary Power Unit
ARINC	Aeronautical Radio Inc.
ARR	Arrival
ASA	Autoland Status Annunciator
A/T	Autothrottle
ATC	Air Traffic Control
B/A	Bank Angle
BAT	Battery
BCD	Binary Coded Decimal
B/CRS	Back Course
BIT	Built-In-Test
BITE	Built-In-Test-Equipment
BNR	Binary Numerical Representation

ACCRONYM	DEFINITION
CAA	Civil Aviation Agency
CAS	Computed Airspeed
CDU	Control Display Unit
CFT	Control Force Transducer
CLB	Climb
CMD	Command
CON	Continuous
CPU	Central Processing Unit
CRS	Course
CRT	Cathode Ray Tube
CRZ	Cruise
CSEU	Control System Electronics Unit
CWS	Control Wheel Steering
D/A	Digital to Analog
DAS	Directional Autopilot Servo
DADC	Digital Air Data Computer
DDM	Difference in Depth of Modulation
DEP	Departure
DES	Descent

ACCRONYM	DEFINITION
DFDAU	Digital Flight Data Acquisition Unit
DH	Decision Height
DITS	Digital Information Transfer System
DME	Distance Measuring Equipment
DPC	Display Processor Computer
D-TO	Derated Takeoff
EADI	Electronic Attitude Director Indicator
EAROM	Electrically Alterable ROM
EAS	Elevator Autopilot Servo
ECS	Environmental Control System
EDHP	Engine Driven Hydraulic Pump
E/E	Electrical/Electronics
EEC	Electronic Engine Control
EFCU	Elevator Feel and Centering Unit
EFIS	Electronic Flight Instrument System
EGT	Exhaust Gas Temperature
EHSI	Electronic Horizontal Situation Indicator
EHSV	Electrohydraulic Servovalve
EICAS	Engine Indication and Crew Alerting System
EMHP	Electric Motor Hydraulic Pump
EMI	Electromagnetic Interference
ENG or ENGA	Engage

ACCRONYM	DEFINITION
EPCS	Electronic Propulsion Control System
EPR	Engine Pressure Ratio
FAA	Federal Aviation Administration
FCC	Flight Control Computer
F/D	Flight Director
F/E	Flight Engineer
FLCH	Flight Level Change
FMA	Flight Mode Annunciator
FMC	Flight Management Computer
FMCS	Flight Management Computer System
FMS	Flight Management System
F/O	First Officer
FPM	Feet Per Minute
FSEU	Flap/Slat Electronic Unit
FSPM	Flap/Stabilizer Position Module
G/A or GA	Go Around
G/S	Glide Slope
G.S.	Gain Schedule

ACCRONYM	DEFINITION
HDG	Heading
HLD	Hold
HR	Height, Radio
HYD	Hydraulic
IAS	Indicated Airspeed
ILS	Instrument Landing System
INTC	Intercept
I/O	Input/Output
IRU	Inertial Reference Unit
IRS	Inertial Reference System
LCCA	Lateral Central Control Actuator
LNAV	Lateral Navigation
LOC	Localizer
LRU	Line Replaceable Unit
LVDT	Linear Variable Differential Transducer
M	Mach
M/ASI	Mach/Airspeed Indicator
MB	Millibars
MCDP	Maintenance Control Display Panel
MCHENG	Multi-Channel Engage
MCP	Mode Control Panel
MCU	Modular Concept Unit
MMO	Maximum Operating Mach
MMR	Multi-Mode Receiver

ACCRONYM	DEFINITION
MSL	Mean Sea Level
NAV	Navigation
NCD	No Computed Data
N1	Percentage of the Defined 100% Fan Rotational Speed
PCA	Power Control Actuator
PDU	Power Drive Unit
PFD	Primary Flight Display
PLA	Power Lever Angle
PROG	Program
PROM	Programmable Read Only Memory
RA	Radio Altitude or Radio Altimeter
RAT	Ram Air Turbine
RCVR	Receiver
RDMI	Radio Distance Magnetic Indicator
REF	Reference
RF	Radio Frequency
RMI	Radio Magnetic Indicator
RNAV	Area Navigation
ROM	Read Only Memory
RTE	Route
RTG	Rating

ACCRONYM	DEFINITION
RVDT	Rotary Variable Differential Transducer
RVR	Runway Visual Range
SAM	Stabilizer Trim and Aileron Lockout Module
SAT	Static Air Temperature
SCM	Spoiler Control Module
SG	Symbol Generator
SID	Standard Instrument Departure
SOV	Shut-Off Valve
SPD	Speed
SSM	Sign Status Matrix
STAB	Stabilizer
STAR	Standard Terminal Arrival Route
STCM	Stabilizer Trim Control Module
STLM	Stabilizer Trim Limit Switch and Position Transmitter Module
TAS	True Airspeed
TAT	Total Air Temperature
TCA	Terminal Control Area
TGT	Target
TLA	Throttle Lever Angle
TMC	Thrust Management Computer
TMS	Thrust Management System

ACCRONYM	DEFINITION
TMSP	Thrust Mode Select Panel
TO or T/O	Takeoff
TOC	Top of Climb
TOD	Top of Decent
TRA	Thrust Resolver Assembly
TRK	Track
TRU	Transformer/Rectifier Unit
VFR	Visual Flight Rules
VHF	Very High Frequency
VMO (Vmo)	Maximum Operating Airspeed (Knots)
VNAV	Vertical Navigation
VOR	VHF Omnidirectional Range
V/S	Vertical Speed
XFMR	Transformer
XMTR	Transmitter
XPNDR	Transponder
Y/D	Yaw Damper
YDS	Yaw Damper Servo
YSM	Yaw Damper/Stabilizer Trim Module

AUTOFLIGHT - DDG MAINTENANCE PROCEDURES

1. General
   A. This procedure contains the maintenance tasks that are necessary to operate the airplane in various non-standard configurations as allowed by the Master Minimum Equipment List. The procedure also contains the maintenance tasks that put the airplane back to its usual condition.
   B. These are the maintenance tasks for the components in the Autoflight System.
      (1) DDG 22-10-1 Preparation - Autopilot Flight Control Computers (FCC) Inoperative
      (2) DDG 22-10-1 Restoration - Autopilot Flight Control Computers (FCC) Inoperative
      (3) DDG 22-10-1 Preparation - Autopilot Electrohydraulic Servos (Elevator Autopilot Servos (EAS), Directional Autopilot SerRvos(DAS), and Lateral Central Control Actuators (LCCA)) Inoperative
      (4) DDG 22-10-1 Restoration - Autopilot Electrohydraulic Servos (Elevator Autopilot Servos (EAS), Directional Autopilot Sevos(DAS), and Lateral Central Control Actuators (LCCA)) Inoperative
      (5) DDG 22-21-1 Preparation - Yaw Damper System Inoperative
      (6) DDG 22-21-1 Restoration - Yaw Damper System Inoperative
      (7) DDG 22-21-1 Preparation - Yaw Damper Servos Inoperative
      (8) DDG 22-21-1 Restoration - Yaw Damper Servos Inoperative
      (9) DDG 22-30-1 Preparation - Autothrottle Disconnect Switches Inoperative
      (10) DDG 22-30-1 Restoration - Autothrottle Disconnect Switches Inoperative

TASK 22-00-00-049-001

2. DDG 22-10-1 Preparation - Autopilot Flight Control Computers Inoperative
   A. General
      (1) This task contains the maintenance steps to prepare the airplane for flight with one Autopilot Flight Control Computer Inoperative.
      (2) One channel of the three channel autopilot system can be inoperative if these conditions hold:
         (a) All of the FCC power circuit breakers are closed.
         (b) If the FCC Single Source Option is not installed, the inoperative flight control computer (FCC) must send sensor data to the other two FCC's.
         (c) The FCC SERVO circuit breaker of the inoperative channel is open and a collar is attached.
         (d) The approach minimums do not require three operational FCC's.

B. References
   (1) AMM 22-00-02/201, Autoflight BITE
   (2) AMM 22-11-01/401, Flight Control Computer
C. Standard Tools and Equipment
   (1) Collar - Circuit Breaker Locking, Paco Plastics P/N S-4933959, Paco Plastics, Santa Fe Springs, CA.
D. Access
   (1) Location Zones
       119/120 Main Equipment Center

   (2) Access Panel
       119AL Main Equipment Center

E. Deactivation Procedure

S 869-002
(1) Make sure that power is applied to the left, center, and right FCC's.

NOTE: The Inoperative FCC must remain on during flight.

(a) Close these FCC Circuit Breakers on the P11 Overhead Circuit Breaker Panel:
   1) 11E17, FLT CONT CMPTR PWR L
   2) 11E20, FLT CONT CMPTR PWR C
   3) 11E35, FLT CONT CMPTR PWR R

S 869-061
(2) Remove power from the inoperative FCC SERVO.

NOTE: The inoperative FCC SERVO circuit breaker must remain open during flight.

(a) Open and collar the applicable FCC SERVO circuit breaker, on the P11 Overhead Circuit Breaker Panel, for the inoperative Autopilot channel.
   1) 11E18, FLT CONT CMPTR SERVO L (left FCC Servo)
   2) 11E21, FLT CONT CMPTR SERVO C (Center FCC Servo)
   3) 11E36, FLT CONT CMPTR SERVO R (Right FCC Servo)

S 749-023
(3) Autopilot FCC Sensor Data Interface Test

NOTE: This test uses MCDP Test 01-FCC to check sensor data transmission between the left, right, and center FCC's.

(a) Do MCDP "Ground Test 01 - FCC", AMM 22-00-02/201.

1) Make sure that this MCDP fault message is not displayed:

NO INFC FCC L(C,R)
X- CH BUS L(C,R)

NOTE: The above fault code is a single message displayed on two lines.

2) Ignore all fault codes caused by the inoperative FCC.

(b) Do this test for the left, center and right FCC's.

S 869-004
(4) Open the applicable FCC SERVO circuit breaker for the inoperative autopilot channel and attach a collar.

TASK 22-00-00-449-005
3. DDG 22-10-1 Restoration - Autopilot Flight Control Computers Inoperative
A. General
(1) This task contains steps to put the airplane back to its usual condition after operation with the Autopilot Flight Control Computer Inoperative.
(2) To put the airplane back to its usual condition, you must do the following:
(a) Replace the inoperative FCC.
(b) Close the FCC SERVO circuit breaker of the inoperative channel.
(c) Make sure the all three FCC's operate correctly by doing the MCDP Ground Test 01 FCC.
B. References
(1) AMM 22-00-02/201, Autopilot Bite
(2) AMM 22-11-01/401, Flight Control Computer
C. Access
(1) Location Zones
119/120 Main Equipment Center

(2) Access Panel
119AL Main Equipment Center

D. FCC Reactivation Procedure

S 029-062

(1) Remove the inoperative FCC.
   (a) Do the "Remove the Flight Control Computer" procedure, AMM 22-11-01/401 (Flight Control Computer - Removal/Installation).

S 429-063

(2) Install the replacement FCC.
   (a) Remove the collar and close the inoperative FCC SERVO circuit breaker.
   (b) Do the "Install the Flight Control Computer" procedure, AMM 22-11-01/401 (Flight Control Computer - Removal/Installation).

S 719-024

(3) Make sure that the AFDS system is operational.
   (a) Do the "AFDS Operational Test", AMM 22-10-00/501 (Autopilot (Flight Control) - Adjustment/Test).

TASK 22-00-00-049-006

4. DDG 22-10-1 Preparation - Autopilot Servos Inoperative

A. General
   (1) This task contains the maintenance steps to prepare the airplane for flight with one of these Autopilot Servos inoperative. Either the Elevator Autopilot Servos (EAS), the Directional Autopilot Servos (DAS), or the Lateral Central Control Actuators (LCCA). Each Autopilot System includes a left, right, and center servo.

B. Equipment
   (1) EAS DEACTIVATION;
       Service Platform, Control Bay Access Door - A51001-19
   (2) DAS DEACTIVATION;
       Nose Gear Steering Valve Lockpin - A09003-1

C. References
   (1) AMM 06-42-00/201, Empennage (Major Zones 300) Access Doors and Panels
   (2) AMM 06-45-00/201, Landing Gear and Gear Doors (Major Zones 700) Access Doors and Panels
   (3) AMM 09-11-00/201, Towing - Maintenance Practices
   (4) AMM 22-12-01/401, Elevator Autopilot Servo - Removal/Installation
   (5) AMM 22-13-01/401, Directional Autopilot Servo - Removal/Installation
   (6) AMM 22-13-03/401, LCCA - Removal/Installation
   (7) AMM 24-22-00/201, Electrical Power - Control

(8) AMM 29-11-00/201, Main (Left, Center, and Right) Hydraulic Systems

D. Access

(1) Location Zone

(a) EAS;
313 Stabilizer Torsion Box Compartment (Left)

(b) DAS;
324 Vertical Stabilizer - Rear Spar to Trailing Edge

(c) LCCA;
732/742 Left/Right Main Landing Gear Body Doors

(2) Access Door or Panel

(a) EAS;
313AL Elevator Controls Bay Access

(b) DAS;
324BL Rudder Controls Bay Access

(c) LCCA;
732/742 Left and Right Main Landing Gear Doors

E. Prepare for Deactivation

S 869-021

(1) Supply electrical power (AMM 24-22-00/201).

S 869-058

(2) DAS SERVO DEACTIVATION;
Do these steps:

<u>WARNING</u>: LOCK THE NOSE GEAR STEERING. THE NOSE GEAR WILL MOVE WITH RUDDER MOVEMENT. THIS CAN CAUSE INJURIES TO PERSONS AND DAMAGE TO EQUIPMENT.

(a) Move the towing lever, on the nose gear metering valve module, to the towing position.

(b) Install the nose gear steering valve lockpin (AMM 09-11-00/201)

S 869-059

(3) LCCA DEACTIVATION;
Do this step:

<u>WARNING</u>: MAKE SURE THAT THE DOWNLOCKS ARE INSTALLED IN ALL OF THE LANDING GEAR. WITHOUT THE DOWNLOCKS, THE LANDING GEAR CAN RETRACT AND CAUSE INJURIES TO PERSONS AND DAMAGE TO EQUIPMENT.

(a) Make sure that the downlocks are installed in the nose and main landing gear (AMM 32-00-20/201).

S 869-056
(4) Remove the hydraulic pressure, on the Hydraulics System Control Panel (P5 - Overhead Panel), from the inoperative Autopilot Servo (AMM 29-11-00/201).
  (a) Left EAS/DAS/LCCA - Set the "L ENG" Hydraulic pump switch to OFF.
  (b) Center EAS/DAS/LCCA - Set the "C 1-ELEC-2" Hydraulic pump switches to OFF.
  (c) Right EAS/DAS/LCCA - Set the "R ENG" Hydraulic pump switch to OFF.

S 869-023
(5) Set the FLT CONTROL HYDRAULIC SHUTOFF switches, on the Miscellaneous Systems Control Panel (P61 Right Side Panel), to OFF for the inoperative SERVO and attach DO-NOT-OPERATE tags.
  (a) EAS/DAS - L, C, and R TAIL FLT CONTROL SHUTOFF switches OFF.
  (b) LCCA - L, C, and R WING FLT CONTROL SHUTOFF switches OFF.

S 759-055
(6) Make sure that the applicable HYDRAULIC FLIGHT CONTROL SHUTOFF switch indicator lights (left, center, and right) go Amber.

S 869-024
(7) Put the L AND C STAB TRIM switches, on the P10 - Control Stand, in the CUTOUT position and attach DO-NOT-OPERATE tags.

S 869-027
(8) Attach DO-NOT-OPERATE tags on the CAPT's and F/O's control columns.

S 869-108
(9) Set the CAPT's and F/O's F/D switches, on the MCP, to the OFF position.

S 869-110
(10) Open these circuit breakers, on the P11 - Overhead Circuit Breaker Panel, and attach DO-NOT-CLOSE tags:
  (a) 11C12, STAB TRIM SHUTOFF L
  (b) 11C13, STAB TRIM SHUTOFF CENTER
  (c) 11H15, FLT CONT SHUTOFF WING L
  (d) 11H16, FLT CONT SHUTOFF WING CENTER
  (e) 11H17, FLT CONT SHUTOFF TAIL L
  (f) 11H18, FLT CONT SHUTOFF TAIL CENTER
  (g) 11H26, FLT CONT SHUTOFF WING R
  (h) 11H27, FLT CONT SHUTOFF TAIL R
  (i) 11E16, MODE CONT PNL L
  (j) 11E34, MODE CONT PNL R
  (k) 11E17, FLT CONT CMPTR PWR L

(l) 11E18, FLT CONT CMPTR SERVO L (Left FCC Servo)
(m) 11E20, FLT CONT CMPTR PWR C
(n) 11E21, FLT CONT CMPTR SERVO C (Center FCC Servo)
(o) 11E35, FLT CONT CMPTR PWR R
(p) 11E36, FLT CONT CMPTR SERVO R (Right FCC Servo)

S 869-049

(11) Open and collar the inoperative FCC SERVO circuit breaker, on the P11 Overhead Circuit Breaker panel .

NOTE: The inoperative autopilot channel FCC SERVO circuit breaker must remain open during flight.

(a) 11E18, FLT CONT CMPTR SERVO L (left FCC Servo)
(b) 11E21, FLT CONT CMPTR SERVO C (Center FCC Servo)
(c) 11E36, FLT CONT CMPTR SERVO R (Right FCC Servo)

F. Autopilot Servo Deactivation Procedure

S 019-028

(1) Gain access to the inoperative EAS, DAS, OR LCCA (AMM 06-42-00/201).
(a) EAS SERVOS;
Do these steps:
1) Open elevator controls bay access door, 313AL (AMM 06-42-00/201).

WARNING: STAY OFF THE SERVICE ACCESS DOOR, 312AR, AND THE ACCESS DOOR, 313AL, TO THE CONTROLS BAY. YOUR WEIGHT CAN RELEASE THE SPRING-LOADED LATCHES ON THE DOOR. IF YOU FALL THROUGH THE DOOR, INJURIES CAN OCCUR.

2) Install the service platform over the controls bay access door, 313AL.
(b) DAS SERVOS;
Remove the Rear spar to trialing edge access panel, 324Bl, located at the bottom of the vertical fin (left side) (AMM 06-42-00/201).

(c) LCCA SERVOS;
Do these steps:

WARNING: OBEY THE INSTALLATION PROCEDURE FOR THE DOOR LOCKS. THE DOORS OPEN AND CLOSE QUICKLY. THE MOVEMENT OF THE DOORS CAN CAUSE INJURIES TO PERSONS AND DAMAGE TO EQUIPMENT.

1) Open the main landing gear doors and install the door locks (AMM 32-00-15/201).
   a) Left and Right Autopilot LCCA - Access the right main landing gear body access door, 742 (AMM 06-42-00/201).
   b) Center Autopilot LCCA - Access the Left main landing gear body access door, 732 (AMM 06-42-00/201).

S 119-052

CAUTION: DO NOT LET ANY CONTAMINATION GET INTO THE ELEVATOR AUTOPILOT SERVO AND VALVES. CONTAMINATION CAN CAUSE EQUIPMENT DAMAGE AND FAILURE.

(2) Fully clean the external areas of the servo and hydraulic line connections.

S 369-087

(3) Disconnect and cap the pressure and return hydraulic lines.

S 369-086

(4) Plug the pressure and return servo hydraulic ports.

S 119-051

CAUTION: QUICKLY CLEAN THE AREA OF ALL HYDRAULIC FLUID. HYDRAULIC FLUID CAN CAUSE EQUIPMENT DAMAGE.

(5) Clean all hydraulic fluid from the area of the servo (AMM 12-25-01/301).

G. Prepare Autopilot Servo Operational Test

S 869-062

(1) To complete this test, these systems or components must be servicable:
   (a) Engine Indicating and Crew Alerting System (EICAS) (AMM 31-41-00).
   (b) Warning System (AMM 31-51-00).
   (c) Master Dim and Test (AMM 33-16-00).
   (d) Air Data System (AMM 34-12-00).
   (e) Inertial Reference System (AMM 34-21-00).
   (f) Electronic Flight Instrument System (EFIS) (AMM 34-22-00).
   (g) Instrument Landing System (AMM 34-31-00).
   (h) Radio Altimeter (AMM 34-33-00).
   (i) Flight Management System (AMM 34-61-00).

S 869-050

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers, on the P11 Overhead Circuit Breaker panel:
   (a) 11C12, STAB TRIM SHUTOFF L
   (b) 11C13, STAB TRIM SHUTOFF CENTER
   (c) 11H15, FLT CONT SHUTOFF WING L
   (d) 11H16, FLT CONT SHUTOFF WING CENTER
   (e) 11H26, FLT CONT SHUTOFF WING R
   (f) 11H17, FLT CONT SHUTOFF TAIL L
   (g) 11H18, FLT CONT SHUTOFF TAIL CENTER
   (h) 11H27, FLT CONT SHUTOFF TAIL R
   (i) 11E16, MODE CONT PNL L
   (j) 11E34, MODE CONT PNL R
   (k) 11E17, FLT CONT CMPTR PWR L
   (l) 11E20, FLT CONT CMPTR PWR C
   (m) 11E36, FLT CONT CMPTR SERVO R

S 869-037

(3) Remove the DO-NOT-CLOSE tags and close the two operative Autopilot FCC SERVO circuit breakers, on the P11 panel:
   (a) 11E18, FLT CONT CMPTR SERVO L

(b) 11E21, FLT CONT CMPTR SERVO C
(c) 11E35, FLT CONT CMPTR PWR R

S 869-038
(4) Remove the DO-NOT OPERATE tags and put the L and C STAB TRIM switches, on the P10 - Control Stand, in the NORM position.

S 869-117
(5) Remove the DO-NOT-OPERATE tags on the two control columns.

S 869-112
(6) Remove the DO-NOT-OPERATE tags and put the L, C, and R TAIL and WING FLT CONTROL SHUTOFF switches, on the P61 - Right Side Panel, in the ON position.

S 869-118

WARNING: KEEP PERSONS AND EQUIPMENT CLEAR OF THE FLIGHT CONTROL SURFACES, THE THRUST REVERSERS, AND THE LANDING GEAR. THESE COMPONENTS CAN MOVE SUDDENLY WHEN YOU SUPPLY HYDRAULIC POWER. THIS CAN CAUSE INJURIES TO PERSONS AND DAMAGE TO EQUIPMENT.

(7) Supply hydraulic power to the L, R, and C hydraulic systems (AMM 29-11-00/201).

S 989-041
(8) EAS SERVO DEACTIVATION;
Manually operate the elevator system three or more times while checking for usual operation.

S 989-055
(9) DAS SERVO DEACTIVATION;
Manually operate the rudder system three or more times while checking for usual operation.

S 989-056
(10) LCCA SERVO DEACTIVATION;
Manually operate the aileron system three or more times while checking for usual operation.

S 869-026
(11) Align the Left, Center, and Right IRU's, AMM 34-21-00/201.

S 869-114
(12) Set the MCDP to OFF.

S 869-030
(13) Set the A/T switch, on the MCP, to OFF.

S 869-029
(14) Set these CAPT's (P1-1 pnl) and F/O's (P3-3 pnl) Instrument Source Select switches to their usual positions:
(a) FLT DIR switches
(b) NAV or FMC select switches as installed
(c) EFI switches
(d) IRS switches
(e) AIR DATA select switches as installed

H. Autopilot Operational Test

S 869-047
(1) Cycle the CAPT's and F/O's F/D switches, on the MCP, from OFF to ON.

S 869-028
(2) Engage one of the two operative Autopilot channels in CMD.

S 759-031
(3) Make sure that HDG HOLD and V/S messages show on the CAPT's and F/O'S EADI's.

S 869-032
(4) Push the HDG SEL control knob on the MCP.

S 759-033
(5) Make sure the HDG HOLD message changed to HDG SEL on the CAPT's and F/O's EADI's.

S 979-034
(6) Record the heading that shows in the HDG display window of the MCP. This is the initial heading.

S 869-035
(7) Turn the HDG SEL control knob to set a heading 100 degrees less than the initial heading (simulated roll left).

S 759-037
(8) Make sure the control wheel moves CounterClockwise (CCW) and the ailerons move in a left-wing-down direction.

S 869-036
(9) Set a heading 100 degrees plus the initial heading (simulated roll right).

S 759-038
(10) Make sure the control wheel moves clockwise (CW) and the ailerons move in a right-wing-down direction.

S 869-063
(11) Set HDG display to the initial heading.

S 869-064
(12) Turn the Vertical Speed Select Knob to set a vertical speed of -1000 FPM in the VERT SPD display window of the MCP.

S 759-041
(13) Make sure the control columns move forward and the elevators move in a nose down (trailing edge down) direction in less than 15 seconds.

S 869-042
(14) Set the vertical speed to +1000 FPM in the VERT SPD display.

S 759-043
(15) Make sure the control columns move aft and the elevators move in a nose-up (trailing edge up) direction in less than 15 seconds.

S 719-044
(16) Push the CAPT's or F/O's control wheel A/P Disengage switch to disengage the autopilot.

S 719-045
(17) Push the CAPT's or F/O's control wheel A/P Disengage switch again to cancel the disengage warning.

S 719-048
(18) Do the Autopilot Operational Test again for the remaining operative autopilot.

I. Prepare Airplane for Flight

S 869-066
(1) DAS DEACTIVATIONS;
Do these steps:

WARNING: STAY AWAY FROM THE NOSE GEAR WHEELS WHEN YOU REMOVE THE LOCKPIN. WHEELS CAN MOVE TO THE CENTER POSITION QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(a) Do these steps to remove the nose gear steering valve lockpin if it is not necessary:
   1) Make sure the nose gear wheels are in the center position.
   2) Remove the nose gear steering valve lockpin (AMM 09-11-01/201).

S 869-065
(2) Remove pressure from the left, center, and right hydraulic systems.

S 219-043
(3) Examine the hydraulic lines and servo ports for leakage and repair as necessary.

S 849-071
(4) Install or close all access panels and doors.

S 089-047
(5) Remove all test and support equipment.

TASK 22-00-00-449-007
5. DDG 22-10-1 Restoration - Autopilot Electrohydraulic Servos Inoperative
A. General
   (1) This task contains the maintenance steps to put the airplane back to its usual condition after operation with the with one of these Autopilot Electrohydraulic Servos (EHVS) inoperative. Either the Elevator Autopilot Servos (EAS), the Directional Autopilot Servos (DAS), or the Lateral Central Control Actuators (LCCA). Each EHVS includes a left, right, and center servo.

B. References
   (1) AMM 22-00-02/201, Autoflight BITE - Maintenance Practices
   (2) AMM 22-12-01/401, Elevator Autopilot Servo - Removal/Installation
   (3) AMM 22-13-01/401, Directional Autopilot Servo - Removal/Installation
   (4) AMM 22-13-03/401, LCCA - Removal/Installation
C. Access
   (1) Location Zone
      (a) EAS;
         313 Stabilizer Torsion Box Compartment (Left)

(b) DAS;
324 Vertical Stabilizer - Rear Spar to Trailing Edge
(c) LCCA;
732/742 Left/Right Main Landing Gear Body Doors
(2) Access Door or Panel
(a) EAS;
313AL Elevator Controls Bay Access
(b) DAS;
324BL Rudder Controls Bay access
(c) LCCA;
732/742 Lateral Controls access

D. Procedure

S 919-070
(1) Replace the inoperative EHSV.
(a) EAS SERVO'S;
Do the Elevator Autopilot Servo - Removal/Installation Task, AMM 22-12-01/401.
(b) DAS SERVO'S;
Do the Directional Autopilot Servo - Removal/Installation Task, AMM 22-13-01/401.
(c) LCCA SERVO'S;
Do the Electrohydraulic Servovalve and Solenoid Valves - Removal/Installation Task, AMM 22-13-02/401.

S 869-116
(2) Put the airplane back to its usual condition.

TASK 22-00-00-049-010
6. <u>DDG 22-21-1 Preparation - Yaw Damper System Inoperative</u>
A. General
(1) This task contains maintenance instructions to prepare the airplane for flight with one Yaw Damper System Inoperative.
(2) One of the yaw damper channels can be inoperative if these conditions hold:
(a) The remaining yaw damper is serviceable.

<u>NOTE</u>: AIRPLANES WITHOUT SB 767-22-7 OR PRODUCTION EQUIVALENT INSTALLED;
See AFM limitations.

B. References
(1) AMM 22-21-00/501, Yaw Damper System
(2) AMM 22-21-04/401, Yaw Damper Module

C. Access
(1) Location Zones
119/120 Main Equipment Center

D. Deactivation Procedure

S 749-011
(1) Do the "Operational Test" for the remaining (Operational) Yaw Damper Module, AMM Task 22-21-00/501 Yaw Damper System.
(a) Make sure that the remaining Yaw Damper Module successfuly completes this test.

NOTE: Ignore all Faults caused by the inoperative Yaw Damper Module.

TASK 22-00-00-449-012
7. DDG 22-21-1 Restoration - Yaw Damper System Inoperative
A. General
(1) This task contains steps to put the airplane back in its usual condition after operation with the Yaw Damper System Inoperative.
(2) To put the airplane back in its usual condition, you must do as follows:
(a) Replace the inoperative Yaw Damper Module (YDM).
1) Do the YDM Removal/Installation tasks.
(b) Check the Yaw Damper system for correct operation.
B. References
(1) AMM 22-21-04/401, Yaw Damper Module
(2) AMM 22-21-00/501, Yaw Damper System
C. Access
(1) Location Zones
119/120 Main Equipment Center

(2) Access Panel
119AL Main Equipment Center Access Door

D. Reactivation Procedure

S 449-013
(1) Replace the Inoperative Yaw Damper Module (AMM 22-21-04/401).

S 719-102

(2) Do the "Operational Test", AMM 22-21-00/501 Yaw Damper System - Adjustment/Test, for the remaining (operative) Yaw Damper Module.

(a) Make sure that the remaining (operative) Yaw Damper Module successfully completes this test.

NOTE: Ignore faults caused by the inoperative yaw damper module.

S 869-067

(3) Put the airplane back to its usual condition.

TASK 22-00-00-049-008

8. DDG 22-21-1 Preparation - Yaw Damper Servo Inoperative

A. General

(1) This task contains maintenance instructions to prepare the airplane for flight with one yaw damper servo Inoperative due to a hydraulic leak.

B. References

(1) AMM 06-42-00/201, Vertical Stabilizer and Rudder Access Doors and Panels
(2) AMM 24-22-00/201, Electrical Power - Control
(3) AMM 27-61-00/201, Spoiler/Speedbrake Control System
(4) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System

C. Access

(1) Location Zone
320 Vertical Stabilizer and Rudder

(2) Access Panel
324EL Vertical Stabilizer, Rear Spar to Trailing Edge

D. Prepare for Deactivation

S 049-101

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 869-074
(2) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
(a) 11A18, YAW DAMPER L
(b) 11C6, FLT CONT ELEC 1L AC
(c) 11C7, FLT CONT ELEC 1L DC
(d) 11C8, FLT CONT ELEC 2L AC
(e) 11C9, FLT CONT ELEC 2L DC
(f) 11F34, YAW DAMPER R
(g) 11G17, FLT CONT ELEC 1R AC
(h) 11G18, FLT CONT ELEC 1R DC
(i) 11G26, FLT CONT ELEC 2R AC
(j) 11G27, FLT CONT ELEC 2R DC

S 869-075
(3) Supply electrical power (AMM 24-22-00/201).

S 869-076
(4) For the right yaw damper system servo (left side of airplane), remove the pressure in the left hydraulic system (AMM 29-11-00/201).

S 869-077
(5) For the left yaw damper system servo (right side of airplane), remove the pressure in the center hydraulic system (AMM 29-11-00/201).

S 869-078
(6) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches on the right side panel P61 in the OFF position.

S 869-079
(7) Put the L and C STAB TRIM switches on the control stand panel, P10, in the CUT OUT position.

S 869-080
(8) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11H17, FLT CONT SHUTOFF TAIL L
(d) 11H18, FLT CONT SHUTOFF TAIL CENTER

S 869-081

WARNING: MAKE SURE PERSONNEL STAY A MINIMUM OF 6 FEET AWAY FROM THE VERTICAL STABILIZER WHEN THE HF SYSTEM TRANSMITS. RF ENERGY FROM THE HF COMMUNICATION ANTENNA CAN CAUSE INJURIES TO PERSONNEL.

(9) Make sure the HF system does not transmit.

E. Deactivation Procedure

S 019-082

(1) Open the service access panel 324EL (AMM 6-42-00/201) and access the inoperative yaw damper servo.

S 119-090

CAUTION: DO NOT LET ANY CONTAMINATION GET INTO THE ELEVATOR AUTOPILOT SERVO AND VALVES. CONTAMINATION CAN CAUSE EQUIPMENT DAMAGE AND FAILURE.

(2) Fully clean the external areas of the yaw damper servo and hydraulic line connections.

S 369-084

(3) Disconnect and cap the pressure and return hydraulic lines from the yaw damper servo.

S 369-085

(4) Plug the hydraulic ports on the yaw damper servo.

S 119-089

CAUTION: QUICKLY CLEAN THE AREA OF ALL HYDRAULIC FLUID. HYDRAULIC FLUID CAN CAUSE EQUIPMENT DAMAGE.

(5) Clean all hydraulic fluid from the area of the servo (AMM 12-25-01/301).

F. Yaw Damper Servo Test

S 719-103

(1) Do the "Operational Test", AMM 22-21-00/501 Yaw Damper System - Adjustment/Test, for the remaining (operative) Yaw Damper Module.
 (a) Make sure that the remaining (operative) Yaw Damper Module successfully completes this test.

 NOTE: Ignore faults caused by the inoperative yaw damper module.

G. Check for hydraulic leakage.

S 869-104

(1) Put the L, C and R TAIL FLT CONTROL SHUTOFF switches in the OFF position.

S 869-105

(2) Put the L and C STAB TRIM switches in the CUT OUT position.

S 869-100

(3) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
 (a) 11C12, STAB TRIM SHUTOFF L
 (b) 11C13, STAB TRIM SHUTOFF CENTER
 (c) 11H17, FLT CONT SHUTOFF TAIL L
 (d) 11H17, FLT CONT SHUTOFF TAIL L
 (e) 11H27, FLT CONT SHUTOFF TAIL R

S 869-099

WARNING: MAKE SURE PERSONNEL STAY A MINIMUM OF 6 FEET AWAY FROM THE VERTICAL STABILIZER WHEN THE HF SYSTEM TRANSMITS. RF ENERGY FROM THE HF COMMUNICATION ANTENNA CAN CAUSE INJURIES TO PERSONNEL.

(4) Make sure the HF system does not transmit.

S 219-088

(5) Make an inspection of the servos and hydraulic line connections for hydraulic fluid leakage and correct as necessary.

H. Put the Airplane Back to its Usual Condition

S 419-093

(1) Close service access panel 324EL (AMM 06-42-00/201).

S 869-094
(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11H17, FLT CONT SHUTOFF TAIL L
(d) 11H18, FLT CONT SHUTOFF TAIL CENTER
(e) 11H27, FLT CONT SHUTOFF TAIL R

S 869-098
(3) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches in the ON position.

S 869-095
(4) Put the L and C STAB TRIM switches in the NORM position.

S 869-096
(5) Remove the power from the left and center hydraulic systems (AMM 29-11-00/201).

S 869-097
(6) Remove electrical power if it is not necessary (AMM 24-22-00/201).

TASK 22-00-00-049-009
9. <u>DDG 22-30-1 Preparation - Autothrottle Disconnect Switches Inoperative</u>
A. General
(1) This task contains the maintenance steps to prepare the airplane for flight with the Autothrottle Disconnect Switches Inoperative.
B. Access
(1) Location Zones
211/212 Flight Compartment

C. Deactivation Procedure

S 869-010
(1) Open these Autothrottle System circuit breakers, on the P11 Overhead Circuit Breaker Panel, and attach a collar.
(a) 11F14, TMC AC
(b) 11F15, TMC DC
(c) 11F16, TMC SERVO

S 869-011
(2) Set the A/T ARM switch, on the AFCS Mode Control panel (P55 PNL), to OFF and attach a DO-NOT-OPERATE tag.

TASK 22-00-00-449-119

10. DDG 22-30-1 Restoration - Autothrottle Disconnect Switches Inoperative

A. General
   (1) This task puts the airplane back to its usual condition after operation with the Autothrottle Disconnect Switches inoperative.

B. References
   (1) AMM 22-32-02/401, Disengage Switches
   (2) AMM 22-00-02/201, Autoflight BITE

C. Access
   (1) Location Zones
       211/212 Flight Compartment

D. Reactivation Procedure

   S 869-017
   (1) Remove the DO-NOT-OPERATE tag on the A/T ARM switch (on the P55 - AFCS Mode Control panel).

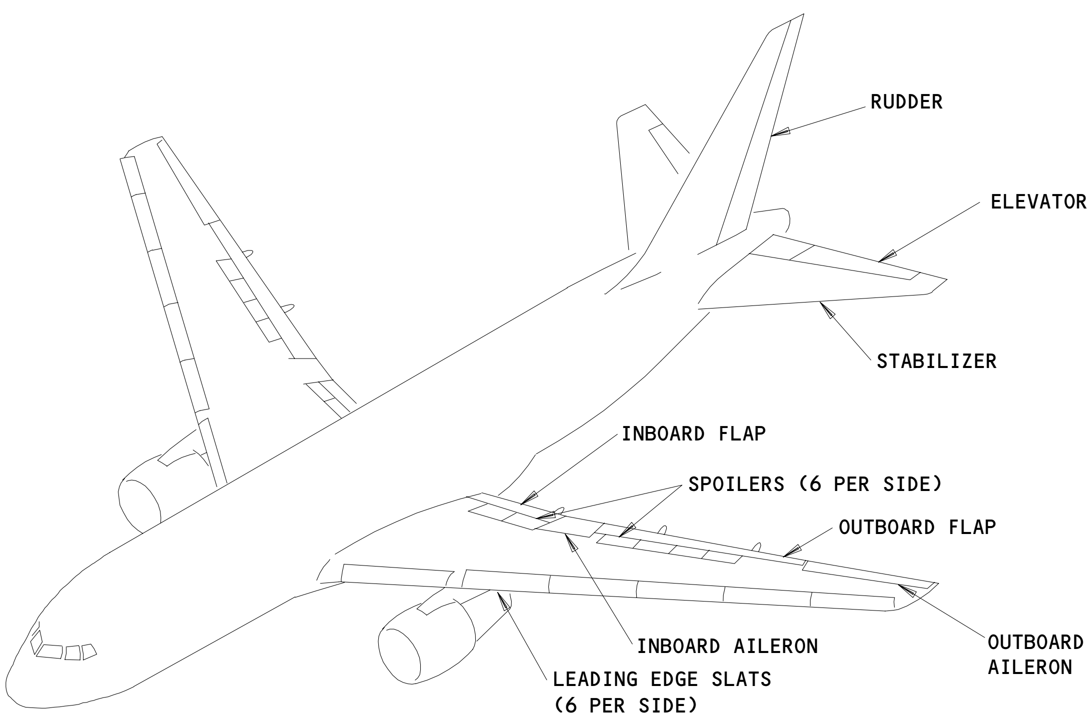
   S 869-015
   (2) Remove these Autothrottle System circuit breaker collars on the P11 Overhead Circuit Breaker Panel.
       (a) 11F14, TMC AC
       (b) 11F15, TMC DC
       (c) 11F16, TMC SERVO

   S 449-018
   (3) Replace the Inoperative Autothrottle Disconnect Switches, AMM 22-32-02/401 Disengage Switches - Removal/Installation).

   S 869-019
   (4) Put the airplane back to its usual condition.

HYDRAULIC AND FLIGHT CONTROLS (AUTOFLIGHT) - DESCRIPTION AND OPERATION

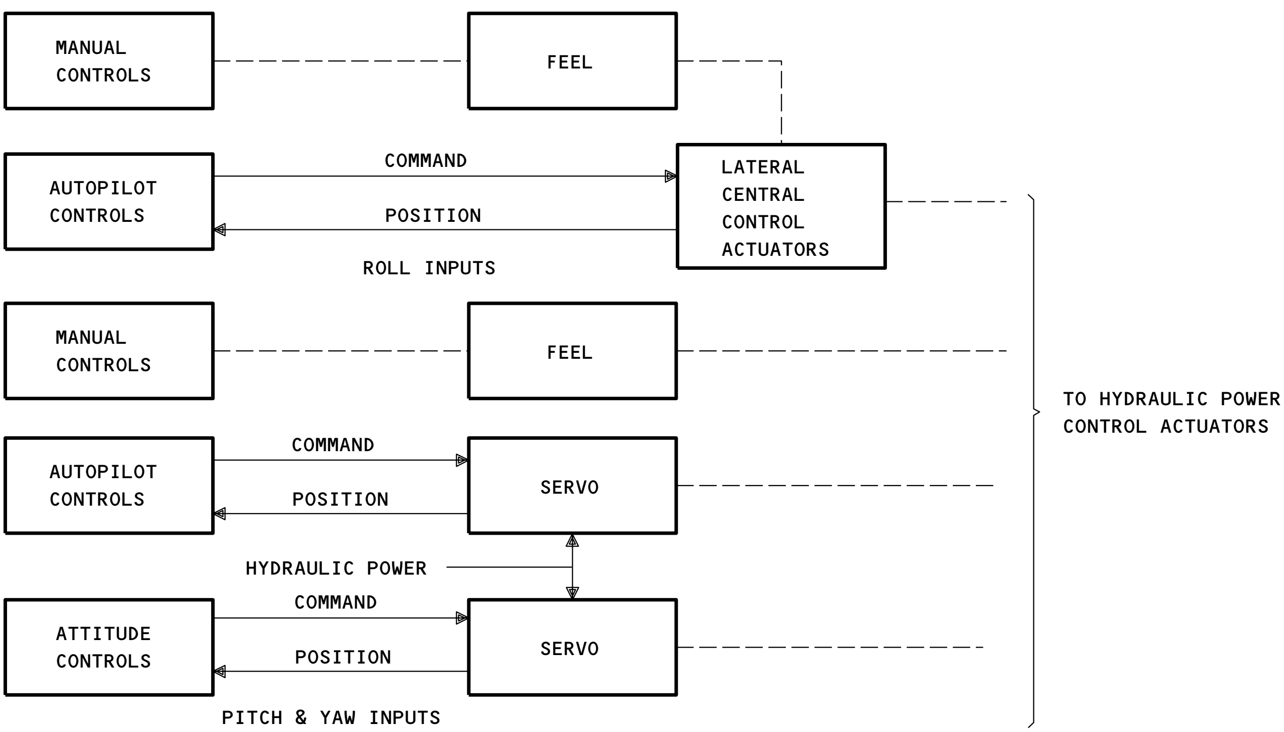
1. General (Fig. 1)
   A. Autoflight systems drive hydraulic powered servo actuators which drive hydraulic Power Control Actuators (PCAs) connected to the ailerons, elevators, and rudder. The PCAs are controlled either manually or from autoflight system servos. Secondary flight controls such as spoilers, flaps, slats, and stabilizer are also discussed. The following pages provide a general discussion of airplane Hydraulic Power (Chapter 29) and Flight Controls (Chapter 27).



Flight Controls - Introduction
Figure 1

B. Flight Controls - Introduction
   (1) The primary flight controls consist of seven movable surfaces for the three control axis:
      (a) Roll control: Inboard and Outboard Ailerons, two surfaces each wing (4)
      (b) Pitch control: Elevators, one surface each side (2)
      (c) Directional control (yaw): Rudder, one surface
   (2) Secondary flight controls consist of thirty movable surfaces for four systems.
      (a) Spoilers: provide lift and drag, augment roll control, six surfaces each wing (12)
      (b) Leading edge slats: provide high lift, six surfaces each wing (12)
      (c) Trailing edge flaps: provide high lift, two surfaces each wing (4)
      (d) Stabilizer: augments pitch control, one surface each side (2)
   (3) Actuation of all control surfaces is by hydraulic servos or actuators. Primary flight control actuators are controlled mechanically by both the pilots and the electro-hydraulic autoflight servos. Spoilers are controlled only with electro-hydraulic actuators. Flaps and slats are driven with hydraulic motors; they also are driven by a backup system using electric motors.

C. Flight Control Inputs (Fig. 2)



Flight Control Inputs
Figure 2

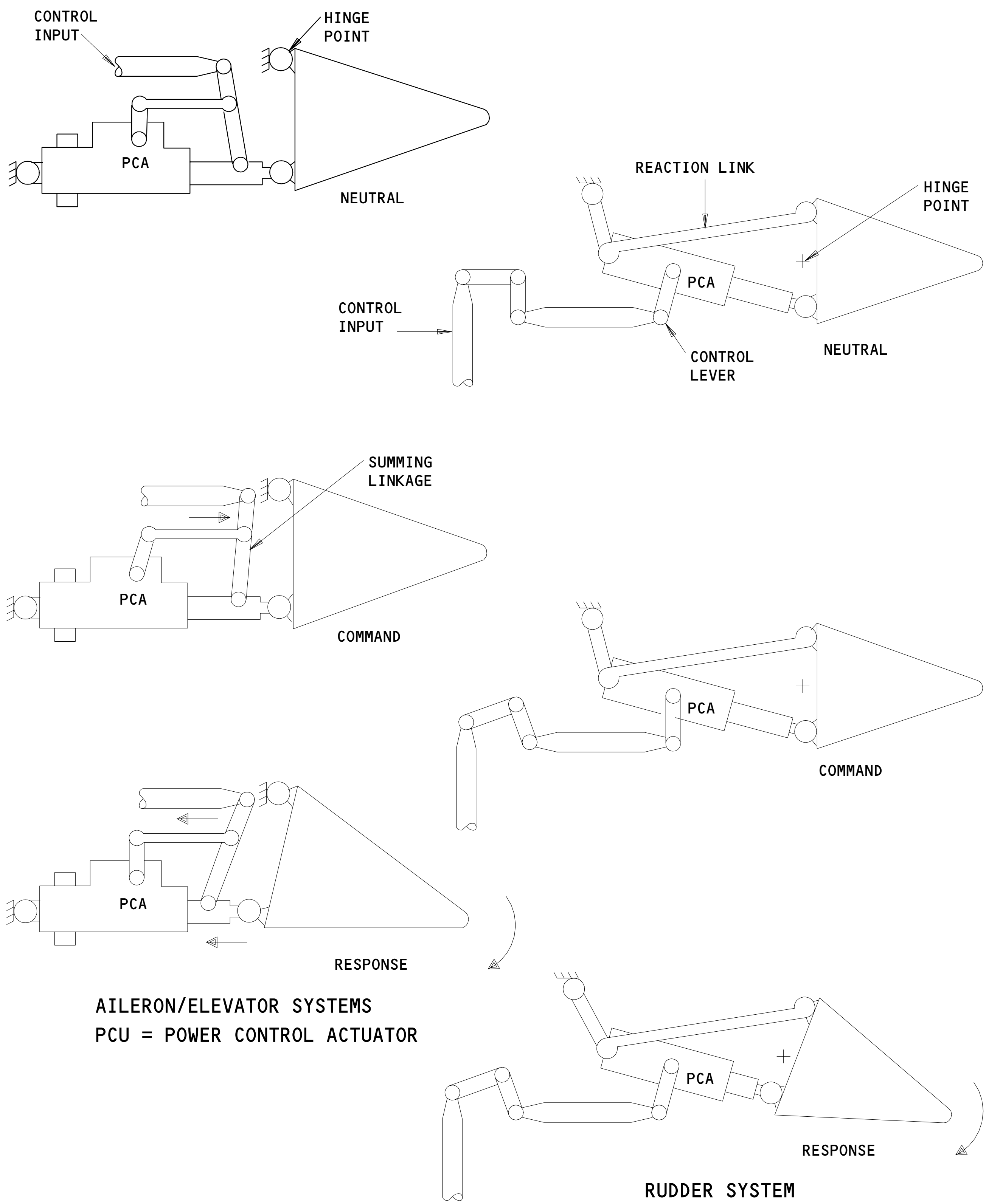
(1) All primary flight control surfaces are controlled by hydraulic power control actuators (PCAs). Mechanical and hydraulic devices are used to provide normal control system feel. The PCAs are controlled by movement of mechanical linkage. The mechanical linkage is moved as a result of:
   (a) Conventional manual flight controls consisting of control columns, cables and quadrants, etc.
   (b) Actuators responding to either manual or autopilot input signals.
   (c) Servos responding to autopilot command signals.

(2) Two or more actuators drive each primary flight control surface. Each PCA is powered by only one hydraulic system. This provides dual or triple path control for each primary control surface ensuring control of the airplane in the event of a hydraulic system or actuator failure. The servos and actuators of controlling systems also use hydraulic power.

(3) Operation of the autopilot attitude actuators and servos is similar except that when electrically engaged the input command is electrical rather than mechanical. The actuators and servos electrical feedback (position) neutralizes the control input (command) from the system computer when the new position is reached.

D. Power Control Actuator Reaction (Fig. 3)

(1) The mechanical inputs to the PCAs provide the commands for movement of the control surfaces. The inputs are neutralized by the reaction of the PCA. For the aileron and elevator systems, the action of the summing linkage maintains the PCA control lever in the neutral position. The linkage sums the motion of the control input and the output piston, which move in opposite directions. The resultant establishes the position of the control lever. For the rudder system, the PCA is basically floating except it is referenced to structure to prevent lateral motion. Longitudinal motion occurs when the rudder is moved in response to a command. That motion, caused by movement of the reaction link, neutralizes the position of the control lever.



Power Control Actuator Reaction
Figure 3

(2) The PCAs continue to move in the commanded direction as long as the control lever is displaced from neutral. When more than one PCA is used on a single surface, movement is synchronized by adjustment of the control linkage. All control surfaces are moved by hydraulic power input to the PCAs. There is no reversion to manual movement.

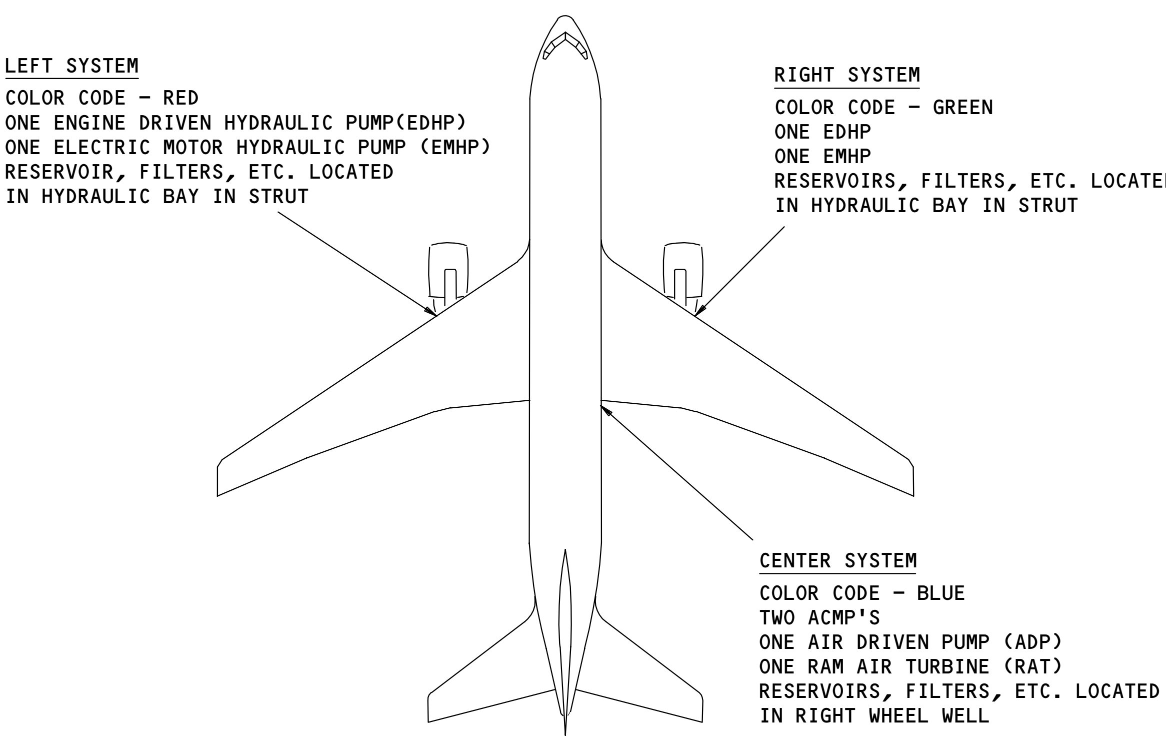
E. Hydraulic Power Systems - Introduction (Fig. 4)

(1) Three independent hydraulic systems are installed. System tubing is color coded for easy identification. The left is red, the center is blue, and the right is green. Each hydraulic system provides power for all essential flight control systems.

(2) Four types of hydraulic pumps are used: Engine Driven Hydraulic Pumps (EDHPs), Electric Motor Hydraulic Pumps (EMHP), Air Driven Pumps (ADP), and Ram Air Turbine (RAT). The EDHPs are in the left and right systems. The EMHPs are in all three systems. The ADP and RAT are in the center system.

(3) Other equipment and components for each system includes independent pressurized reservoirs, filters for each pump, and associated valves, gages, and sensors. Rated pressure of each system is 3000 psi. Fluid used is fire resistant BMS 3-11 Type IV, which is purple in color.

(4) Components of the left system are in the hydraulic bay of the left engine strut. Components of the center system are in the right wheel well. Components of the right system are in the hydraulic bay of the right engine strut.



Hydraulic Power Systems - Introduction
Figure 4

F. Hydraulic System - Simplified (Fig. 5)

(1) The left, center, and right hydraulic systems provide power for primary flight control systems: ailerons, rudder, elevators, spoilers, and autopilot. The left and center hydraulic systems provide power for the stabilizer trim system. The center hydraulic system provides power for nose gear actuation, nose gear steering, trailing edge flaps, leading edge slats, and main landing gear. The center and right hydraulic systems provide power for brakes. The left and right hydraulic systems provide power for the thrust reversers.

(a) Each system uses primary and demand pumps. Primary pumps run continuously. Demand pumps run when required.

(b) The left and right systems use EDHPs as the primary pumps. The center system uses 2 EMHPs as primary pumps.

(c) The left and right systems use EMHPs as demand pumps which start when system pressure is low. The center system uses the ADP as a demand pump which starts when system pressure is low and when flaps or landing gear are actuated.

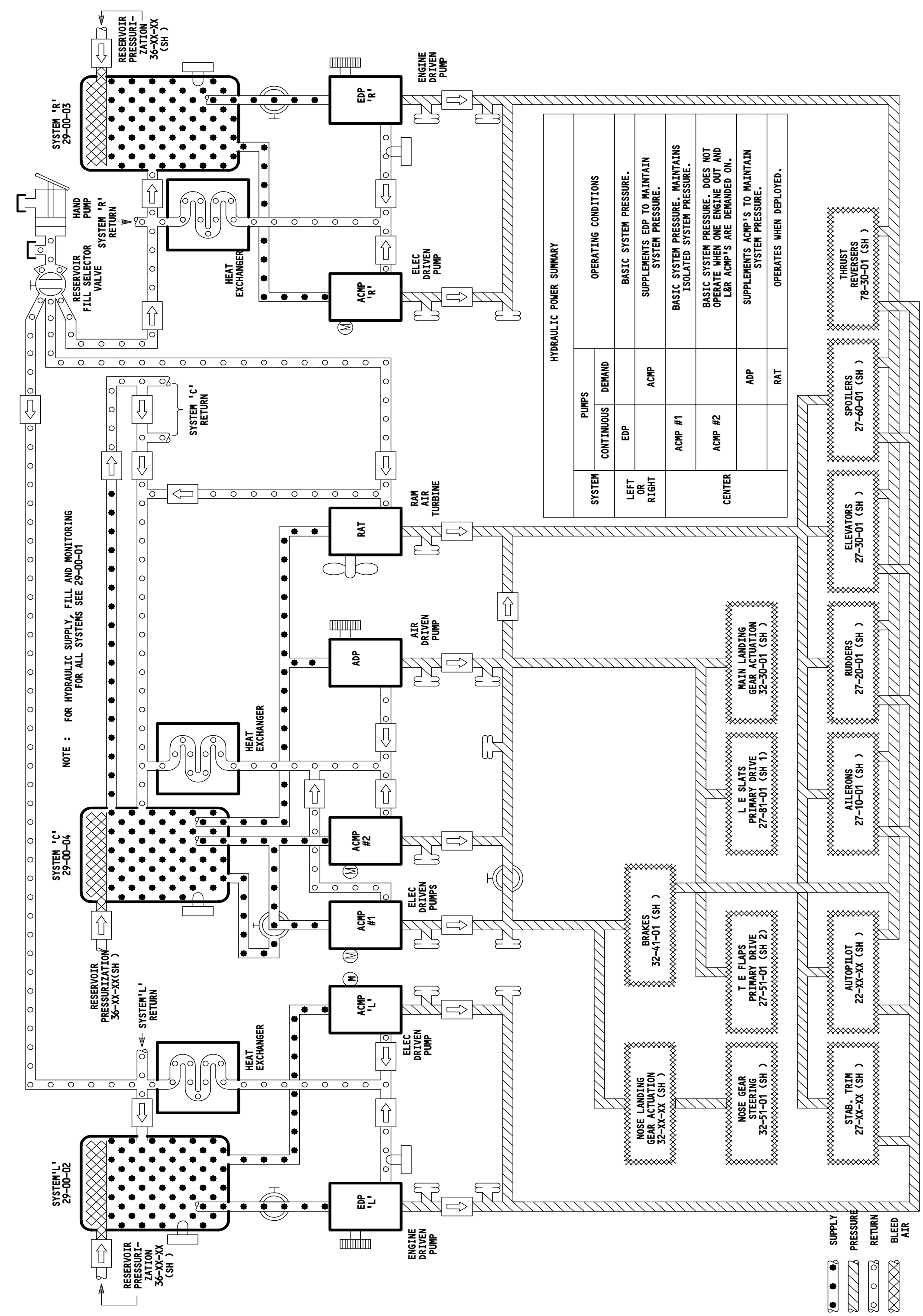
(d) The center system has a RAT for emergency use in the event that both engines fail in flight. The RAT provides hydraulic power for stabilizer trim, autopilot, ailerons, rudder, elevators, and spoilers.

(e) All hydraulic systems have system pressure and temperature sensors. Each pump's pressure is monitored by an independent sensor. Check valves on each pumps output and return lines allow each system to maintain pressure when some of the pumps in the system are not operating. The left and right EDHPs have fluid supply shut-off valves.

(f) Reservoirs can be pressure filled or manually filled. A hand pump and reservoir fill valve (selector) allow filling of the selected reservoir.

G. Left and Right Hydraulic System Components (Fig. 6)

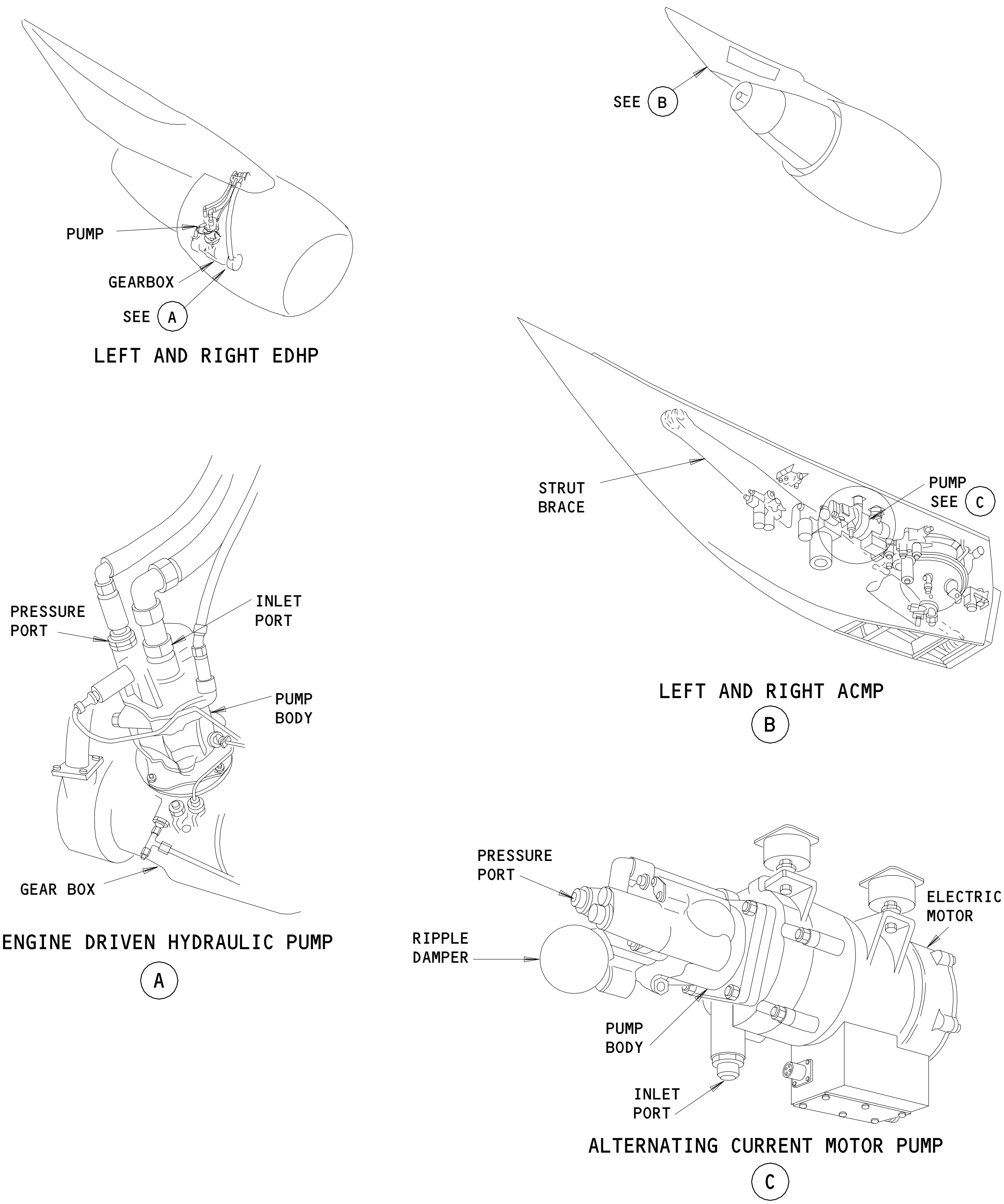
(1) The left and right hydraulic systems each have one engine driven hydraulic pump (EDHP) and one electric motor hydraulic pump (EMHP). The EDHP is attached to the engine gear box on the under side of the engine. The pump is a line replaceable unit (LRU) with a rated output of 37 gpm.



HYDRAULIC POWER SUMMARY

SYSTEM	PUMPS CONTINUOUS	PUMPS DEMAND	OPERATING CONDITIONS
LEFT OR RIGHT	EDP		BASIC SYSTEM PRESSURE.
		ACMP	SUPPLEMENTS EDP TO MAINTAIN SYSTEM PRESSURE.
CENTER	ACMP #1		BASIC SYSTEM PRESSURE. MAINTAINS ISOLATED SYSTEM PRESSURE.
	ACMP #2		BASIC SYSTEM PRESSURE. DOES NOT OPERATE WHEN ONE ENGINE OUT AND L&R ACMP'S ARE DEMANDED ON.
		ADP	SUPPLEMENTS ACMP'S TO MAINTAIN SYSTEM PRESSURE.
		RAT	OPERATES WHEN DEPLOYED.

Hydraulic System Simplified
Figure 5



Left And Right Hydraulic Systems Components
Figure 6

(2) The EMHP is in the strut hydraulic bay. Each bay also contains the associated filters, valves, and reservoir. The electric motor is powered by 115 vac using 45 amps at rated output. The pump is a LRU with a rated hydraulic output of 7 gpm.

H. Center Hydraulic Systems Components (Fig. 7)

(1) The two EMHP primary sources are in the right wheel well. They are mounted on shelves on the keel structure. The associated filters, valves, and reservoir are also in that area. The center EMHPs and identical to the left and right EMHPs.

(2) The Ram Air Turbine (RAT) is in the fairing aft of the right wheel well from which it deploys downward into the slipstream. The RAT is extended and retracted by an electric motor. The swing arm, turbine, and pump are all one LRU.

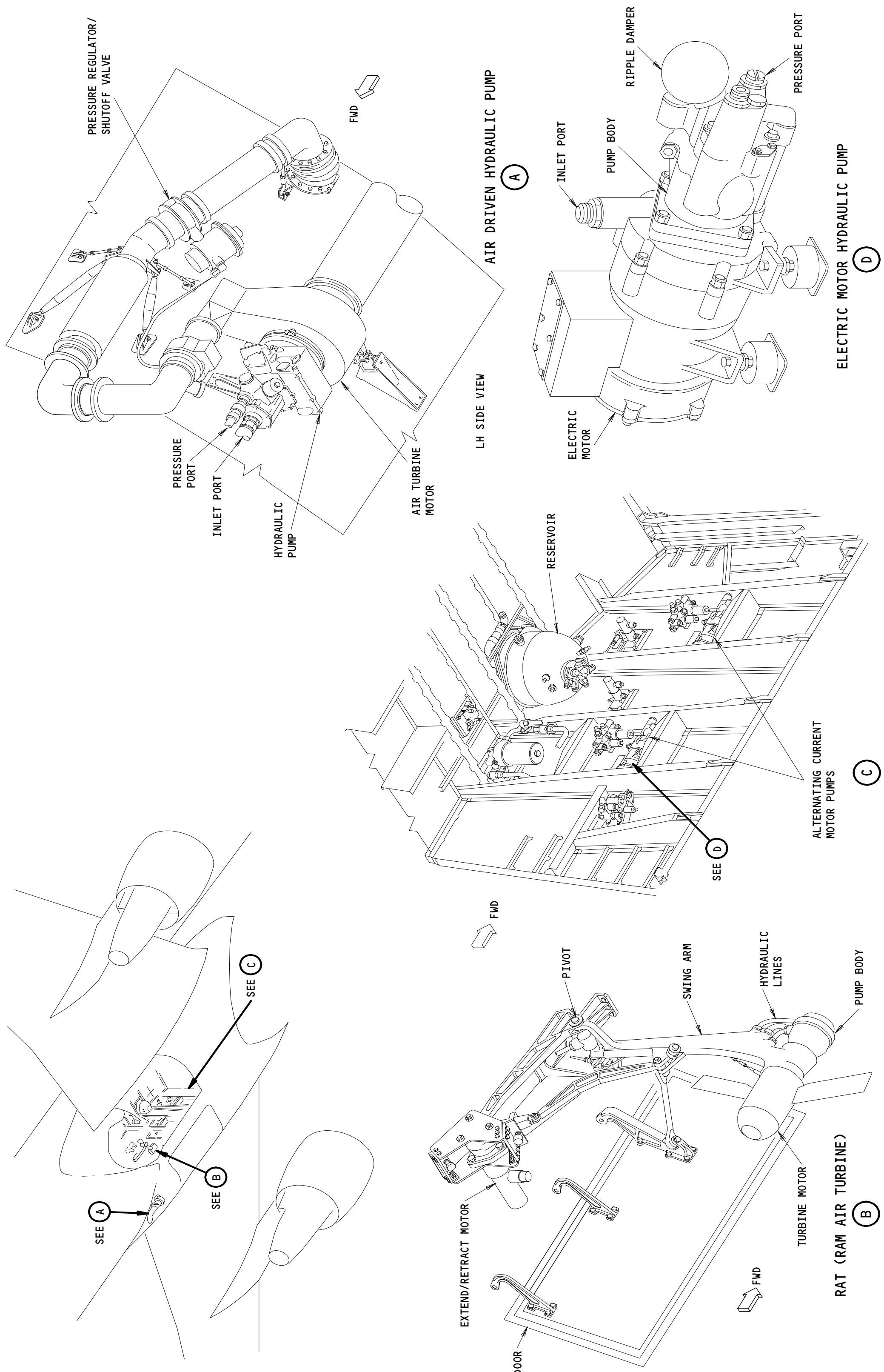
(3) The Air Driven Hydraulic Pump (ADHP) is driven by the pneumatic system through a pressure regulator/shutoff valve. The hydraulic pump is rated at 37 gpm. The unit weighs about 100 pounds.

(4) The hydraulic service panel is aft of the right wheel well. It provides a central capability to service all three system reservoirs.

I. Hydraulic System Control Panel (Fig. 8)

(1) The hydraulic system control panel is on the overhead panel P5. The panel incorporates the switches and indicators required for normal control and monitoring of the three hydraulic systems.

(2) Each system has the following switches and indications:
- (a) System pressure, EICAS display (AMM 31-41-00/001)
- (b) System low pressure warning light (amber)
- (c) Primary pump overheat warning light (1 left, 2 center, 1 right) (amber)
- (d) Primary pump ON/OFF switch (1 left, 2 center, 1 right)
- (e) Primary pump low pressure warning light (1 left, 2 center, 1 right) (amber)
- (f) Demand pump low pressure warning light (amber)
- (g) Demand pump control switch
- (h) System quantity low warning light (amber)
- (i) System reservoir quantity, EICAS display (AMM 31-41-00/001)



Center Hydraulic System Components
Figure 7

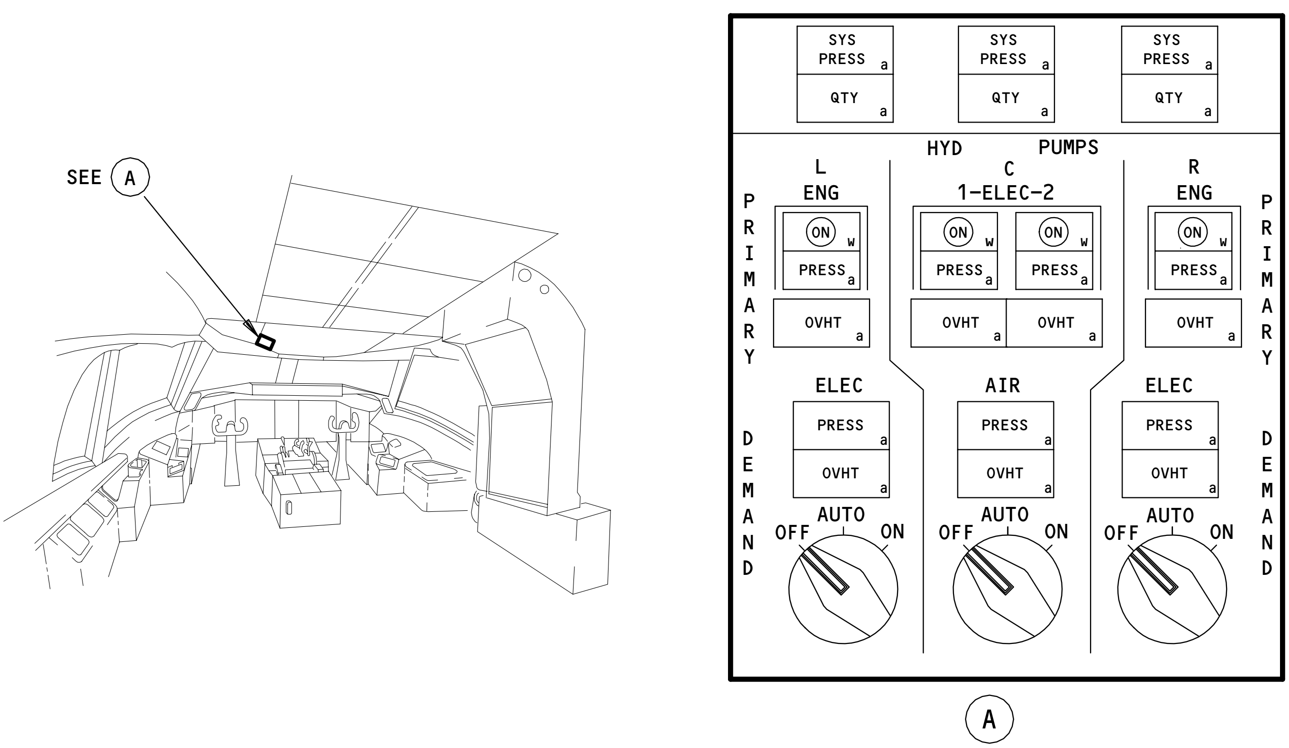
37168

(j) System reservoir temperature, EICAS display (AMM 31-41-00/001)

(3) The primary pump switch/lights are push-on/push-off type. When pushed on the ON legend is illuminated white. The left and right switches control EDHPs. The center switches control EMHPs.

(4) The demand pump switches are 3-position rotary switches normally left in AUTO. The AUTO position allows the pumps to turn on when the demand occurs. The left and right switches control EMHPs. The center switch controls the ADP.

(5) Each pressure, temperature, and quantity indicator has its own independent sensor. All warning lamps are amber.



Hydraulic System Control Panel
Figure 8

J. Hydraulic System Controls (Fig. 9)

(1) Hydraulic Flight Control Panel

(a) The hydraulic flight control panel on P61 contains six push-on/push-off switch/lights. Each switch controls a shutoff valve that selectively interrupts hydraulic power to the flight controls. Three switches isolate hydraulic power to the tail, and three isolate hydraulic power to the wing.

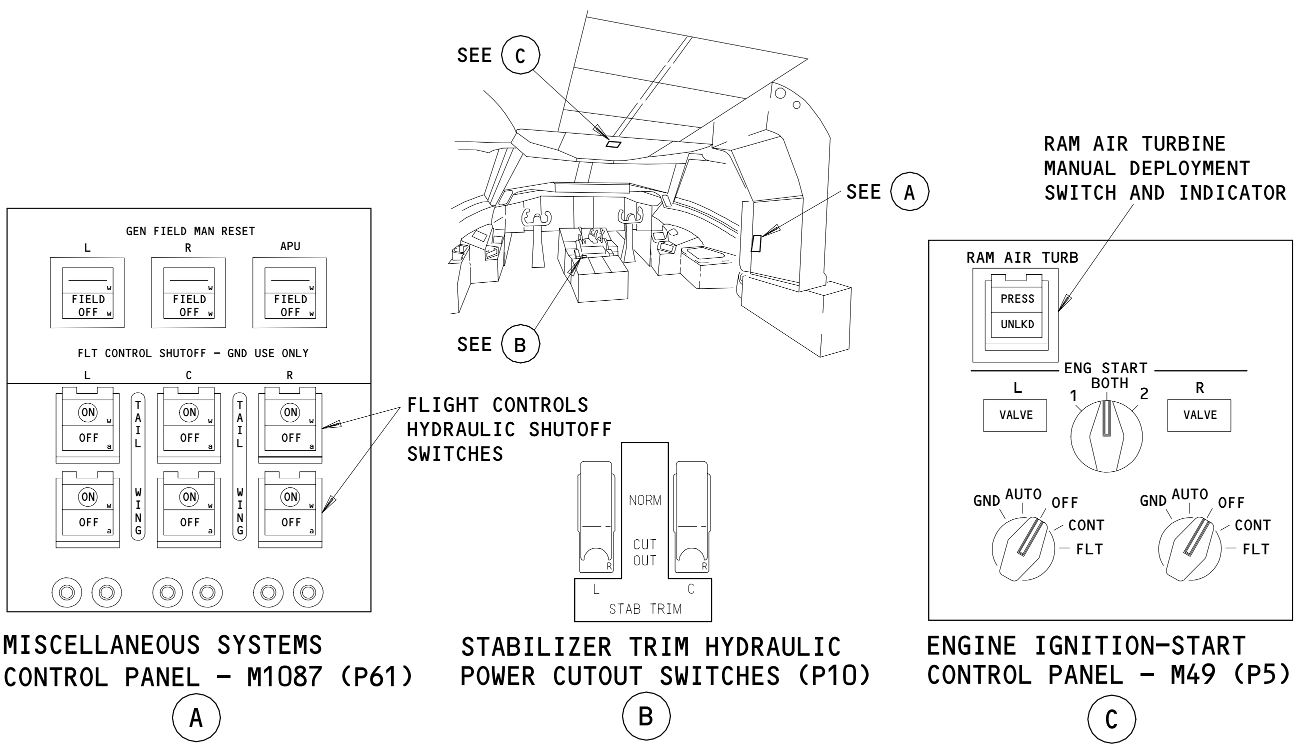
(b) The WING shutoff switches control pressure to all flight control components in the wing, including actuators and servos for ailerons and spoilers.

(c) The TAIL shutoff switches control pressure to all flight control components in the tail, including feel computers, rudder ratio changer, actuators, and servos for the elevators and rudder.

(d) The OFF position of the switch/light illuminates amber when pressed. Isolating hydraulic pressure to the selected areas aids in trouble shooting and allows maintenance without removing all hydraulic power.

(2) Engine Ignition and Start Control Panel

(a) The engine ignition and start control panel incorporates the ram air turbine (RAT) switch/light. Pressing the switch starts an electric motor that extends the RAT. The lower part of the switch/light marked UNLKD (unlocked) illuminates amber. The upper part marked PRESS illuminates green when the RAT is generating at least 1000 psi.



Hydraulic System Controls
Figure 9

(b) Retraction of the RAT is a ground operation. The turbine blades must be centered before the RAT can be retracted. Retraction is done electrically using a control panel in the right wing aft fairing.

(3) Stabilizer Trim Hydraulic Power Cutout Switches

(a) Two stabilizer trim cutout switches (left and right) interrupt hydraulic power to stabilizer trim components in the tail. The switches are guarded to the NORM position. Positioning them to CUTOUT closes the cutout valves, which are part of the stabilizer trim control module. This enables trouble shooting and performance of ground maintenance on the stabilizer trim drive hydraulic motors without removing hydraulic system power.

K. Flight Control - Hydraulic Distribution (Fig. 10)

(1) The left, center and right hydraulic systems provide hydraulic power to the respective left, center and right flight control systems actuators and servos. Hydraulic power required for all flight control systems except flap/slats and stabilizer trim, is routed through tail and wing shut-off valves (SOV). The stabilizer trim system has 2 individual hydraulic power cutout valves, one in the left system and one in the center system.

(2) All three main hydraulic systems provide power for the following components:

(a) Ailerons - Lateral Central Control Actuators (LCCAs) and Power Control Actuators (PCAs)

(b) Spoilers - PCAs

(c) Elevators - PCAs and Elevator Autopilot Servos (EASs)

(d) Rudder - PCAs and Directional Autopilot Servos (DASs)

(3) The Elevator Feel Computers (EFCs) and Yaw Damper Servos (YDSs) hydraulic power is from the left and center systems. The left hydraulic system provides power for the Rudder Ratio Changer (RRC).

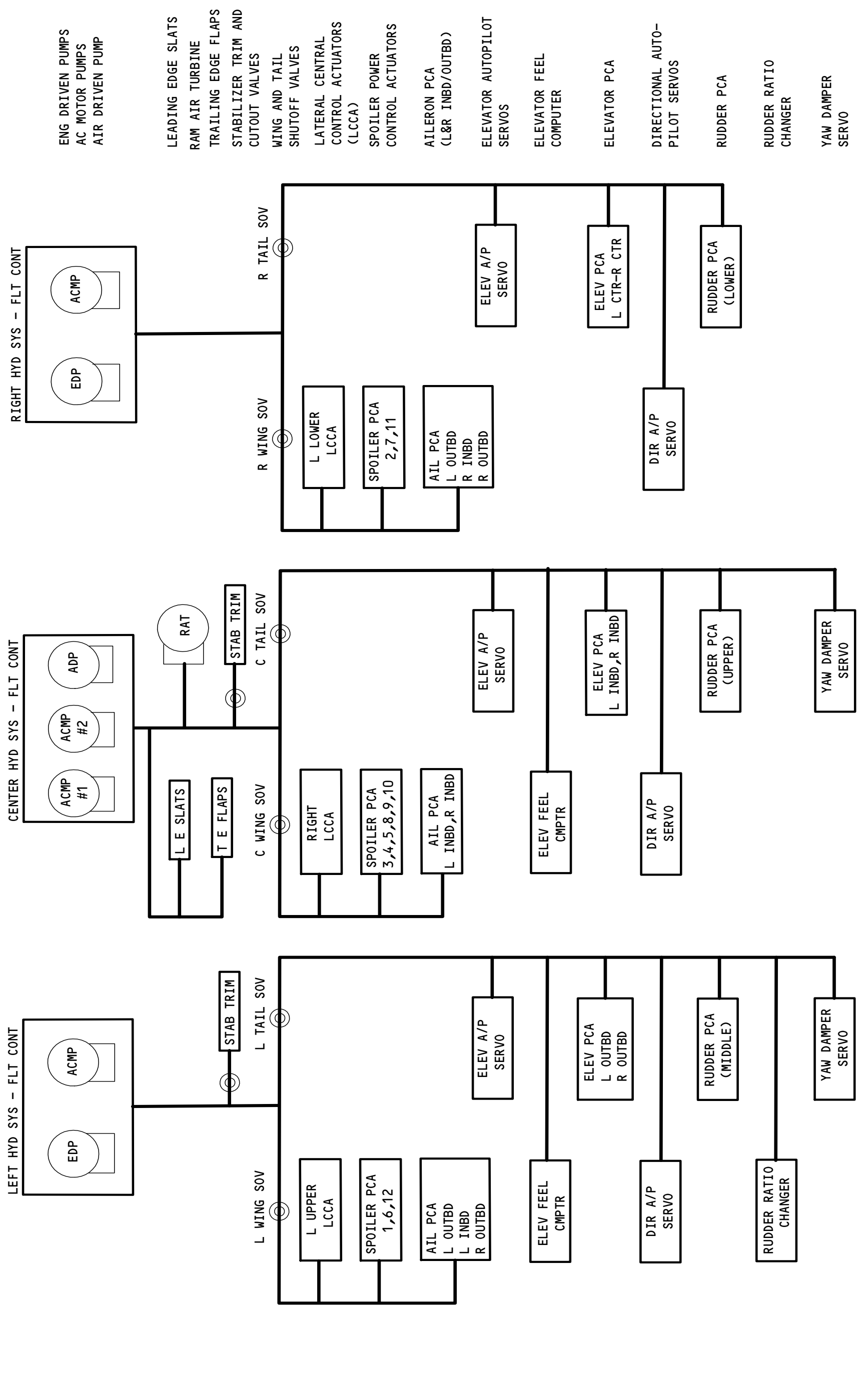
L. Flight Control - Flight Deck Inputs (Fig. 11)

(1) Primary Flight Control Inputs

(a) Primary flight control inputs are made using the control column/wheel, rudder pedals, and trim switches.

(b) The captains's and first officer's control column/wheels provide the manual control cable input for pitch and roll control. They also drive spoiler system transducers located under the flight compartment floor. The spoiler system transducers assist in roll control of the airplane.

(c) The outboard horn of each control wheel has an autopilot disengage switch, stabilizer trim switch, and switch for communication systems. An aileron trim indicator is on the top of each control wheel hub.

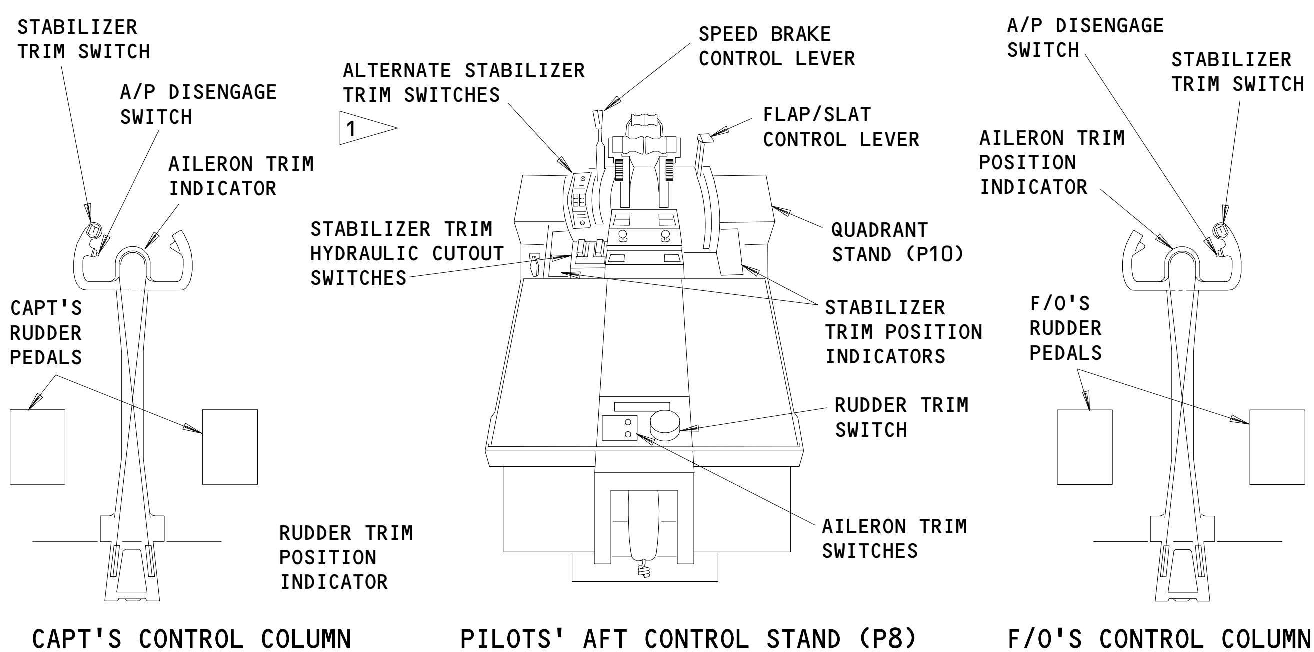


Flight Control - Hydraulic Distribution
Figure 10

(d) The captain's and first officer's rudder pedals provide the manual cable input for yaw control. The autopilot and yaw damper systems also provide yaw control inputs.
(e) The aft section of the control stand (P8) contains aileron and rudder trim switches, and a rudder trim position indicator.

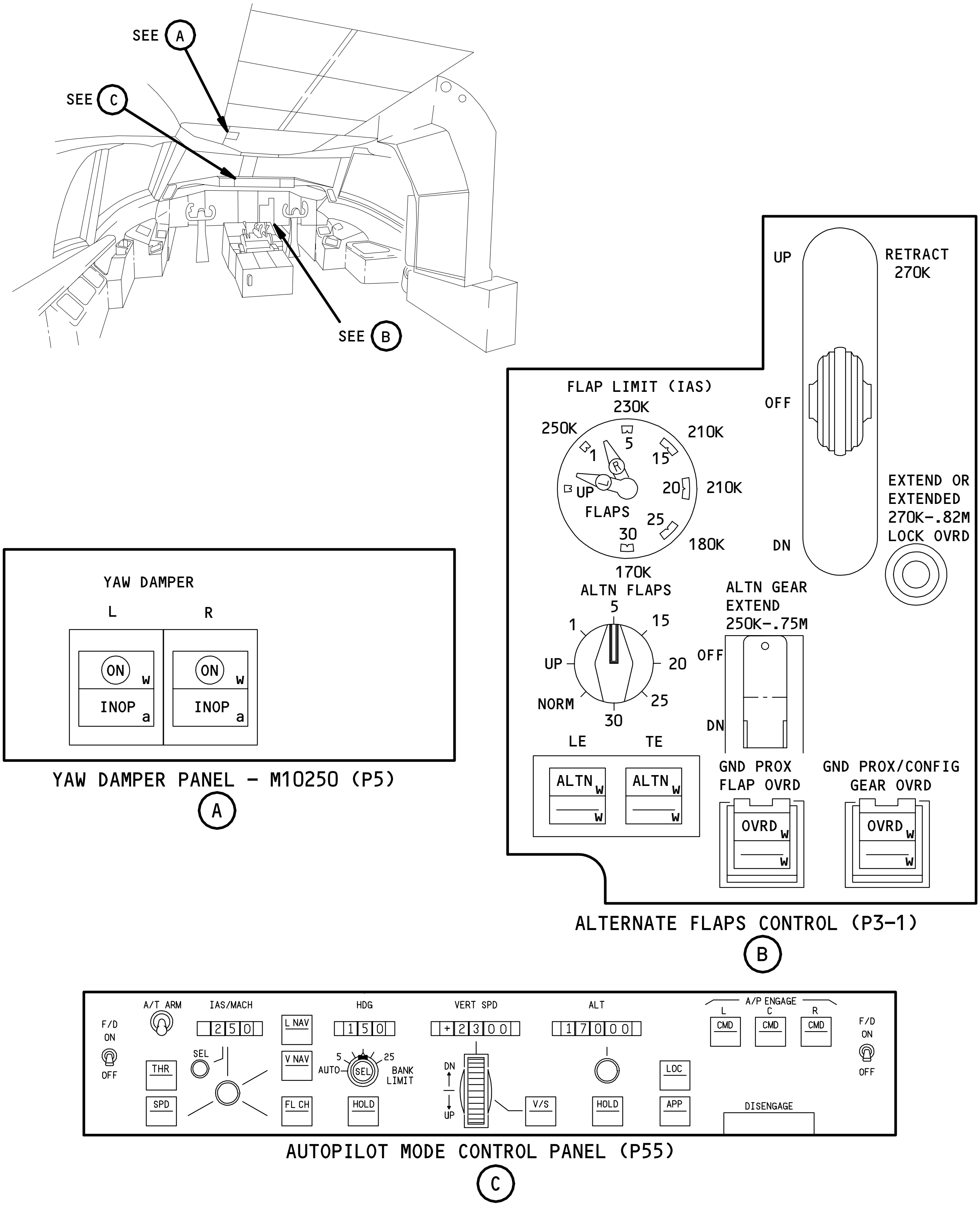
(2) Secondary Flight Control Inputs
(a) The stabilizer trim switches (thumb operated) on each control wheel provide an electrical input to drive the stabilizer up and down.
(b) The center section of the control stand (quadrant stand - P10) incorporates the following items:
1) ON SAS 150-153;
stabilizer trim ARM and CONTROL levers on the left side for manual trimming of the stabilizer.
2) ON ALL MTH AIRPLANES; SAS 050-149, 154-999;
alternate stabilizer trim switches on the left side provide an electrical input to drive the stabilizer up and down.
3) Stabilizer trim position indicators on each side.
4) Stabilizer trim hydraulic power CUTOUT switches (left and center hydraulic systems) on the left side.
5) Speedbrake control lever on the left side for manually extending the speedbrakes.
6) Flap/slat control lever on the right side for manually extending the flaps and slats.

M. Flight Deck Flight Control Panels and Indicators (Fig. 12)



1 ON SAS 150-153, STABILIZER TRIM ARM AND CONTROL LEVERS

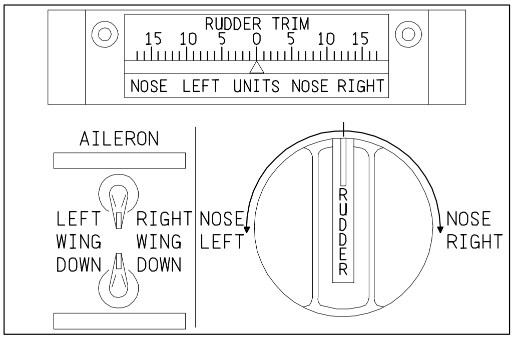
Flight Control - Flight Deck Inputs
Figure 11



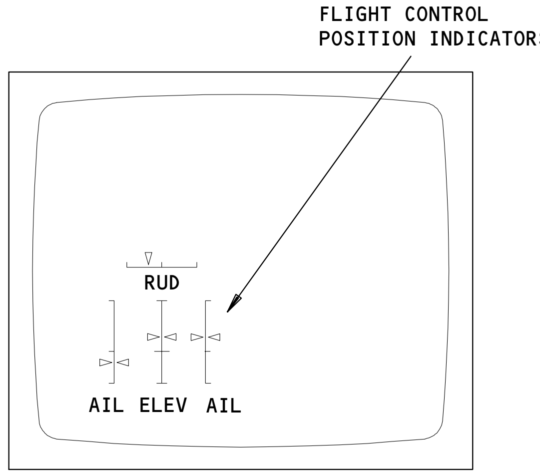
Flight Deck Flight Control Panels and Indicators
Figure 12

(1) Annunciator Lights Panel
  (a) The six amber annunciator lights indicate faulty operation of various flight control systems. The annunciator lights panel is on P5.
(2) Yaw Damper Panel
  (a) The two independent yaw damper systems are controlled by individual push-on/push-off switch/lights. The yaw damper is engaged when the switch translucent on legend illuminates white. The lower part of the switch/light illuminates amber INOP for a yaw damper failure. A YAW DMPR test switch on P61 enables ground testing for the yaw damper system.
(3) Alternate Flaps Control Panel
  (a) Two push-on/push-off switch/lights arm individual electrical control of leading edge slats and trailing edge flaps. The degree of extension of the flaps/slats is controlled with an 8 position rotary switch. The switch is left in NORM for normal hydraulic control of the flaps and slats.
(4) Autopilot Mode Control Panel
  (a) The autopilot mode control panel enables autopilot control of the primary flight control surfaces in various selected modes. Thrust management system arming and mode selection switches are on the left side of the panel. Captain's and first officer's flight director on-off switches at each end of the panel cause flight director commands to be displayed on electronic attitude director indicators (EADIs).

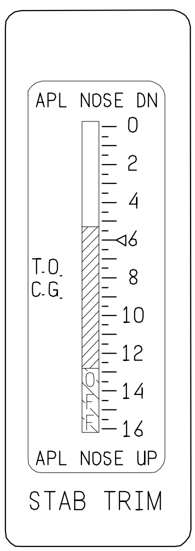
N. Flight Deck Flight Control Indicators (Fig. 13)
(1) Trim Indicators
  (a) Two stabilizer trim indicators are installed, one on each side of the quadrant section of the control stand (P10). The indicators have a green band and units of stabilizer position. The green band indicates the safe takeoff stabilizer position range. The units indicate relative stabilizer position for airplane nose-up and nose-down. The indicators are driven by position transmitters at the stabilizer.
  (b) Rudder trim position is on the aft electronics control panel P8. The indicator is electrically driven by position transmitters at the rudder.



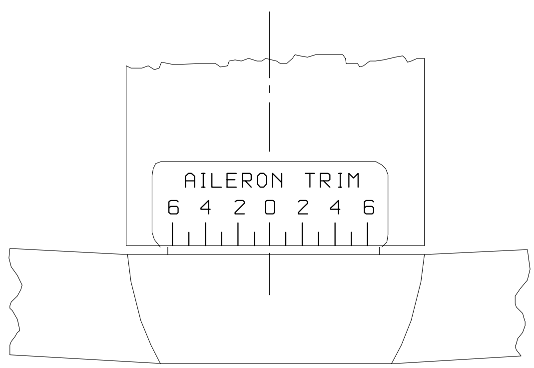
RUDDER TRIM POSITION INDICATOR
RUDDER TRIM SWITCH
AILERON TRIM SWITCHES
(P8)



LOWER EICAS DISPLAY (P2)



STABILIZER TRIM
POSITION INDICATORS
(EXAMPLE)
(2) (P10)



AILERON TRIM
POSITION INDICATOR
(CONTROL WHEEL)

Flight Deck Flight Control Indicators
Figure 13

(c) Aileron trim is indicated on top of the hub of each control wheel. Trim is indicated by the physical offset of the control wheel.

(2) Aileron and Rudder Trim Switches

(a) Aileron and rudder trim switches are on the aft section of the control stand (P8). The aileron trim switches (2) are spring loaded to the center position. When both are held toward LEFT WING DOWN or RIGHT WING DOWN, they activate arm and control signals that cause the ailerons to be trimmed.

(b) The rudder trim switch is a single rotary switch spring loaded to the center position. When held toward NOSE LEFT or NOSE RIGHT, it activates signals that cause the rudder to be trimmed.

(c) The switches cause trim motors to move cam/roller mechanisms. These change the center position of the control system.

(3) Flight Control Position Indicators

(a) The flight control position indicators are on the lower EICAS display unit when the status page is selected (Ref 31-41-00). The pointers indicate the position of each aileron (4), elevator (2), and rudder (1). A position transmitter at each of the following control surfaces drives the corresponding pointer on the indicators:

1) Inboard ailerons
2) Outboard ailerons
3) Left elevator
4) Right elevator
5) Rudder

O. Flight Control System Electronics (Fig. 14)

(1) The two Control System Electronic Units (CSEU) consist of 8 separate modules in the left system and 8 separate modules in the right system (Ref Chapter 27). The modules are on the electronic equipment racks in the main electrical/electronics equipment compartment.

(2) The modules drive flight control actuators and stabilizer trim control modules. Sensor inputs are from electrical, mechanical, and information (as shown) systems. The CSEU modules act as buffers for inputs, define required control movements, and then modify and program the inputs to provide the output command signal.

(3) The two power supplies provide power monitoring and necessary voltages for each module.

(4) The stabilizer trim/aileron lockout module (SAM) has dual functions. It processes autopilot automatic trim signals, manual electric trim signals, and drives the stabilizer to relieve elevator loads and maintain pitch authority. At cruise speeds, it operates the outboard aileron lockout mechanism to prevent movement of the outboard ailerons which may put undesirable loads at the wing tip.

(5) The rudder ratio changer module increases rudder travel limits as airspeed is reduced. This maintains rudder authority at low airspeeds.

(6) The yaw damper module controls the rudder to maintain directional control during gusting and dutch roll frequencies. It also provides turn coordination when making turns at low airspeeds.

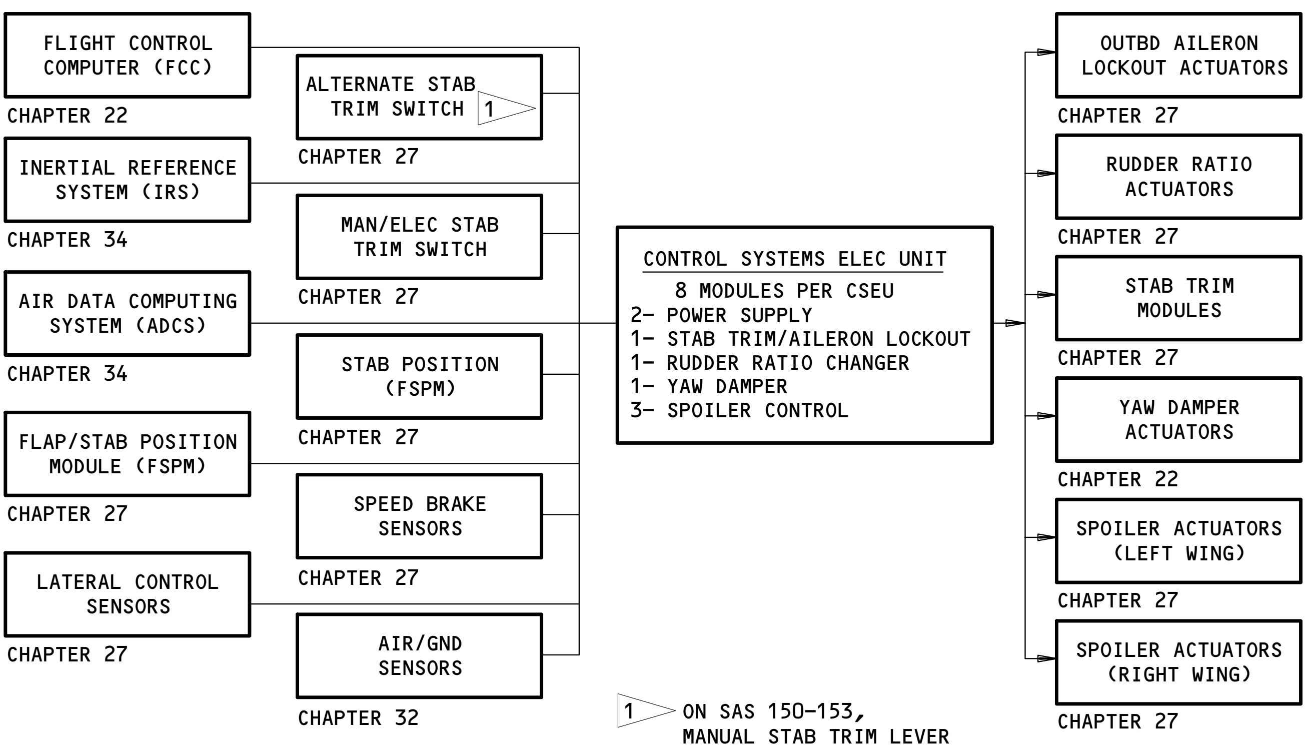
(7) The three spoiler control modules drive spoiler power control actuators which augment roll authority and act as speed brakes.

P. Hydraulic Power Actuators and Servos (Fig. 15)

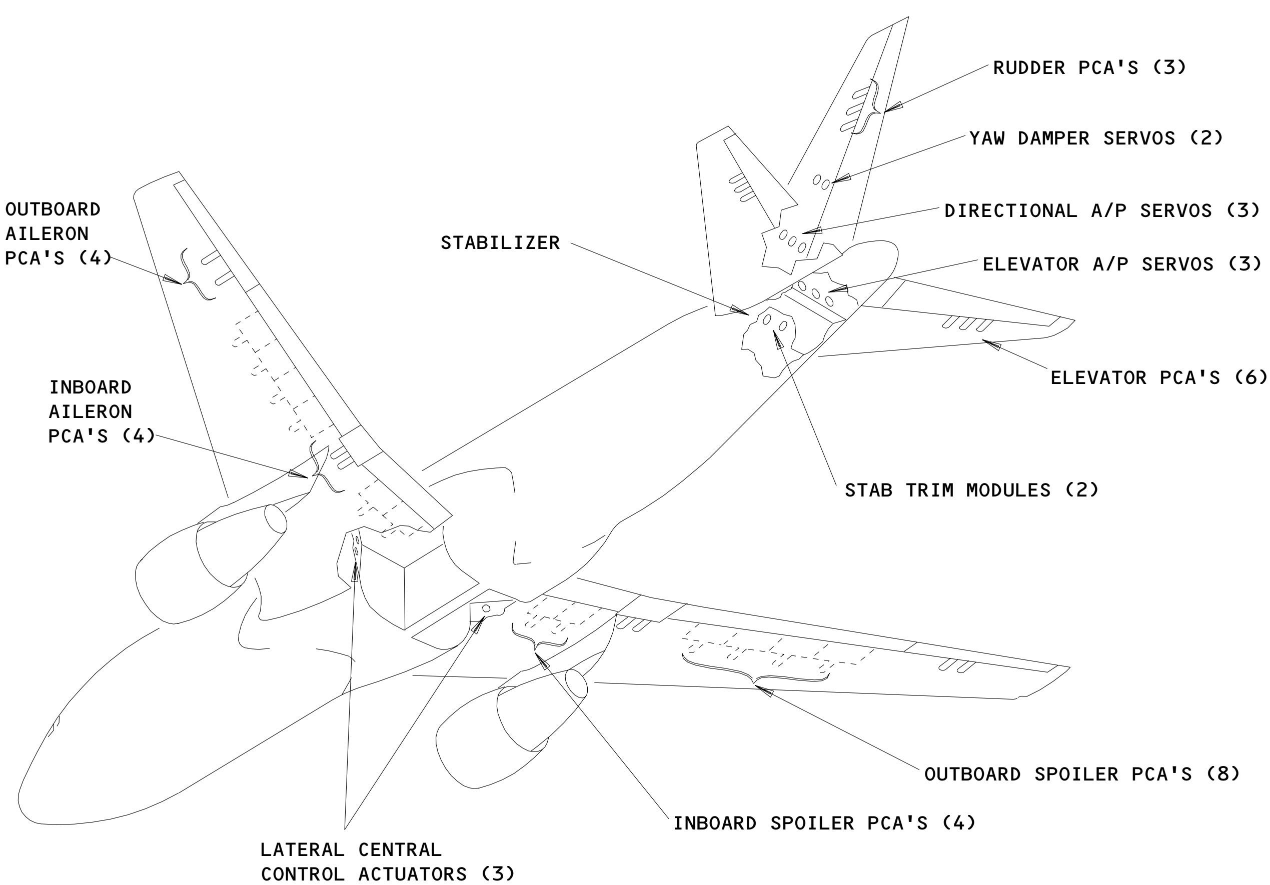
(1) Hydraulic Actuators for Manual Control

(a) Elevators, ailerons, spoilers, and rudder are driven by Power Control Actuators (PCAs). Each PCA uses hydraulic power from one of the three main hydraulic systems.

1) Each elevator (2) uses three PCAs.



Flight Control System Electronics
Figure 14



PCA = POWER CONTROL ACTUATOR

Hydraulic Power Actuators and Servos
Figure 15

2) Each aileron (4) uses two PCAs.
3) Each spoiler (12) uses one PCA.
4) The rudder uses three PCAs.

(b) Mechanical aileron control is boosted by three Lateral Central Control Actuators (LCCAs) on the wing aft spar. Two LCCAs are at the left wing root, one above the other. The third LCCA is at the right wheel well.

(2) Hydraulic Actuators For Automatic Stability Control

(a) Two Yaw Damper System (YDS) servos provide stability control through the rudder. The YDS servos are in the vertical fin below the rudder PCAs.

(3) Hydraulic Actuators For Autopilot Control

(a) Autopilot control of the ailerons is through the LCCAs. The LCCAs use mechanical inputs or autopilot electrical inputs to drive the aileron PCAs.

(b) Three Elevator Autopilot System (EAS) servos drive the elevator PCAs. The EAS servos are aft of the stabilizer.

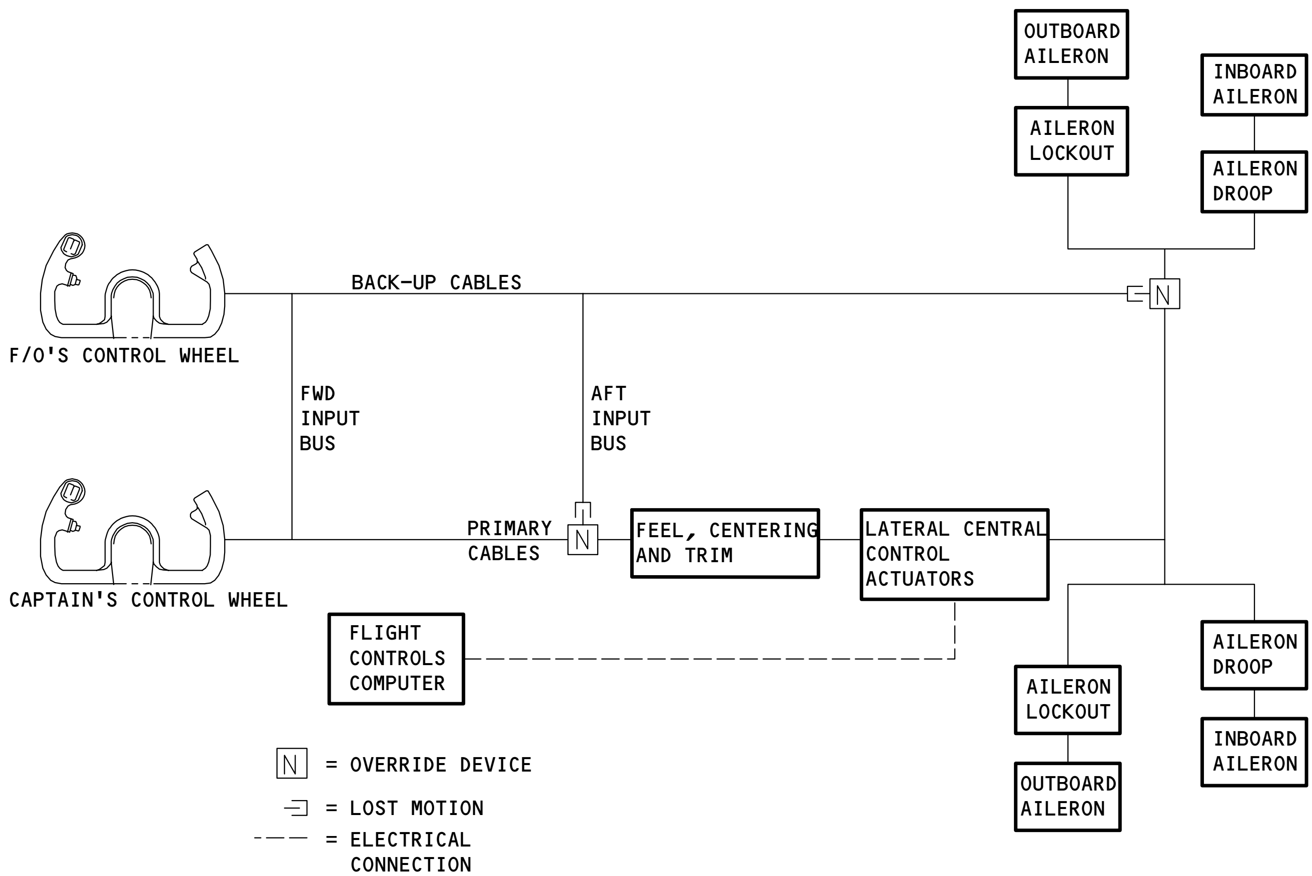
(c) Three Directional Autopilot System (DAS) servos drive the rudder PCAs. The DAS servos are in the vertical fin.

(d) The PCAs are the same one used with manual control of the airplane.

Q. Aileron Control System (Fig. 16)

(1) The primary input for roll control is from the captain's control wheel. The captain's input bus and drive drums move input torque tube assemblies. Linkages from the input torque-tube assemblies provide the input to three LCCAs. The LCCAs then drive output torque-tube assemblies that control the cables to the aileron PCAs through an aileron droop mechanism and outboard aileron lockout mechanism.

(2) The first officer's control wheel drives the primary system with the cables between the captain's and first officer's bus drum assemblies. Inputs are conditioned by the feel, centering, and trim assembly connected to the captain's input torque-tube assembly. Back-up roll control is through override devices, which bypass the LCCAs, connected to the right aileron control quadrant. Lost motion devices allow primary system motion before the back-up system operates.



Aileron Control System
Figure 16

(3) Inboard ailerons droop when the flaps are extended. Aileron drop fills the gap between the inboard and outboard flaps. This reduces drag and maintains flap effectiveness. Outboard ailerons are locked out (disabled) at cruise speeds. This prevents overcontrol and application of unnecessary loads at the wing tip which may damage the structure. An aileron position transmitter at each aileron drives the position indicator in the control cabin.

R. Lateral Central Control Actuators (Fig. 17)

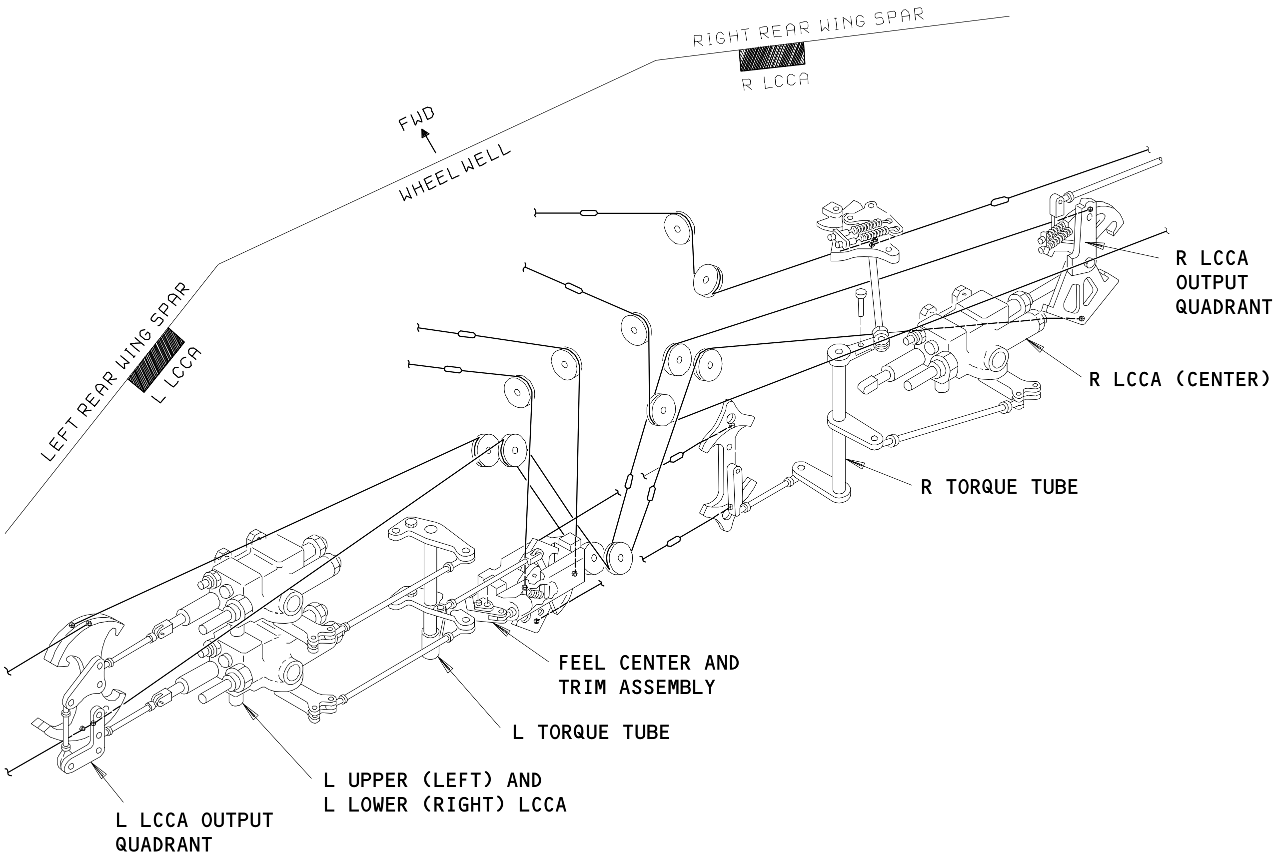
(1) Three Lateral Central Control Actuators (LCCAs) are on the rear wing spars in the wheel wells. Two are in the left wheel well, one above the other. The upper LCCA uses hydraulic power from the left hydraulic system. The lower LCCA uses hydraulic power from the right hydraulic system. The third LCCA is in the right wheel well and uses hydraulic power from the center hydraulic system. The LCCAs move the aileron control cable system in the wings.

(2) The LCCA responds to manual inputs to the input crank and to electrical autopilot inputs. The LCCA weighs 35 lbs. and consists of:

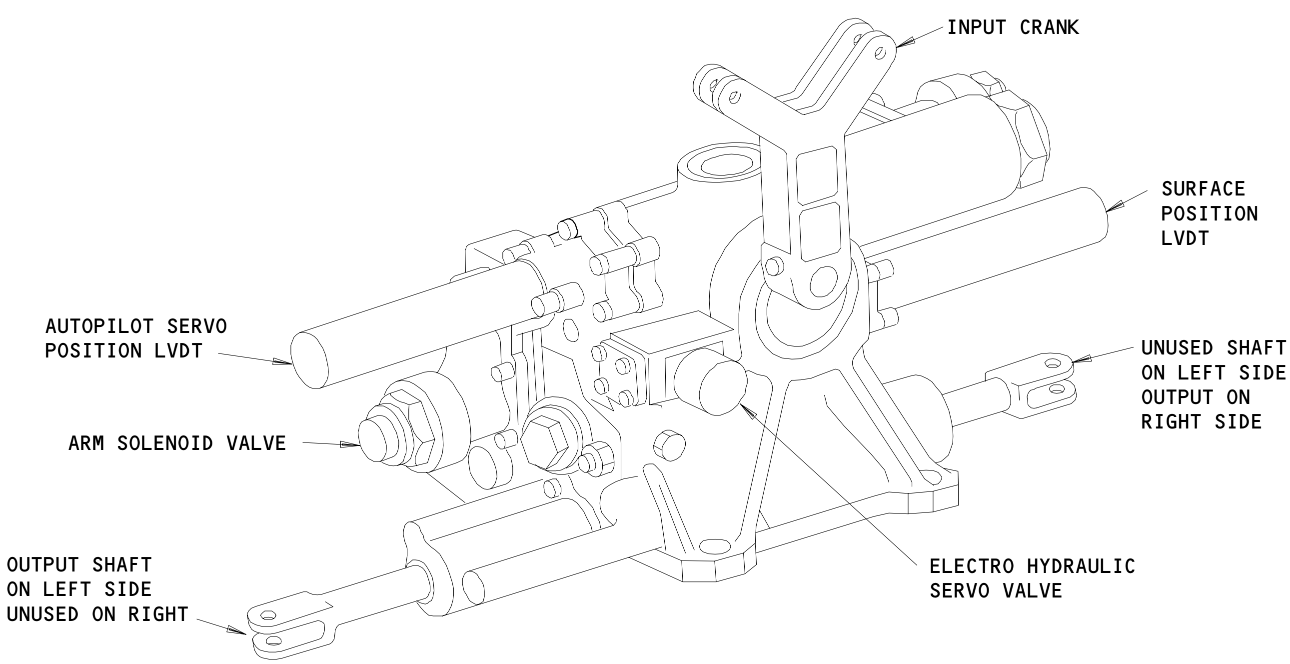
(a) Input crank
(b) Internal control valve, autopilot piston, detent pistons, output piston, and associated linkage
(c) Two linear variable differential transformers (servo position and surface position LVDTs)
(d) Electrohydraulic servo valve (EHSV)
(e) Two electrohydraulic solenoid valves
(f) Filter

(3) Movement of the input crank causes an internal control valve to port hydraulic fluid to one side or the other of a piston on the output shaft. Internal linkage neutralizes the control valve when the output shaft position corresponds with the input crank position.

(4) The autopilot applies arm and engage voltages to the solenoid valves. When the autopilot is engaged, the solenoid valves port hydraulic fluid to the EHSV and to the detent pistons. The detent pistons clamp the autopilot piston (servo) to the internal linkage. Autopilot command signals cause the EHSV to port hydraulic fluid to one side or the other of the autopilot piston, which then moves the input crank. Operations of the LCCA is then as described in the previous paragraph.



WHEEL WELL COMPONENTS



Lateral Central Control Actuators
Figure 17

(5) The LVDTs provide a feedback voltage to the autopilot representing actuator piston (servo) position and output shaft (surface) position. The actuator servo position LVDT feedback nulls the autopilot command when the servo has reached a position corresponding to the initial autopilot command.

S. Lateral Control Feel, Centering and Trim (Fig. 18)

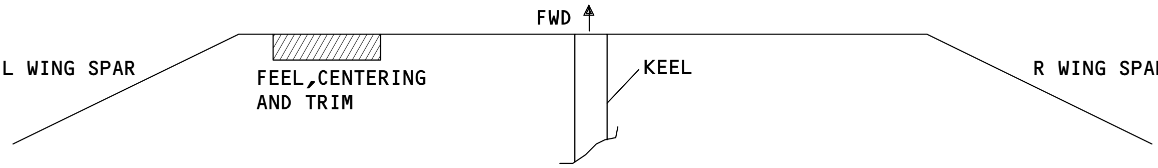
(1) The lateral control feel, centering and trim assembly is part of the left aft quadrant. It is mounted on the lower part of the left wheel well forward bulkhead. The assembly provides a sense of feel and centering for the pilots' control wheels. Artificial feel is necessary because aerodynamic pressure is interrupted by the hydraulic power control actuators.

(2) Feel is provided by a moving cam with a spring loaded cam follower. The cam is on the aft quadrant shaft, which is rotated by cables to the control wheels. The follower is a roller referenced to structure through the trim actuator. Movement of the aft quadrant shaft and cam is resisted by the spring loaded cam follower. The cam causes the follower to stretch the spring, and provide a centering force.

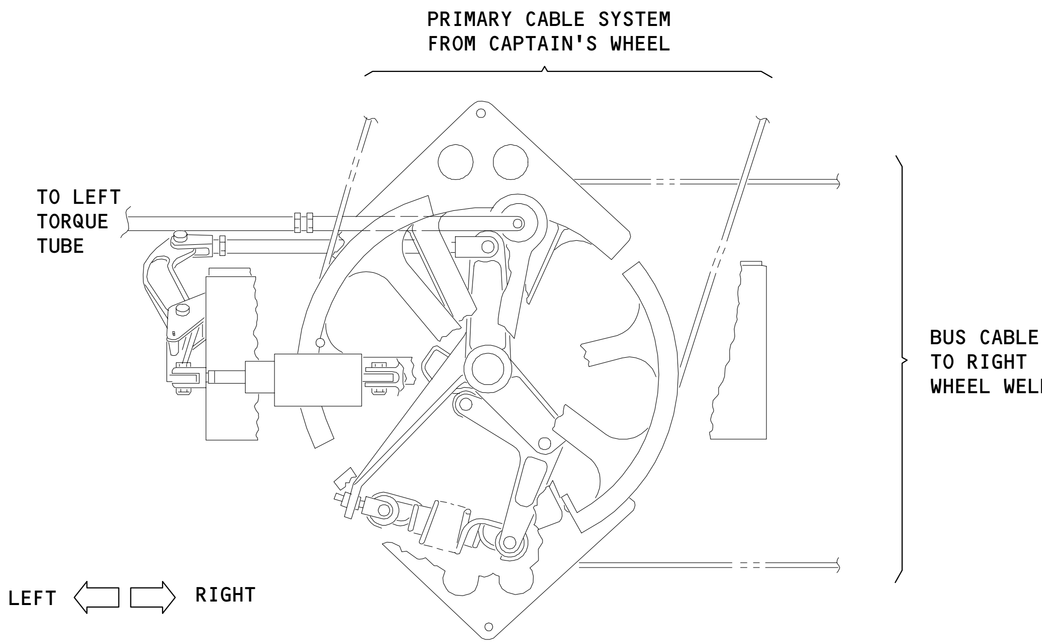
(3) The trim actuator changes the relative position of the cam follower, thereby changing the center position of the cam. The actuator is an electric motor with a jackscrew drive shaft. Trim centering occurs when the actuator moves the spring loaded cam follower causing the roller to ride up the sides of the cam. This increases tension on the spring and the cam. The quadrant and control wheels relieve this spring tension by centering the cam on the roller.

T. Outboard Aileron Lockout Device (Fig. 19)

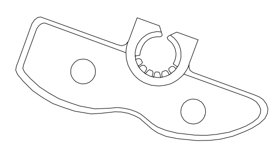
(1) Aileron lockout eliminates outboard aileron movement at cruise airspeeds. It is controlled by the stabilizer trim/aileron lockout module (SAM) of the CSEU modules. Lockout is based on airspeed, Mach, and altitude inputs from the Air Data Computer (ADC) to the SAM. Lockout is at 235 to 275 knots or .50 to .52 Mach, depending on altitude. The lateral control system continues to operate normally, except the outboard power control actuators (PCAs) are not operated.



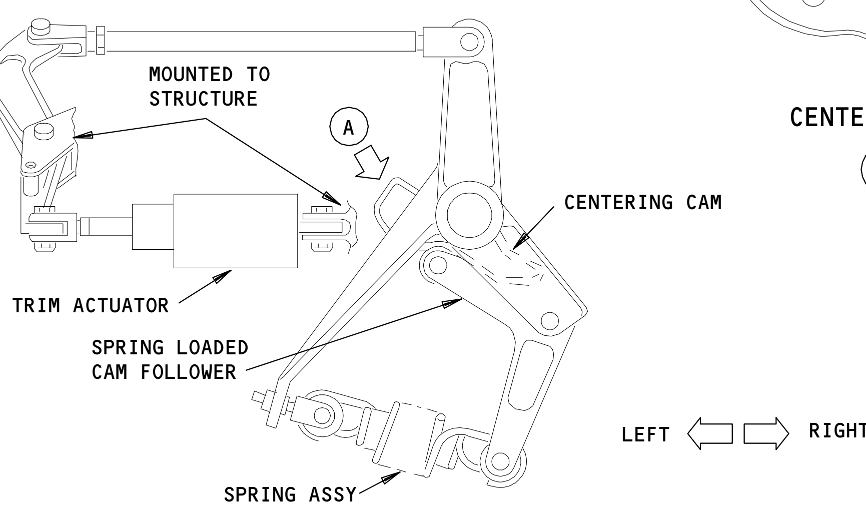
LEFT WHEEL WELL



LEFT WHEEL WELL QUADRANT
(LEFT AFT QUADRANT)

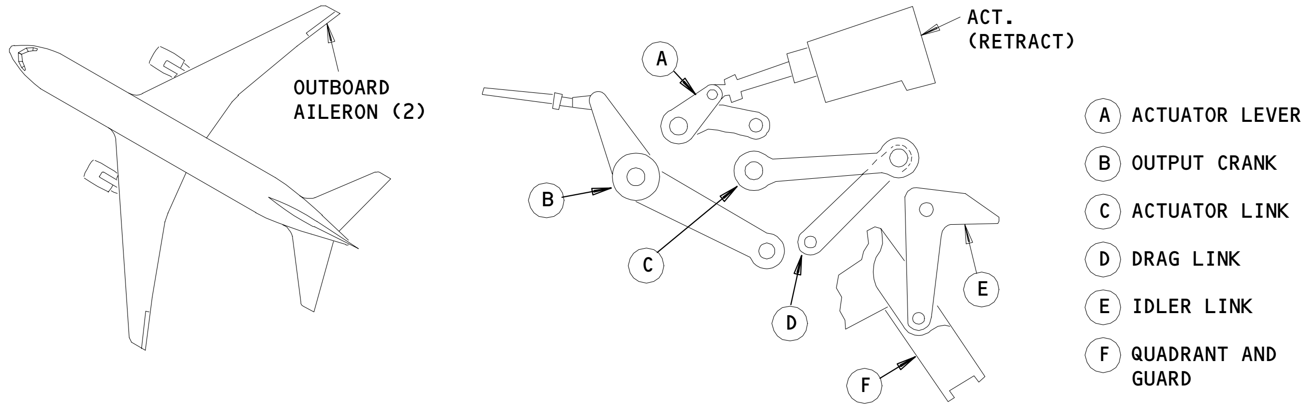


CENTERING CAM
A

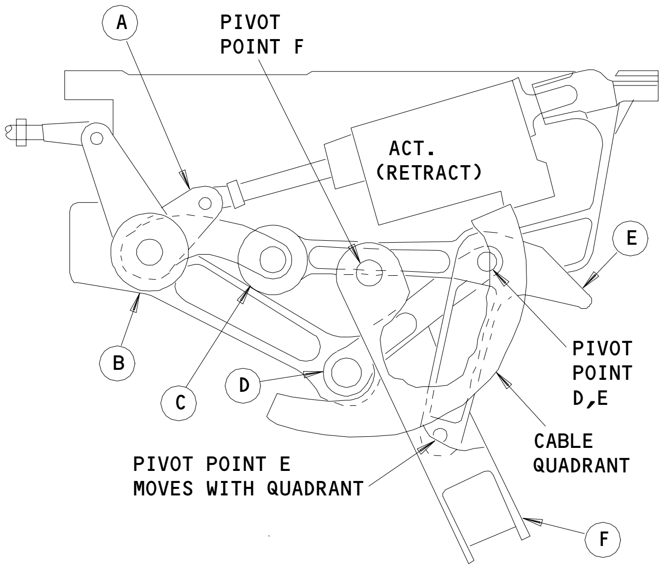


FEEL,CENTERING & TRIM ASSY

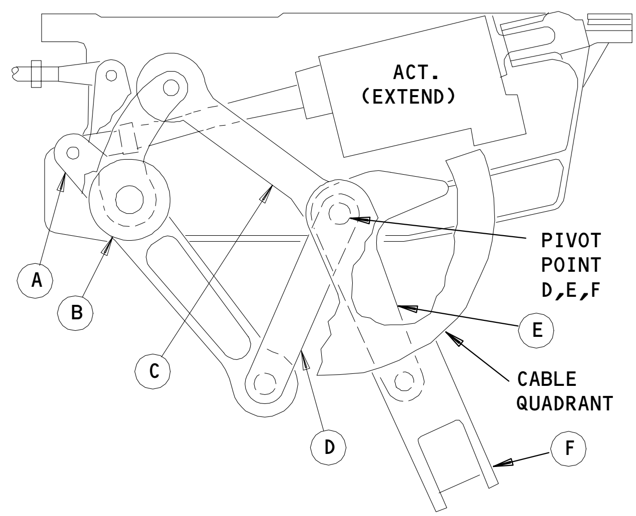
Lateral Control Feel, Centering & Trim
Figure 18



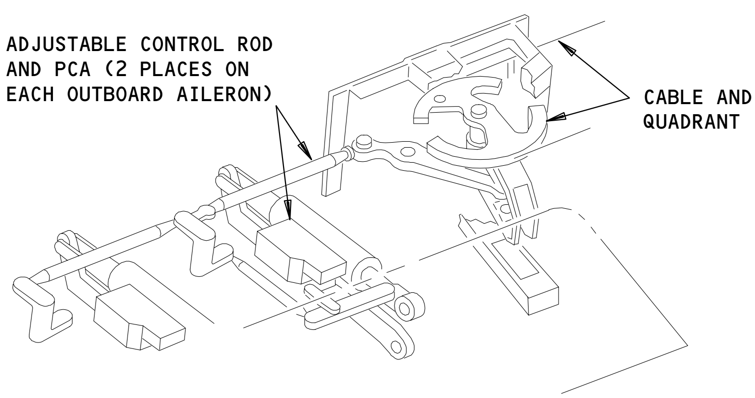
LOCKOUT LINKAGE



LOCKED IN - CONTROL WHEEL RIGHT



LOCKED OUT - CONTROL WHEEL RIGHT



LEFT AILERON LOCKOUT (TYP)

Outboard Aileron Lockout Device
Figure 19

(2) The SAM controls an electric motor, which drives an actuator lever and link between the cable quadrant and the PCA control rods. When the electric motor jackscrew extends, the actuator lever and link pivot until the idler link pivot point axis aligns with the quadrant pivot point axis. The output crank drives the PCAs until the outboard aileron is in the faired position. Quadrant rotation then moves only the idler link.

(3) The SAM also controls unlocking, reversing the actuator, and linkage movement. The position of the linkage is such that the aileron is either locked-out or locked-in with no intermediate positions. Lockout malfunctions are annunciated on the annunciator panel on overhead panel P5.

U. Spoiler RVDTs (Fig. 20)

(1) Spoiler RVDTs

(a) One Rotary Variable Differential Transformer (RVDT) is on each forward quadrant. Each RVDT senses control wheel movement and provides independent electrical signals to three Spoiler Control Modules (SCMs) (CSEU modules). The signals are processed by the SCMs to command spoiler movement that augments roll control. The spoiler RVDT primary winding uses 26 Vac from the CSEU power supplies.

V. Aileron Control System Operation (Fig. 21)

(1) The relative position of components previously discussed is shown with interconnecting cables and push-rods. Control wheel movement is either by manual force, or is back-driven by a LCCA. Any one LCCA can drive the primary system through interconnects in the wheel well. Aerodynamic feel is artificially applied by the feel, trim, and centering assembly, since hydraulic force overcomes aerodynamic forces.

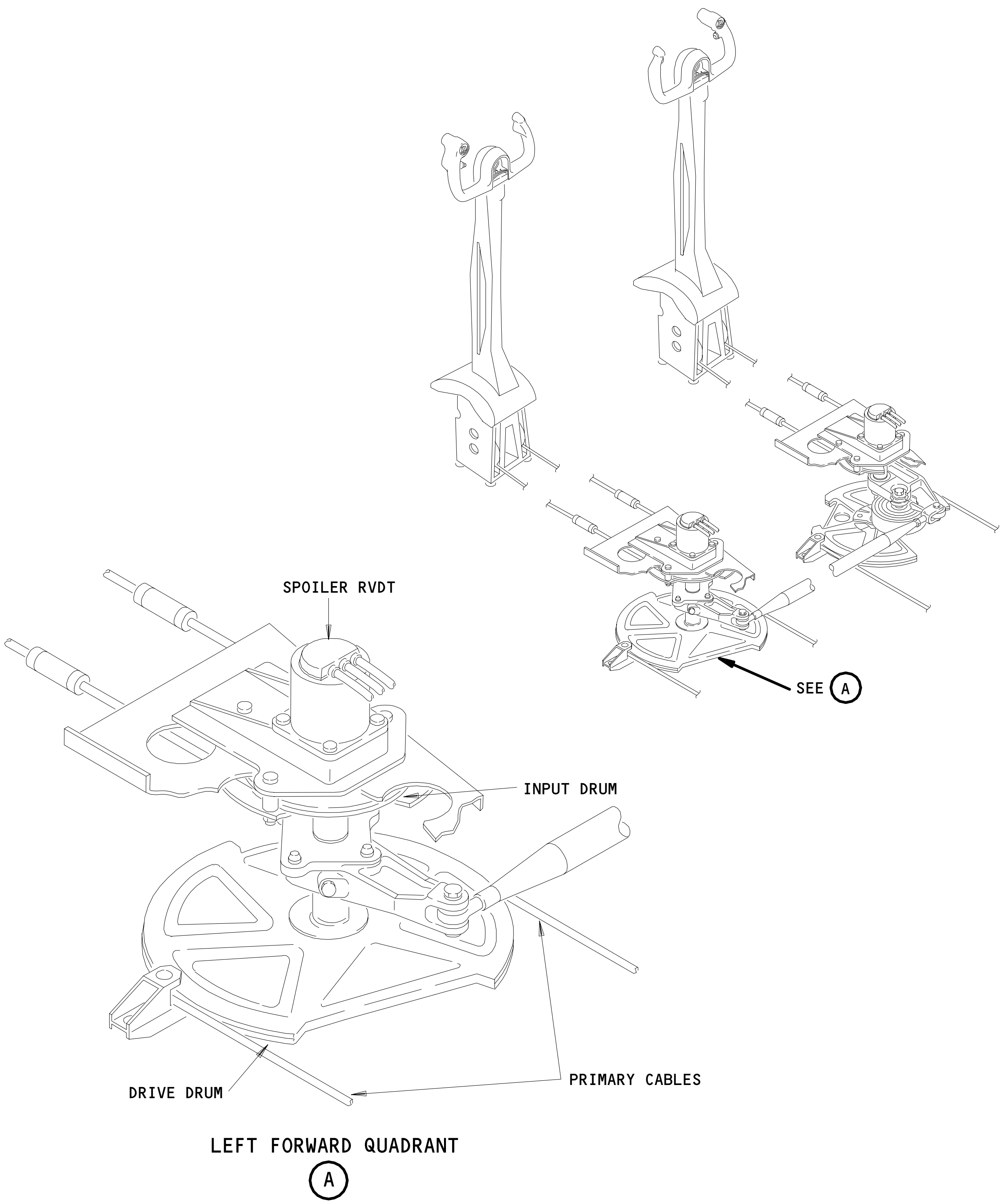
(2) Components and operational functions in each wing are as follows:

(a) Inboard ailerons drop 10 degrees when flaps extend from 5 to 15 units.

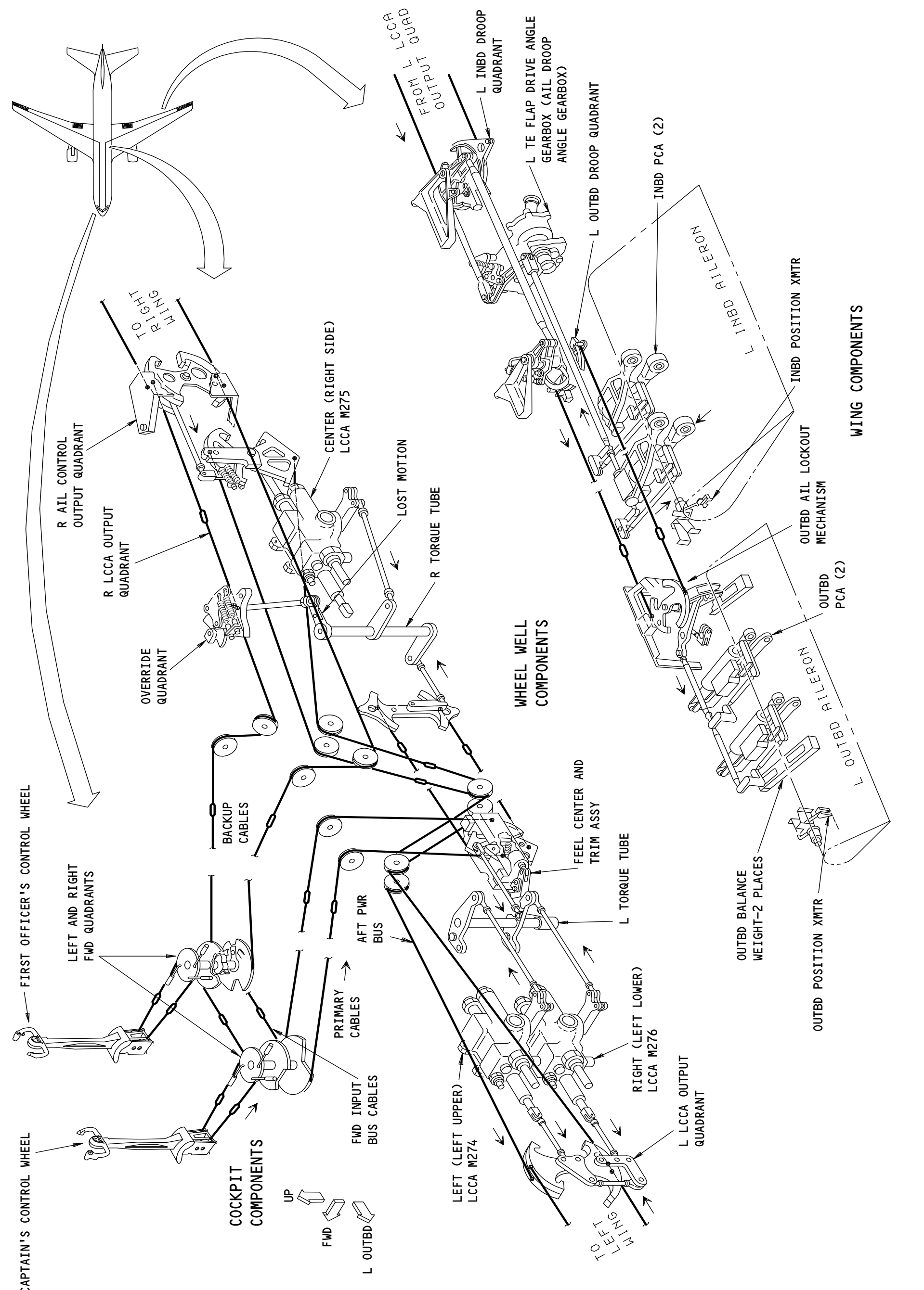
(b) Inboard aileron is powered by two Power Control Actuators (PCAs).

(c) Inboard aileron position transmitter drives an indicator on the lower EICAS display unit.

1) Outboard aileron lockout mechanism is driven by the SAM (CSEU module).



Spoiler RVDT
Figure 20



Aileron Control System-Operation
Figure 21

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2) Outboard aileron is powered by two PCAs.
3) Outboard aileron position transmitter drives an indicator on the lower EICAS display unit.

(d) In the event that the primary control system should jam, backup cables bypass the LCCAs to drive the PCAs directly. Each PCA and LCCA uses one of the three main hydraulic systems. Two separate hydraulic systems supply power for each pair of PCAs.

W. Spoiler Control System (Fig. 22)

(1) Twelve spoilers are on the airplane wings. Spoilers one thru six are on the left. Spoilers seven thru twelve are on the right. Roll control augmentation starts at 6.3 degrees of control wheel rotation. Inputs from RVDTs on the forward quadrants are proportional to control wheel movement. The inputs are processed by six spoiler control modules (SCMs) (CSEU modules).

(2) Each SCM drives two PCAs, one per spoiler. Feedback from each PCA LVDT provides panel position to the SCMs. The SCMs command appropriate spoiler deflection in symmetrical pairs as follows:

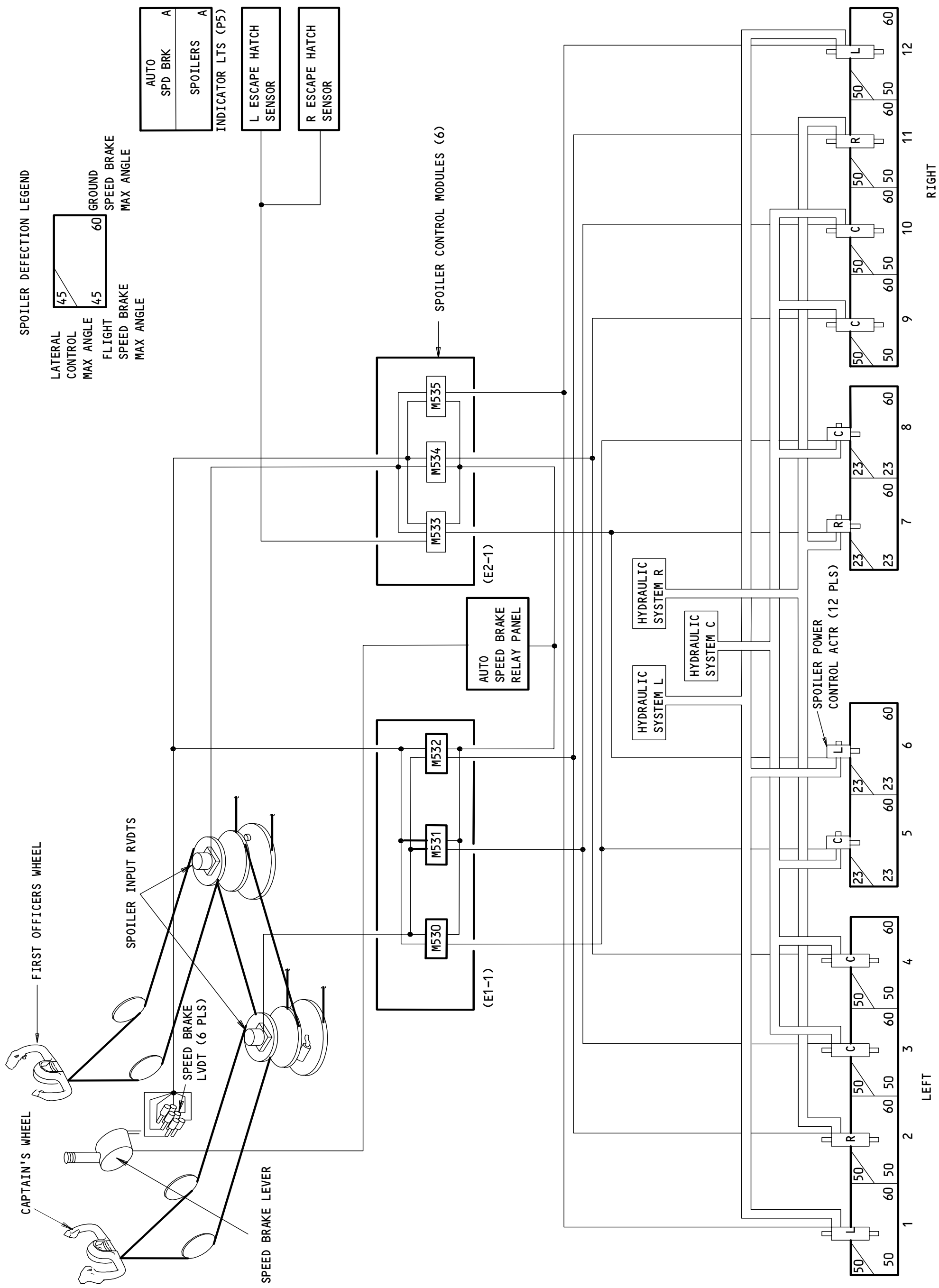
(a) Spoilers 1-4 and 9-12 pick up at 6.3 degrees of control wheel rotation and extend to a maximum of 50 degrees at 63 degrees of control wheel rotation.

(b) Spoilers 5 and 8 pick up at 12.3 degrees control wheel rotation and extend to a maximum of 23 degrees at 63 degrees control wheel rotation.

(c) Spoilers 6 and 7 pick up at 30 degrees control wheel rotation and extend to a maximum of 23 degrees at 63 degrees control wheel rotation.

(d) The speed brake lever moves six LVDTs, which provide signals to the SCMs. When the airplane is in flight, the maximum angle that the speed brakes can be deployed is 23 or 50 degrees. When the speedbrake lever is positioned to FLIGHT SPEEDBRAKE, the spoilers raise to the in-flight maximum and then are lowered in response to control wheel inputs.

(3) When the airplane lands, the automatic speedbrake system deploys all speedbrake panels to the ground angle of 61 degrees. Panels 6 and 7 retract when fuselage escape hatches are opened.



Spoiler Control System
Figure 22

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X. Elevator Control System (Fig. 23)

(1) The elevator control system is operated either manually or by the autopilot. Manual pitch control is obtained with separate cable systems. Control columns on a torque tube move tension regulators, cable, aft quadrants, linkages, PCAs, and elevators. A slave cable loop between elevator PCAs provides a backup interconnect. Manual control is unassisted mechanical movement of the PCAs, which then move the elevators. A stick shaker on each control column is used for stall warning. The aft quadrants are aft of the stabilizer. The PCAs are on the stabilizer aft spar.

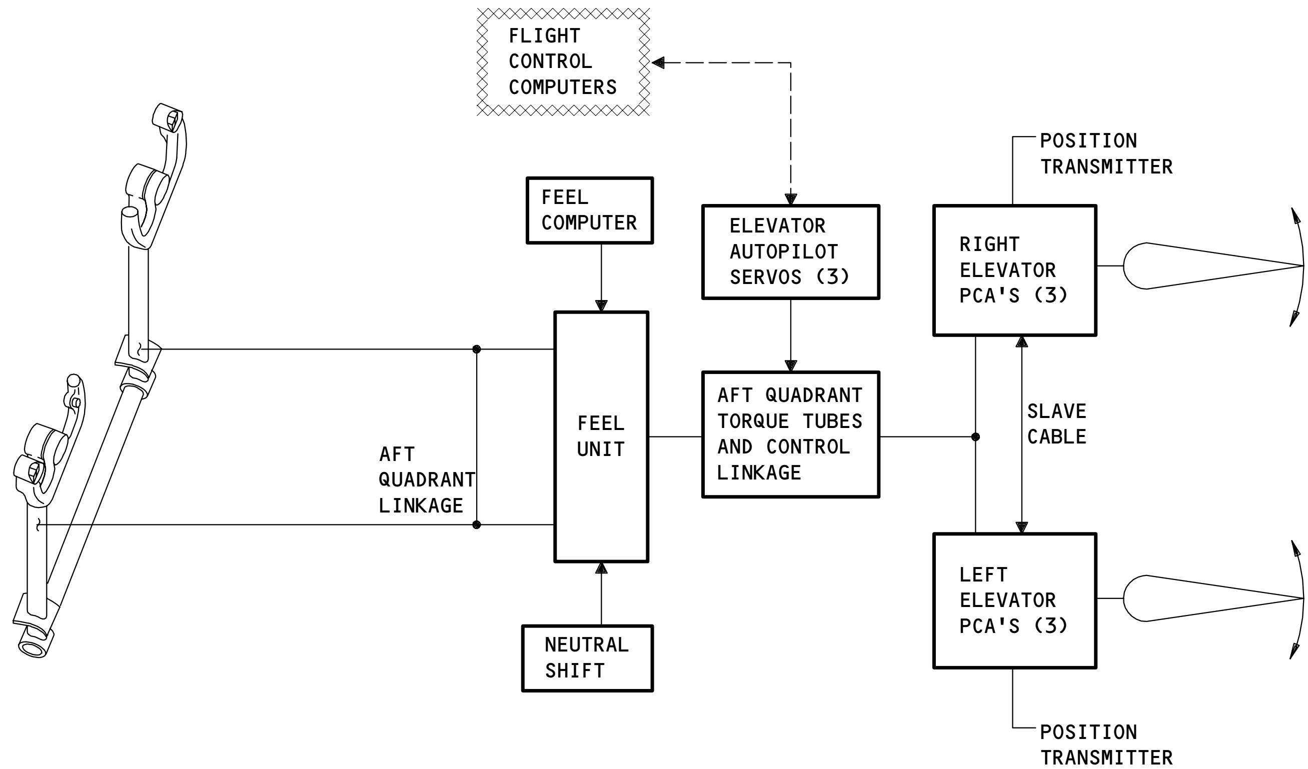
(2) Control inputs are conditioned to provide a sense of feel and centering. Feel is introduced by a feel actuator and feel unit. Inputs are from the neutral shift and a feel computer. The feel computer uses dynamic pressure (CAS) and stabilizer position to vary the feel. The feel unit includes the centering device. Faults are indicated with an ELEV FEEL message on the upper EICAS display unit (Ref 31-41-00).

(3) Autopilot input is with three Flight Control Computers (FCCs) controlling three independent pitch autopilot servos. Each servo independently drives the aft quadrant torque tubes. The servos contain LVDTs which provide servo and surface position feedback to the FCCs.

(4) Elevator control movements are finally routed to both left and right power control actuators by mechanical linkage. There is an interconnection between elevator PCA linkages by slave cable to prevent major asymmetry. Position transmitters are located at each elevator to provide control surface position the lower EICAS display unit status page.

Y. Elevator Control Modes (Fig. 24)

(1) All elevator control is accomplished by movement of a common set of linkages. Both aft quadrant torque tubes operate together. The mechanical linkage translates torque tube movement to PCA control lever movement. The hydraulically powered PCAs move the elevators. Linkage between the PCA output shaft and input linkage centers the control lever when the elevator position corresponds with the input linkage position.



Elevator Control System
Figure 23

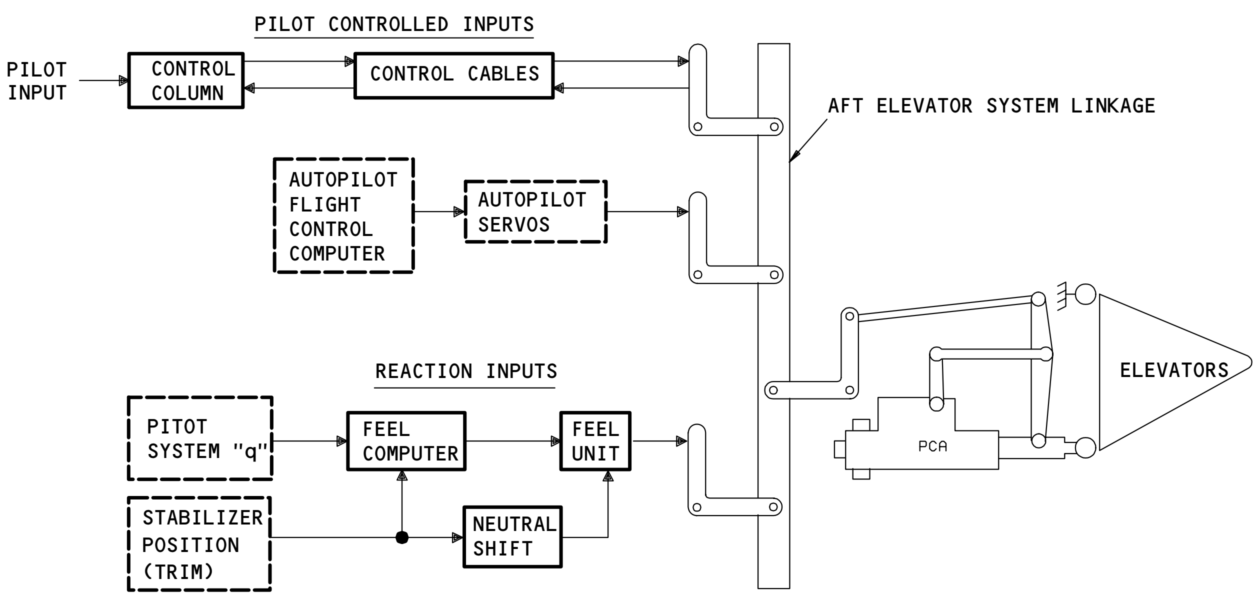
(2) Pitch control is accomplished by pilot controlled inputs through the control columns and autopilot. Manual input, through the control columns, mechanically moves the PCAs.

(3) Pilot controlled inputs are conditioned by reaction inputs. The feel computer reacts to stabilizer position and dynamic pressure, and then provides appropriate inputs to the feel unit. Neutral shift reacts to stabilizer position and maintains the relative stabilizer-to-elevator position by mechanically moving the feel unit.

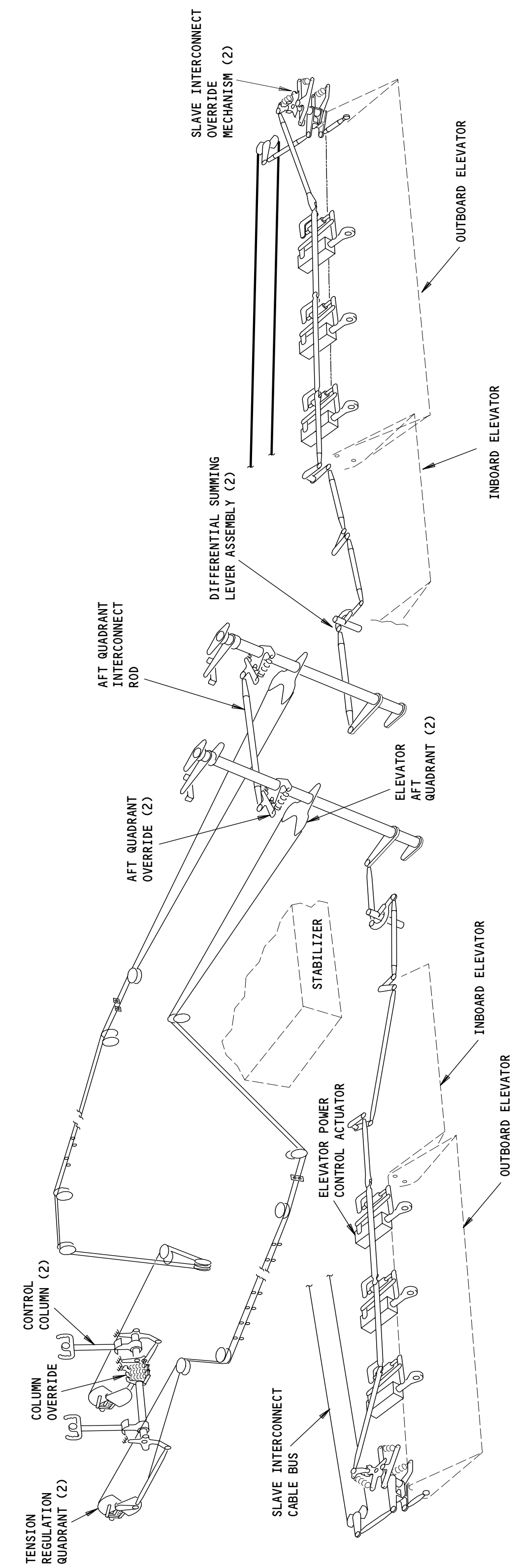
Z. Elevator Control System Details (Fig. 25)

(1) Manual pitch control is accomplished by direct mechanical connection to the six PCAs. Three PCAs are on each elevator. Cable systems are widely separated throughout the fuselage to prevent cross-connections. The aft quadrant interconnect includes override devices to allow control in the event one side should jam. Left and right elevators are interconnected by slave cables through override devices. There is no direct mechanical connections between the two elevators.

(2) The outboard segment of each elevator is rigged approximately 3 degrees down relative to the inboard segment. This reduces PCA size and weight requirements and increases stability through stabilizer aeroelasticity. The two segments are rigidly attached to each other.



Elevator Control Modes
Figure 24



Elevator Control System Details
Figure 25

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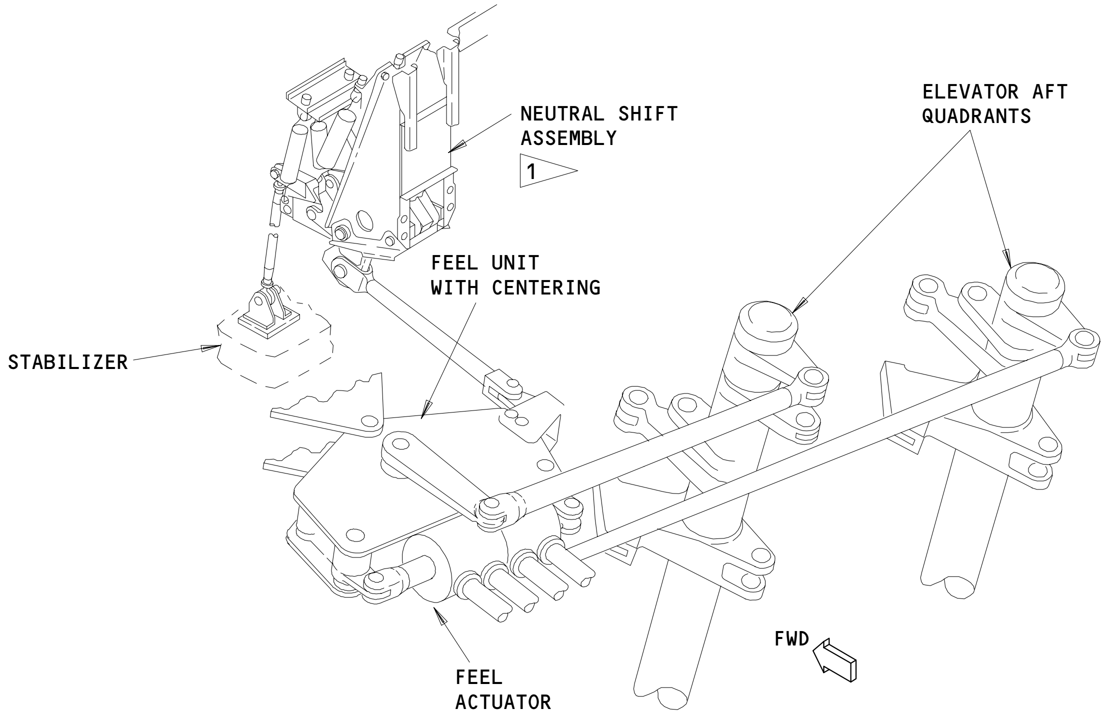
(3) The PCAs are attached to the outboard segment of each elevator. Each of the three PCAs are powered by a different hydraulic system. Manual control inputs do not connect directly to the elevators but control only the PCAs, which then drive the elevators. No elevator trim is provided. Pitch trim is accomplished only with the stabilizer.

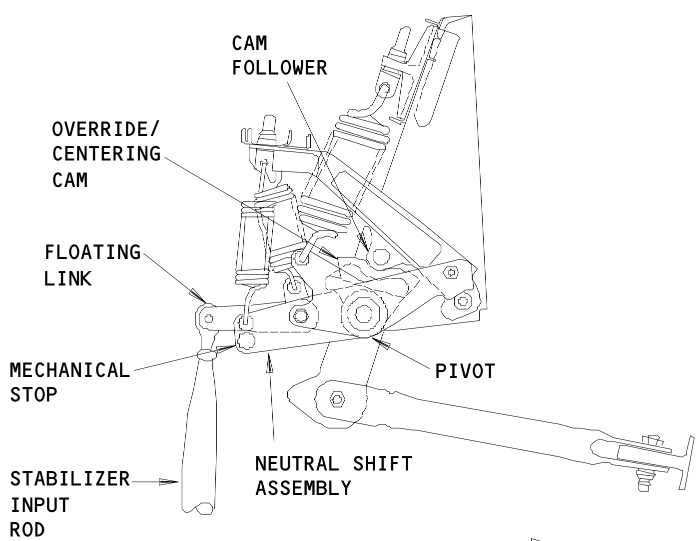
AA. Elevator Control Conditioning Devices (Fig. 27)

(1) ON 767-200 AIRPLANES;
Elevator Neutral Shift

(a) The elevator neutral shift assembly recenters the elevators in proportion to stabilizer position to maintain elevator authority. The effective range is between 8 and 12 1/2 degrees of stabilizer leading edge down (airplane nose-up). The assembly moves the feel and centering unit through an override cam and follower. The feel and centering then moves linkages to the PCAs, which reposition the elevators. Elevators move from a normal position of 3 degrees down to a maximum of 1 degree up.

Not Used
Figure 26





NEUTRAL SHIFT MECHANISM 1

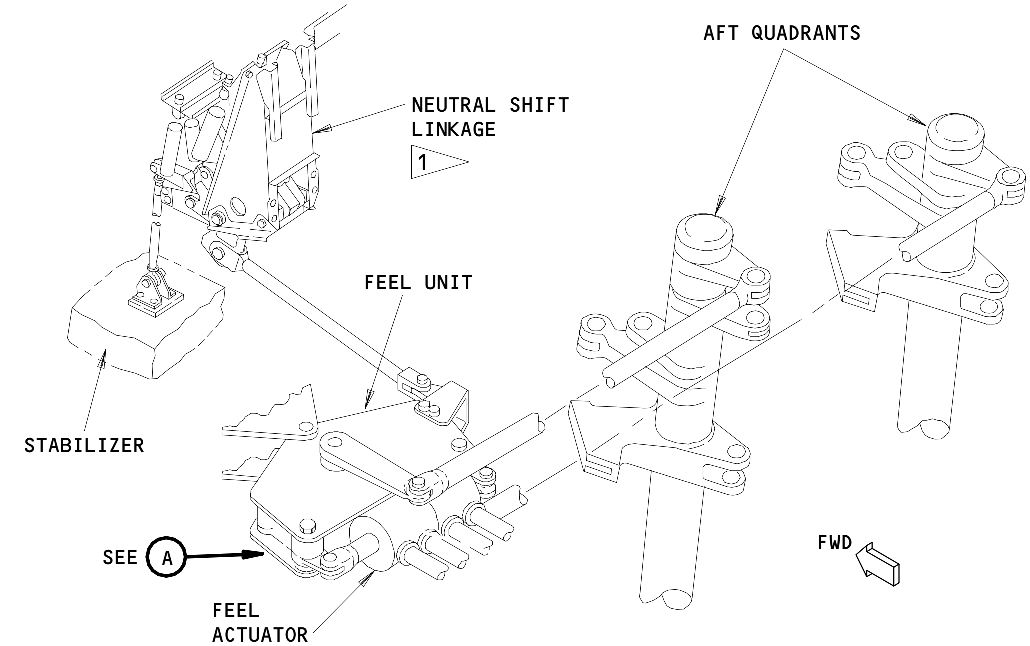
1 767-200 AIRPLANES

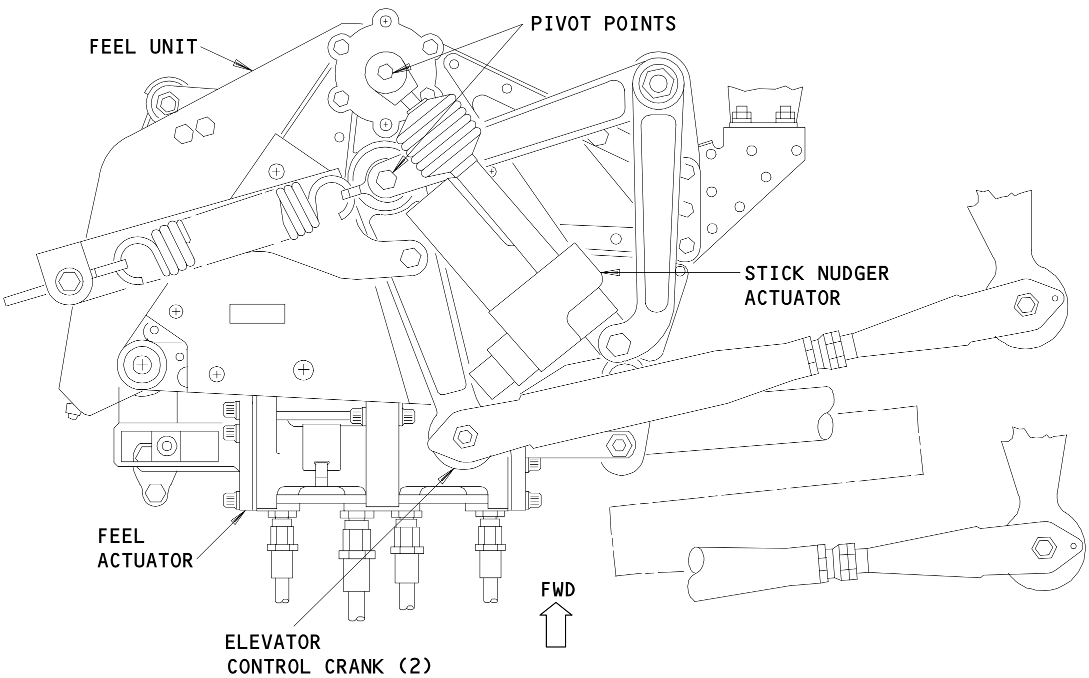
Elevator Control Conditioning Devices
Figure 27

(2) Elevator Feel and Centering Unit
(a) Mechanical feel and centering is provided by a cam and follower inside the feel unit. They are between a feel actuator and the feel unit control arms. Control arms are connected to the elevator aft quadrant torque tubes. The feel computer varies hydraulic pressure to the feel actuator in proportion to airspeed and stabilizer position. This provides the appropriate pressures for the sense of feel and centering required by the pilots.

AB. Elevator Feel Unit (Fig. 28)
(1) The elevator feel unit generates increasing force with increased control system movement. The scheduling of force application is determined by the feel computer, which uses dynamic pressure and stabilizer position to meter hydraulic pressure to the feel actuator.
(2) The feel actuator resists motion of both left and right elevator control cranks on the feel unit. Resistance varies with hydraulic pressure from the feel computer. Crank motion in either direction pulls the outer levers together, which extends the feel actuator piston rods.
(3) Mechanical feel is then accomplished with a cam and spring-loaded cam follower on the first officer's control crank working against the feel actuator. A cam detent also creates a centering force to keep elevators in the neutral (faired) position.
(4) Total elevator feel force is the combination of mechanical and hydraulic force from the feel unit, plus system friction. Feel force returns the controls to neutral and resists their displacement from neutral.

AC. Elevator Control Actuators And Linkage (Fig. 29)
(1) Manual input is through control cables to the aft quadrants and torque tubes. Torque tubes pivot about a vertical line through the top and bottom lever arms. Push rod linkages interconnect the two torque tubes through override devices. Ground springs help return linkages to neutral.
(2) The feel unit inputs to each torque tube with separate rods at the top. The neutral shift linkage is input to the top of the feel unit. A pushrod from the stabilizer drives the neutral shift assembly. The feel computer has a pushrod input from the forward end of the stabilizer center section.





ELEVATOR FEEL UNIT
A

1 767-200 AIRPLANES

Elevator Feel Unit
Figure 28

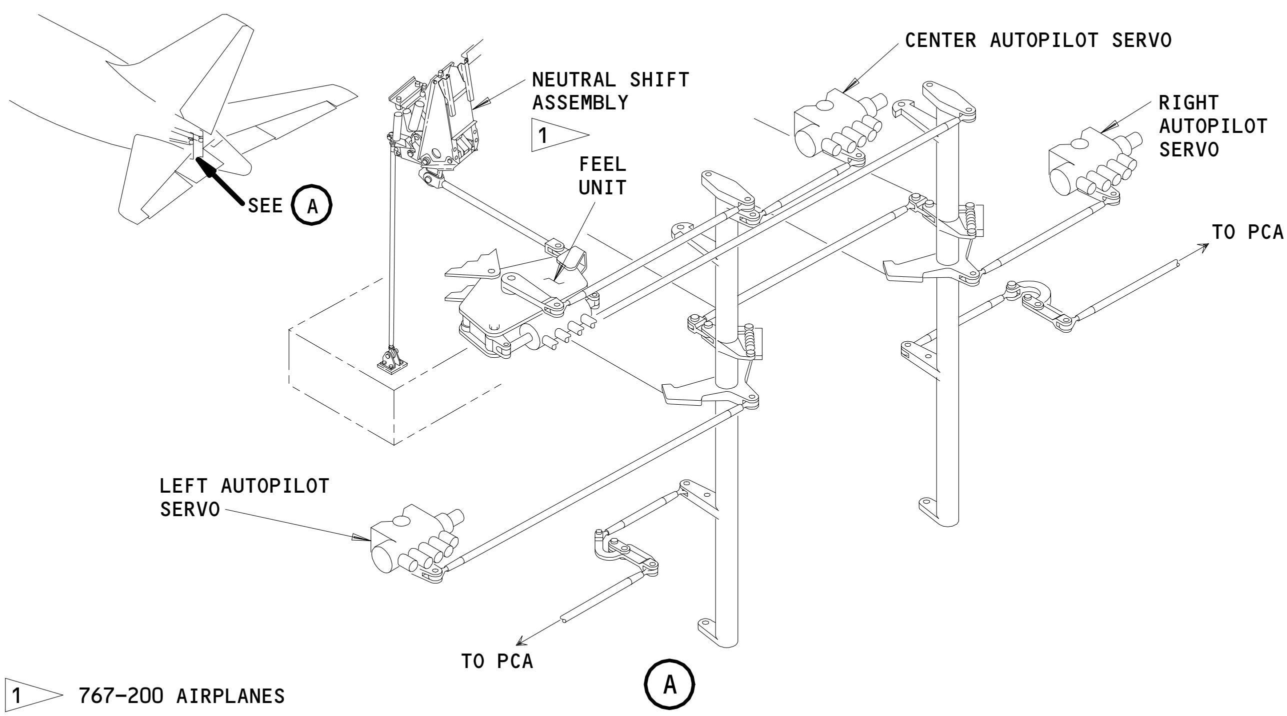
(3) Three elevator autopilot servos input directly to the torque tubes. The left and center autopilot servos drive the left torque tube. The right autopilot servo drives the right torque tube.

AD. Stabilizer Trim Mechanical Components (Fig. 30)

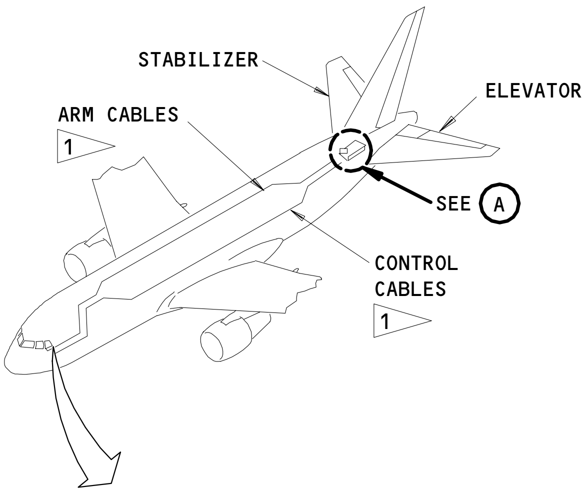
(1) The stabilizer consists of two outboard control surfaces joined together with a center section to form one assembly. The stabilizer pivots about a point at the aft end of the center section. The leading edge of the center section is moved by a ballscrew actuator assembly. The actuator is controlled by two stabilizer trim control modules.

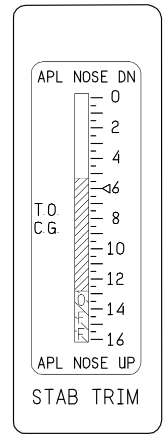
(2) ON SAS 150-153;
the stabilizer trim control modules respond to either mechanical or electrical inputs. Manual stabilizer trim levers drive cables which move ARM and CONTROL hydraulic valves on both stabilizer trim control modules. The left cables provide ARM inputs. The right cables provide CONTROL inputs. Electrical inputs from the SAM control solenoid operate ARM and CONTROL valves on each module.

(3) ON ALL MTH AIRPLANES; SAS 050-149, 154-999;
the stabilizer trim control modules respond to electrical inputs from the alternate stabilizer trim switches and the SAM control solenoid which operate ARM and CONTROL valves on each module.

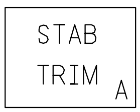


Elevator Control Conditioning Interrelationship
Figure 29

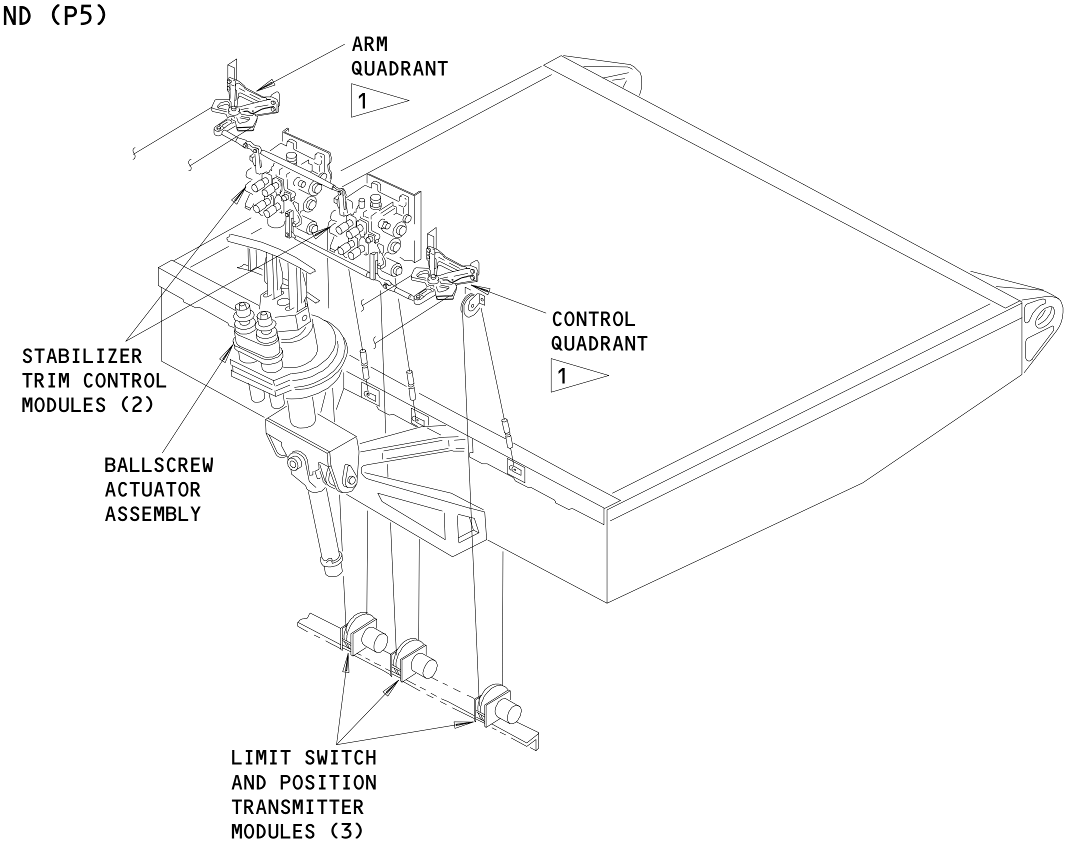




STABILIZER TRIM POSITION INDICATOR (P10) (TYPICAL)



FAULT IND (P5)



STABILIZER CENTER SECTION

A

1 SAS 150-153

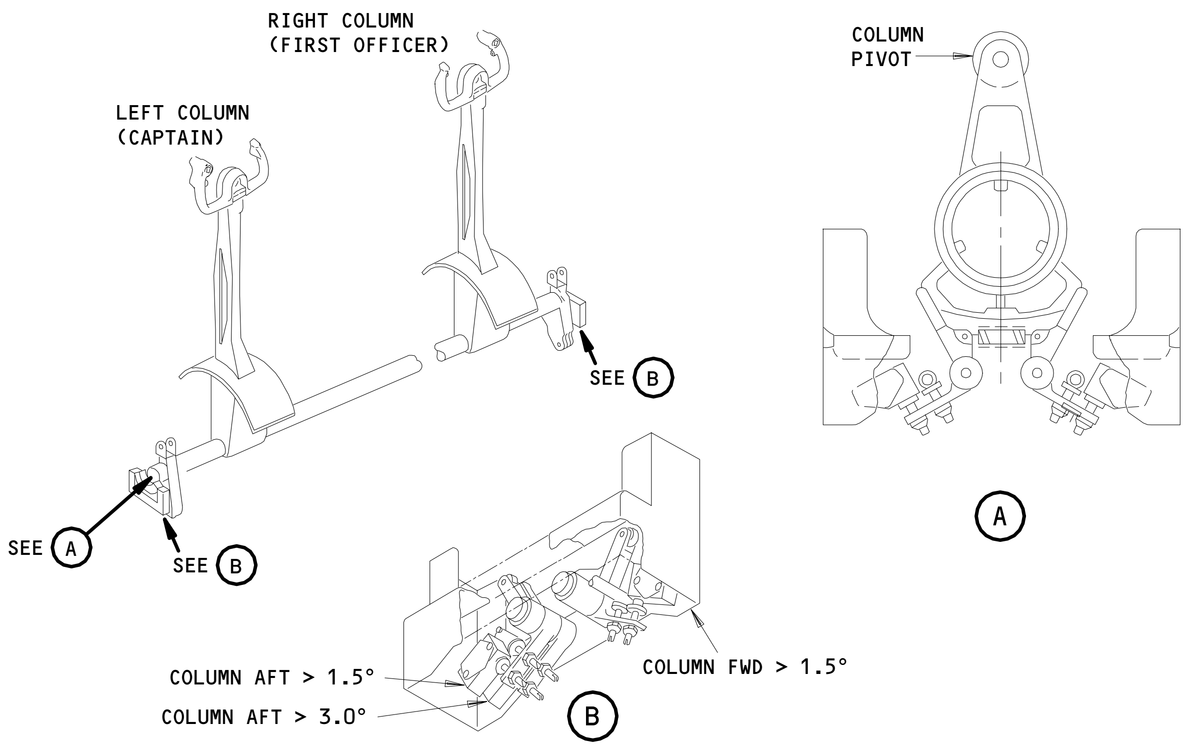
Stabilizer Trim Mechanical Components
Figure 30

(4) Three limit switch and position transmitter modules are mounted on fixed structure below the center section leading edge. Switches and transmitters in the modules are positioned by control cables driven by the stabilizer. Each module includes a Rotary Linear Variable Transformer (RVDT), a synchro, and limit switches. The RVDT provides stabilizer position inputs to the Flap and Stabilizer Position Module (FSPM). The synchro provides position signals for the stabilizer trim position indicator on the control stand.

(5) Stabilizer trim faults are indicated by an amber STAB TRIM light on the P5 panel.

AE. Column Cutout Switches (Fig. 31)

(1) Electric trim operation is inhibited if the control column is moved in opposition to stabilizer trim. Column cutout switches on the forward quadrant interrupt control signals to the control modules. Column movement forward greater than 1.5 degrees interrupts nose-up trim signals. Column movement aft greater than 1.5 degrees interrupts nose-down trim signals. If the neutral shift program is in effect, the column must be moved aft greater than 3 degrees.



Column Cutout Switches
Figure 31

AF. Stabilizer Trim Control and Operation (Fig. 32)

(1) Stab Trim Control

(a) Three priorities of trim are available.

1) ON SAS 150-153;
first priority trim control is manual via the trim levers and cables. The manual ARM input to the Stabilizer Trim Control Modules (STCM) inhibits all electrical inputs (Ref 27-41-00, Horizontal Stabilizer Trim Control System).

2) ON ALL MTH AIRPLANES; SAS 050-149, 154-999;
first priority trim control is electric via the alternate stabilizer trim switches. The electrical ARM input to the Stabilizer Trim Control Modules (STCM) inhibits all other inputs (Ref 27-41-00, Horizontal Stabilizer Trim Control Systems).

3) Second priority is pilots' electric trim (using the thumb switches) if only one autopilot FCC is engaged. Thumb switch electric trim is third priority if more than one FCC is engaged.

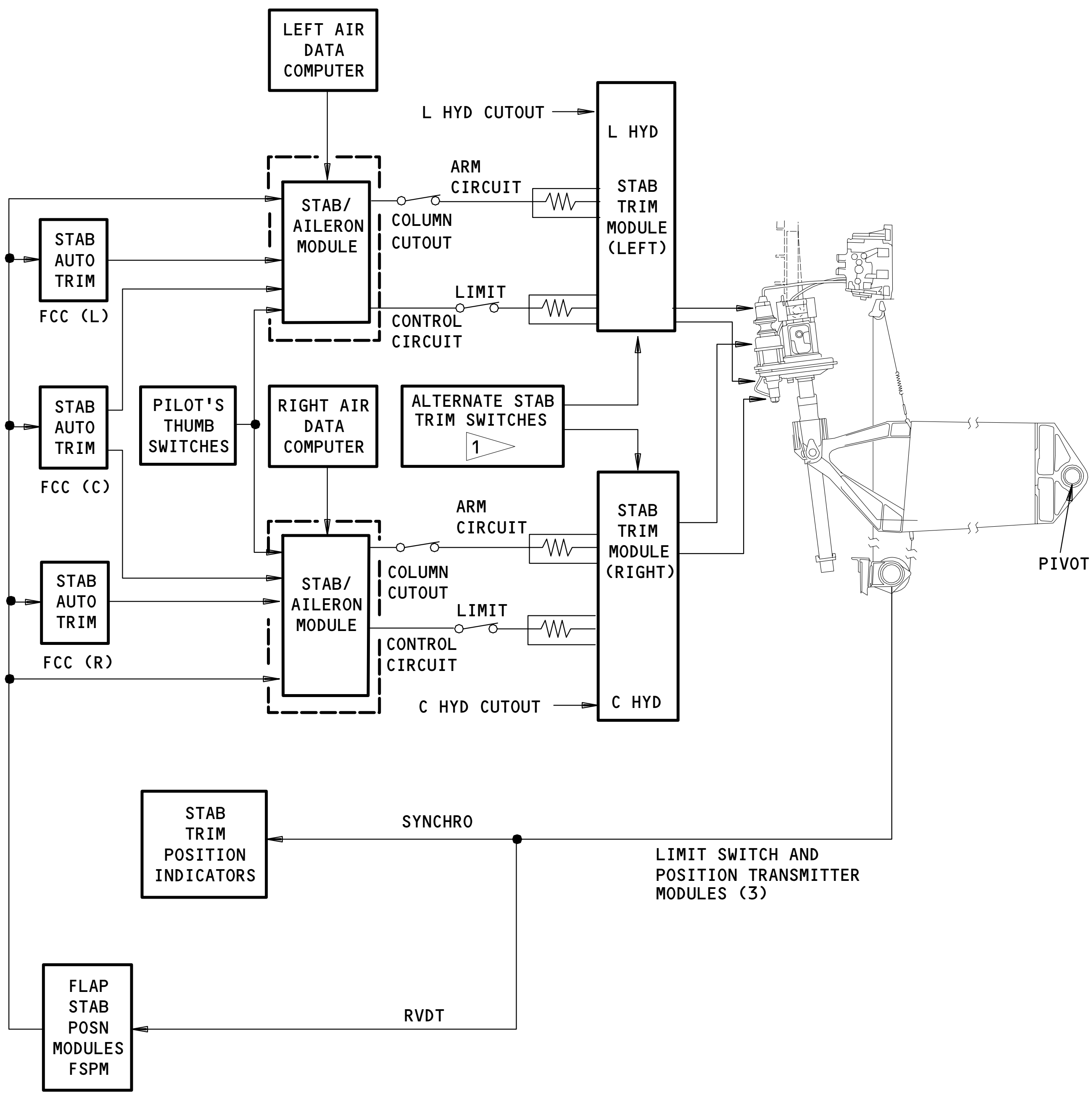
4) Third priority is autopilot automatic stabilizer trim control if one FCC is engaged. Automatic stabilizer trim is second priority if more than one FCC is engaged (Ref 22-22-00, Automatic Stabilizer Trim).

(b) The two hydraulic stab trim CUTOUT switches on the control stand operate shutoff valves on the stabilizer trim control modules.

(2) Stab Trim Operation

(a) ON SAS 150-153;
the STCMs are operated from two inputs, mechanical and electrical. Mechanical input is with the manual trim levers, which move control arms on both STCMs. All electrical ARM and CONTROL input signals come from the stabilizer trim and aileron lockout modules (SAM). These signals are interrupted by column cutout and limit switches between each SAM and STCM.

(b) ON ALL MTH AIRPLANES; SAS 050-149, 154-999;
the STCMs are operated from two electrical inputs. The alternate stabilizer trim switches and the stabilizer trim and aileron lockout modules (SAM) provide electrical ARM and CONTROL input signals. The signals from the SAM are interrupted by column cutout and limit switches between each SAM and STCM.



1 ON SAS 150-153, MANUAL TRIM LEVERS

Stabilizer Trim Control and Operation
Figure 32

(c) The SAM output signals are initiated by operating manual electric trim or automatic stabilizer trim. Inputs which initiate SAM outputs are:
   1) Pilots' thumb operated switches - analog discrete 28 Vdc for manual electric trim.
   2) FCC - Analog discrete ARM and digital CONTROL signals for automatic stabilizer trim.

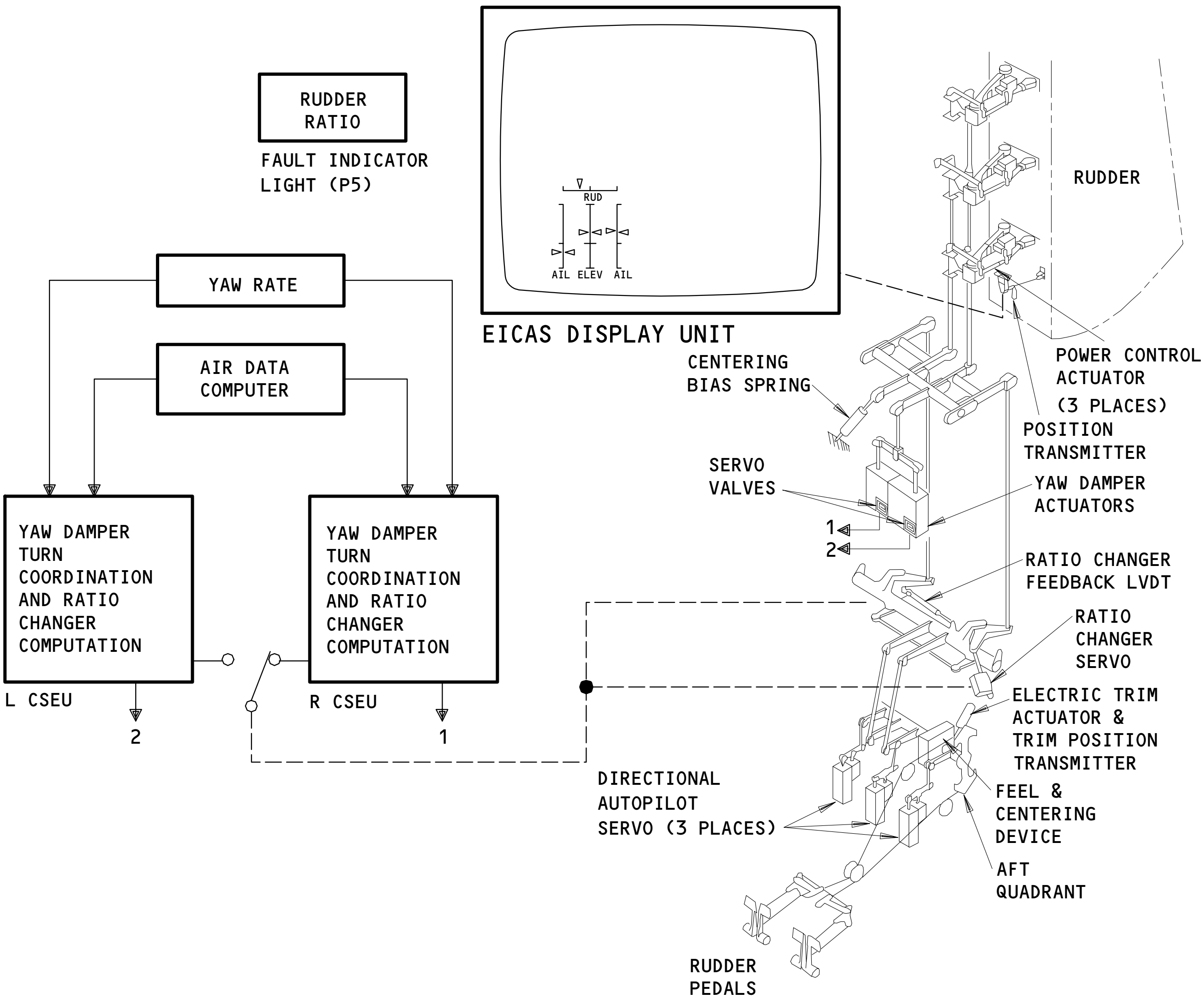
(d) The STCMs provide the hydraulic pressure to the hydraulic motors on the ballscrew actuator assembly. The maximum rate of stabilizer movement is 0.5 degrees per second.
   1) ON SAS 150-153;
      both motors operate with manual lever and thumb switch inputs.
   2) ON ALL MTH AIRPLANES; SAS 050-149, 154-999;
      both motors operate with alternate switch and thumb switch inputs.

(e) The automatic stabilizer trim control uses only one hydraulic motor. Only one SAM is in command at any time as determined by the FCC engaged. The rate of stabilizer movement is 1/2 of that used for manual or manual electric control. Maximum rate is 0.25 degrees per second.

(f) A hydraulic pressure signal from the elevator feel computer modulates the flow rate of fluid to the hydraulic motors via a rate valve in the STCM. This signal causes the stabilizer trim rate to vary as a function of airspeed. The trim rate produced by one STCM is between 0.05 and 0.25 degrees per second. Two STCMs produce a trim rate of 0.1 to 0.5 degrees per second.

AG. Rudder Control System (Fig. 33)

(1) Rudder system control components are primarily located in the vertical stabilizer. They consist of the following items:
   (a) Power Control Actuators (PCA) (3)
   (b) Position Transmitter
   (c) Yaw Damper Servo (2) - Actuator
   (d) Rudder Ratio Changer - Servo and Assembly
   (e) Electric Trim Actuator and Trim Position Transmitter



Rudder Control System
Figure 33

(f) Feel and Centering Device
(g) Aft Quadrant
(h) Directional Autopilot Servos (3)
(i) Interconnecting Linkages with dual (primary and secondary) output rods.

(2) Manual control is accomplished by a cable system from the pilots' rudder pedals to the aft quadrant. Autopilot rudder control is only used for directional guidance during landing rollout. Autopilot control signals from the FCCs operate the directional autopilot servos, which provide an input to the aft quadrant.

(3) Automatic stability control is by the CSEU yaw damper modules. Yaw damping and turn coordination is provided through the yaw damper servos.

(4) Control inputs are conditioned by feel and centering, and by the trim and ratio changer. Feel, centering, and trim devices are on the aft quadrant. The ratio changer servo is controlled by the CSEU rudder ratio changer modules. All inputs are translated to move each PCA control valve. The control valve causes the PCA to extend or retract, which moves the rudder. Each PCA is powered by a separate hydraulic system.

(5) Rudder surface position is indicated on the control surface position indicator on the lower EICAS display unit status page (Ref 31-41-00). The rudder position transmitter on the rudder provides an input to the indicator.

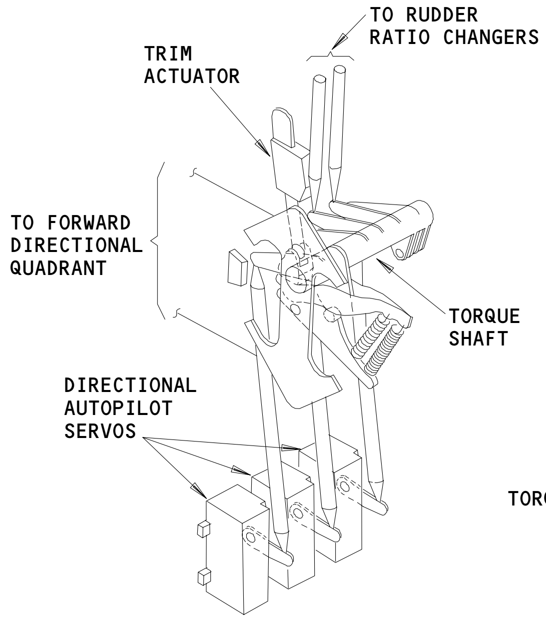
(6) An amber RUDDER RATIO light is on the panel P5. Faults in the rudder ratio changer turn on the light.

(7) A rudder trim control switch is on the aft end of the control stand panel P8. The trim control drives the rudder trim actuator which recenters the rudder.

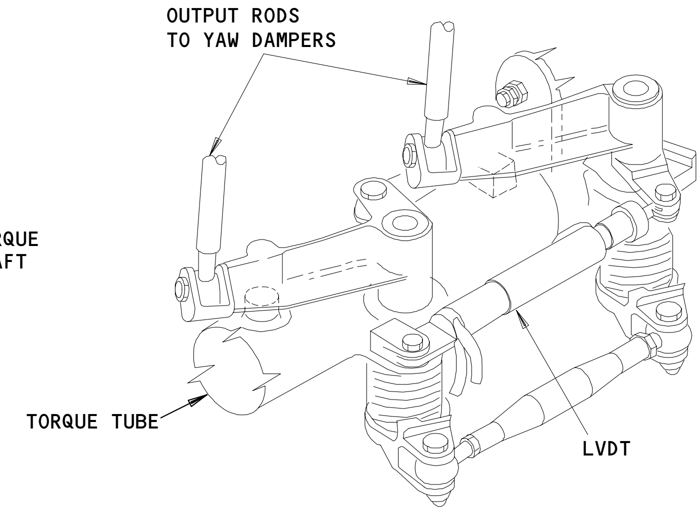
AH. Rudder Control Conditioning Devices (Fig. 34)

(1) Rudder Feel, Centering, and Trim

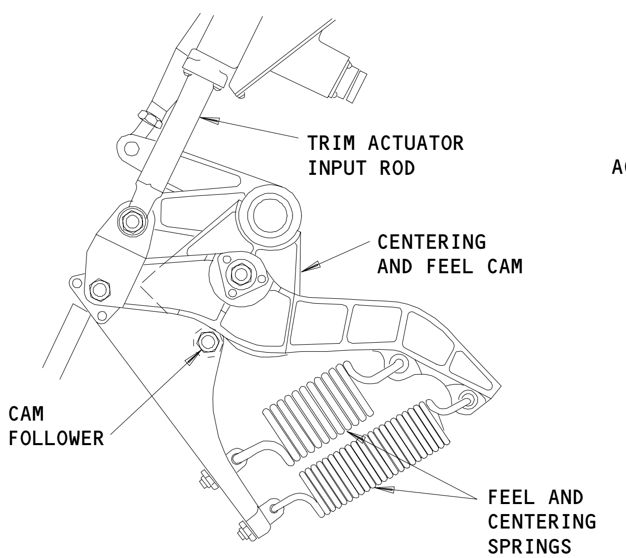
(a) The rudder feel, centering, and trim device is beside the aft quadrant in the vertical stabilizer. Feel and centering is accomplished by a cam and spring loaded cam follower. Feel force increases as the cam follower is forced up the cam's slope by rotation of the aft control quadrant. Centering force is a result of the cam detent being forced to align with the cam follower after input forces to the aft quadrant are removed.



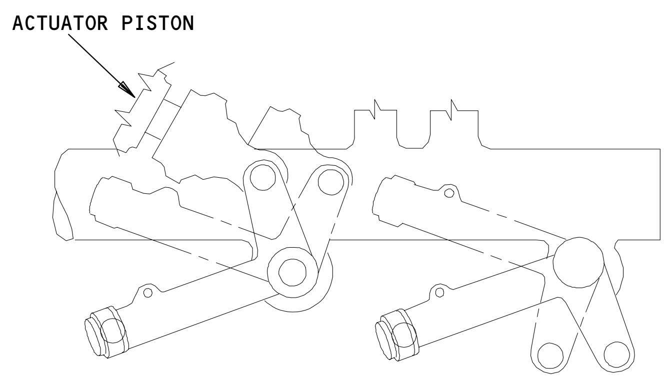
RUDDER AFT QUADRANT ASSEMBLY



RUDDER RATIO CHANGER



RUDDER FEEL, CENTERING AND TRIM ASSEMBLY



RUDDER RATIO CHANGER
TOP VIEW

Rudder Control Conditioning Devices
Figure 34

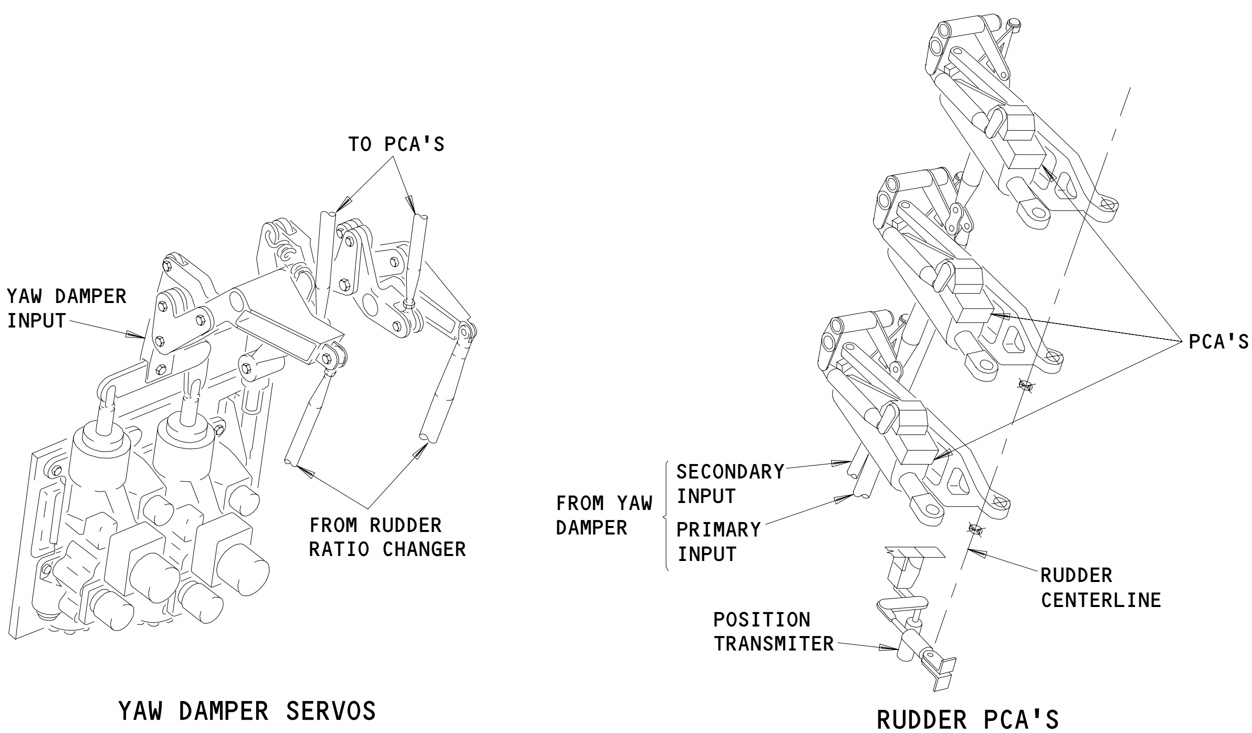
(b) Rudder trimming is accomplished when the trim actuator rotates the feel cam to establish a new center position. As the feel cam detent shifts, the cam follower moves and provides an input to the PCAs, which then repositions the rudder. The rudder trim switch on the control stand provides electrical inputs to the trim actuator.

(2) Rudder Ratio Changer

(a) The rudder ratio changer is above the aft quadrant in the vertical stabilizer. The mechanism changes the output control linkage movement in proportion to aft quadrant and CSEU rudder ratio changer module inputs. Rudder displacement is increased at low airspeeds to maintain rudder authority.

(b) A hydraulic actuator responds to inputs from the CSEU rudder ratio changer modules. The rudder ratio changer modules provide an output in proportion to airspeed. The actuator repositions output lever arms to vary the ratio of control rod input to output. The yaw damper system servos also connect to the rudder ratio changer output rods that drive the rudder PCAs.

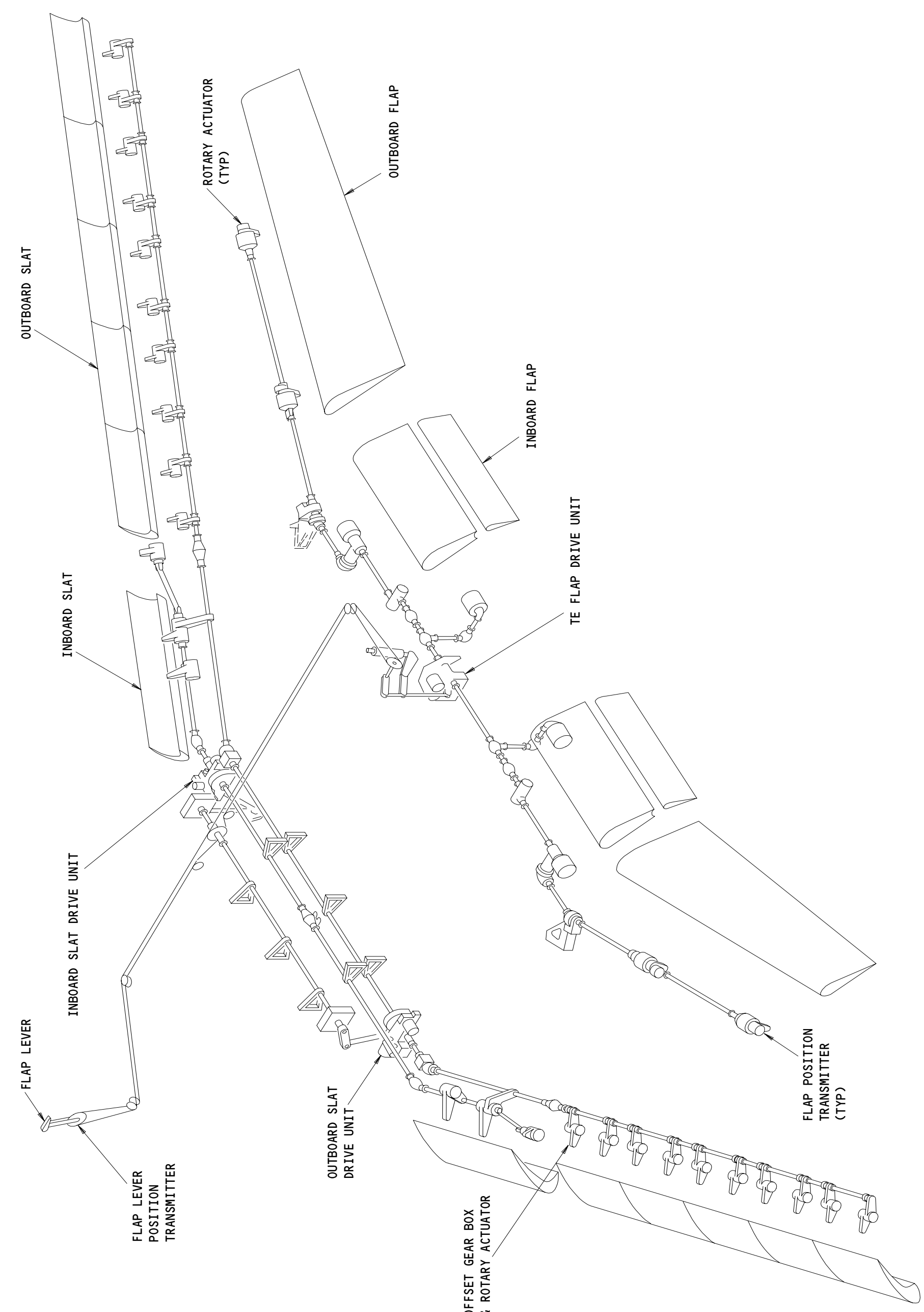
AI. Yaw Damper And Rudder PCAs (Fig. 35)



Yaw Damper and Rudder PCA
Figure 35

(1) Yaw Damper Assembly
   (a) The yaw damper assembly is above the rudder ratio changer assembly. Two yaw damper servos respond to commands from CSEU yaw damper modules to counteract undesired yaw motion. The yaw damper modules provide an output in response to yaw rate and turn coordination (roll) inputs.
   (b) Summing linkages combine the output of the two yaw damper servos with the output of the rudder ratio changer. The final control input to rudder PCAs is with two rods, a primary input rod and a secondary input rod.
(2) Rudder PCAs
   (a) Three PCAs are above the yaw damper assembly. The PCAs translate control lever inputs to rudder surface movement. Each PCA uses a separate hydraulic system.
   (b) All PCA control valves are driven by the PCA control levers, while all are interconnected. The primary input rod from the yaw damper assembly and ratio changer is connected to the center PCA. The secondary input rod is connected to the lower PCA. The PCAs cause movement of the rudder about the hinge line in opposition to the reaction link. The rudder position transmitter provides an output to the control surface position indicator on the lower EICAS display unit status page.

AJ. High Lift System - Introduction (Fig. 36)
(1) Leading Edge Slats
   (a) Six leading edge slat segments are on each wing. Control is by movement of the flap lever on the center section of the control stand (P10). Control cables from the flap lever operate the slat and flap Power Drive Units (PDUs) which use hydraulic or electrical power. Primary control is hydraulic.
   (b) The flap lever also enables the Flap/Slat Electronic Unit (FSEU), which controls the hydraulic shutoff valve for each PDU. The PDUs drive a gearbox and rotary actuator at each slat segment. Slat position is transmitted to the FSEU for fault detection and annunciation.
   (c) Alternate control is via the alternate flaps control panel on the pilots center instrument panel (P3-1). This panel controls the FSEU which provides electrical output signals that cause the PDUs to extend or retract slats (or flaps) to the selected position.



High Lift System - Introduction
Figure 36

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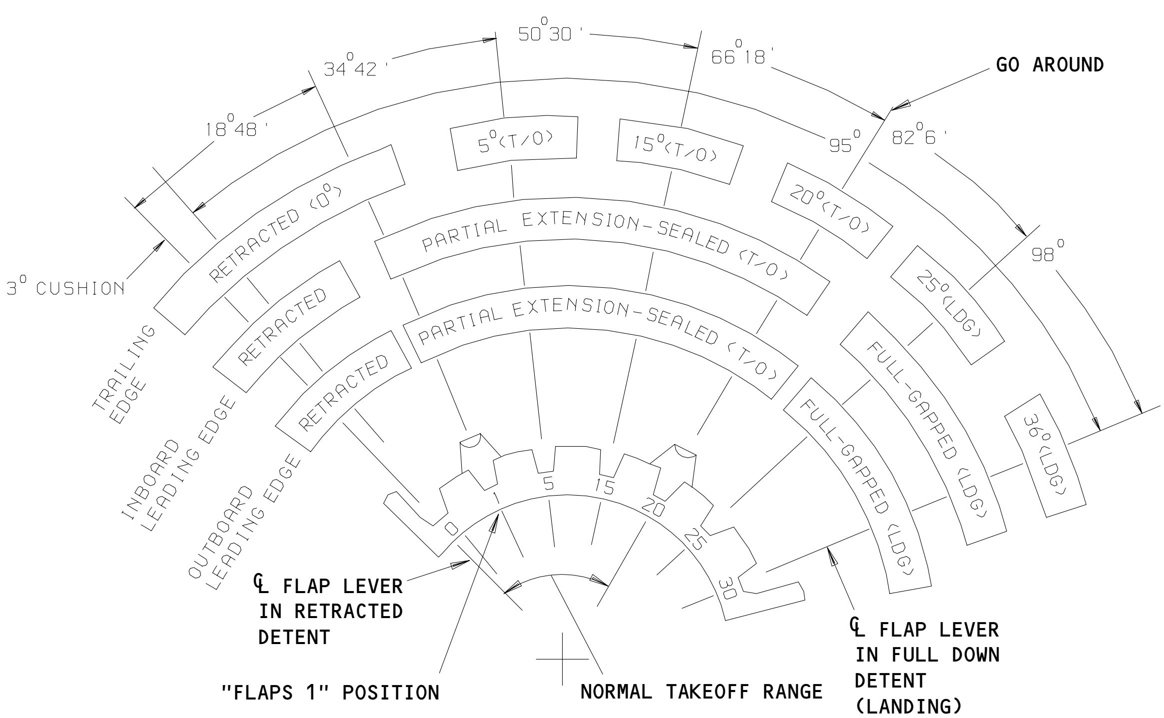
(2) Trailing Edge Flaps
  (a) Two trailing edge flap segments are on each wing. The outboard flap is composed of two flap surfaces.
  (b) The flap lever controls flap PDUs and the FSEUs as previously described. Hydraulic power for the flap PDUs is enabled by the FSEU. Flap position is sent to FSEU for fault detection and annunciation. It is also sent to the FSEU for interfacing with other autoflight systems.
  (c) Alternate control is via the alternate flaps control panel, as previously described.
  (d) The primary drive is powered from the center hydraulic system. The alternate drive uses 115 Vac.

AK. Flap Lever/Flap-Slats Positions (Fig. 37)
  (1) The flap lever can be positioned at between 0 and 30 units. The units do not necessarily relate to degrees of flap extension. Detents are at 0, 1, 5, 15, 20, 25 and 30 units. Trailing edge flap and inboard and outboard leading edge slat relative positions are indicated.
    (a) Position 0 fully retracts the flaps and slats.
    (b) Position 1 results in partial slat extension with the slats still sealed (no gap). The flaps remain fully retracted.
    (c) Position 5 results in the flaps extending to 5 degrees after the slats are partially extended (no gap).
    (d) Position 15 and 20 extend the flaps to 15 and 20 degrees with the slats still partially extended (no gap).



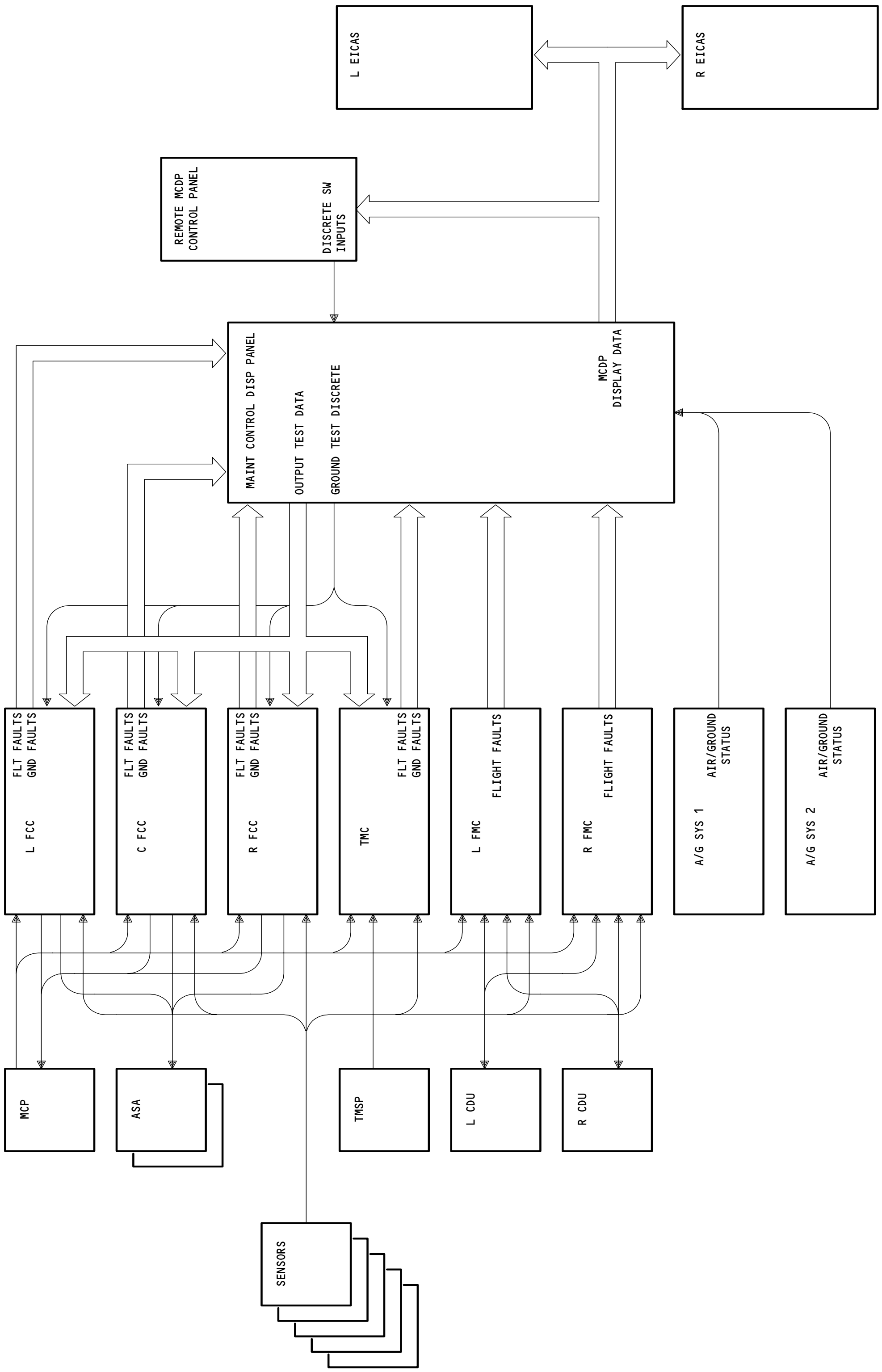
Flap Lever/Flap Slat Positions
Figure 37

(e) Position 25 and 30 extends the flaps to 25 and 36 degrees, and fully extends (full-gapped) the slats.

(f) Detent positions 5, 15, and 20 are normally used for takeoff. Detent positions 25 and 30 are normally used for landing.

AUTOFLIGHT BITE - DESCRIPTION AND OPERATION

1. General
   A. Autoflight BITE provides an automated, rapid means of testing and verifying the operational status of line replaceable units associated with autoflight. BITE testing can be performed with a minimum of effort using only equipment installed on the airplane. Functions contributing to BITE include:
      (1) Built-In-Test-Equipment in individual LRUs.
      (2) Built-In-Tests in individual LRU software.
      (3) Monitor and reporting functions in individual LRUs.
      (4) Control and display functions provided by the Maintenance Control Display Panel (MCDP).
   B. Autoflight BITE includes monitoring and reporting in-flight faults and conducting ground tests.
      (1) Flight fault data is available in two categories.
         (a) Last flight faults contains all fault data detected during the last flight leg, and represents current fault data.
         (b) Previous flight faults contains historical fault data from previous flights.
      (2) Ground tests provide a means of exercising and testing hardware on the ground. Three general categories of tests are available.
         (a) LRU tests provide testing of specific LRUs in the autoflight systems. There are 15 LRU tests.
         (b) System tests exercise the autoflight systems, including LRUs, interfaces, and supporting systems. There are 2 system tests.
         (c) Support tests exercise and monitor supporting functions and components. There are 12 support tests.
   C. BITE messages are provided in four categories.
      (1) Fault messages identifying specific problems.
      (2) Diagnostic codes providing amplifying data on selected faults.
      (3) Status information related to selected functions or components.
      (4) Instruction/questions related to BITE operation and interactive testing requiring operator actions.
2. Component Details
   A. Autoflight BITE is supported by a hardware network and series of software controlled tests. Hardware components and interfaces are shown in Fig. 1.
      (1) Most autoflight BITE functions are associated with the Maintenance Control Display Panel (MCDP). Three major systems interface with the MCDP, providing fault data for system components and interfacing LRUs. These systems are:
         - Autopilot/Flight Director System
         - Thrust Mangement System
         - Flight Management Computer System
      (2) The Yaw Damper System and associated LRUs provide a BITE independent of the MCDP.



Autoflight Bite Components
Figure 1

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B. Autopilot/Flight Director System (Fig. 2)

(1) The Autopilot/Flight Director System consists of three Flight Control Computers (FCCs) cross channel data buses, one Mode Control Panel (MCP), and two Autoland Status Annunciators (ASAs).

(2) The FCCs receive and report fault data from the following LRUs.

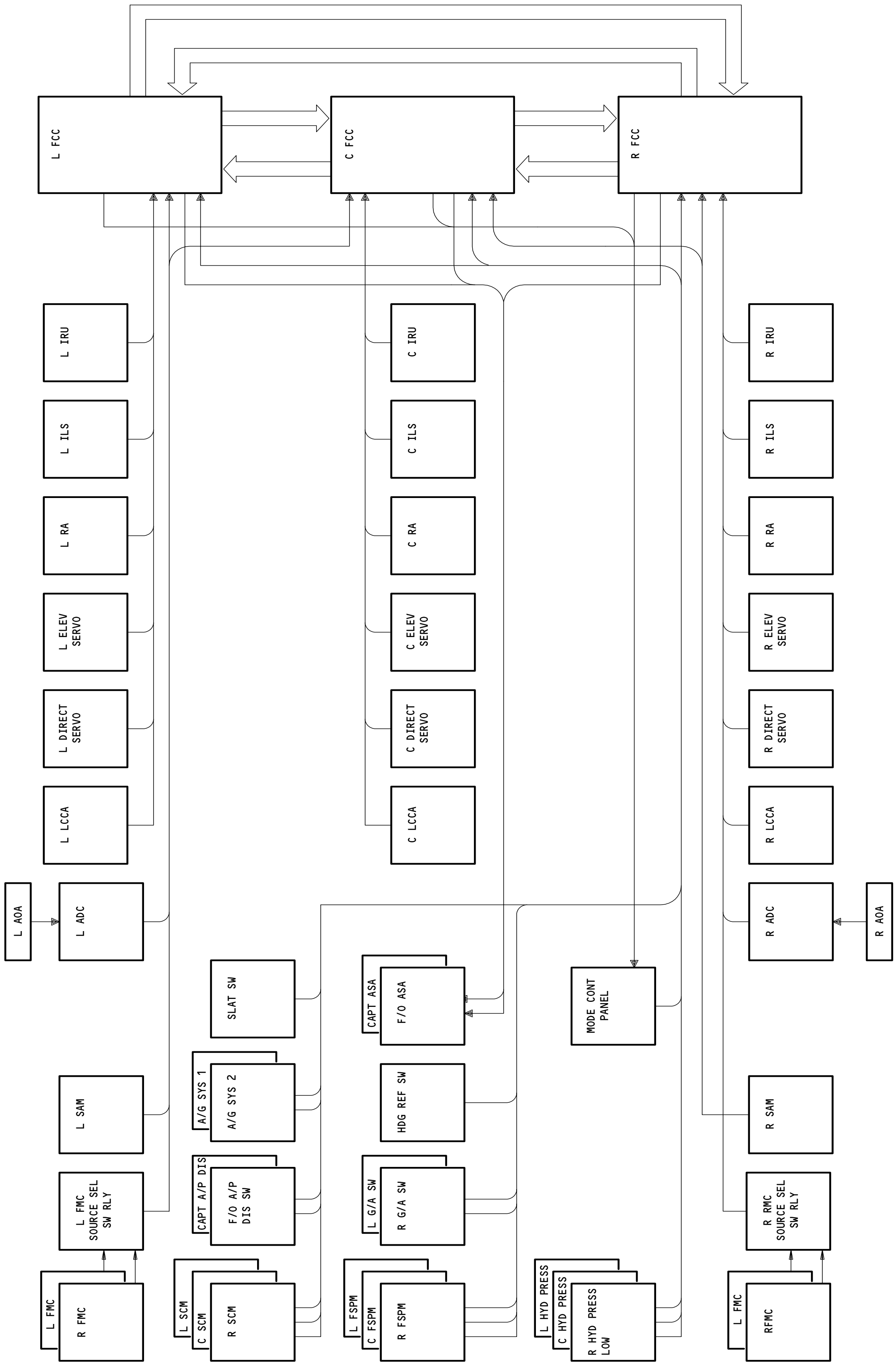
(a) Inertial Reference Unit (IRU) L,C,R
(b) Instrument Landing System (ILS) Receiver L,C,R
(c) RA L,C,R
(d) Elevator Autopilot Servo (EAS) L,C,R
(e) Directional Autopilot Servo (DAS) L,C,R
(f) Lateral Central Control Actuator (LCCA) L,C,R
(g) Air Data Computer (ADC) L,R.
(h) Flight Management Computer (FMC) L,R
(i) Stabilizer Trim/Aileron Lockout Module (SAM) L,R
(j) Low Hydraulic Pressure Light L,C,R
(k) Flap/Stabilizer Position Module (FSPM) L,C,R
(l) Go-around switch L,R
(m) Heading reference (MAG/TRUE) switch
(n) Spoiler Control Module (SCM) L,C,R
(o) F/O and Capt. A/P Disconnect Switch
(p) Slat switch

C. Thrust Management System (Fig. 3)

(1) The Thrust Management System consists of one TMC, one MCP, and one Thrust Mode Select Panel (TMSP).

(2) The TMC receives and reports fault data from the following LRUs.

(a) Electronic Engine Control (EEC) L,R
(b) Inertial Reference Unit (IRU) L,R
(c) Flight Management Computer (FMC) L,R
(d) L FSPM
(e) A/T Servomotor Generator
(f) A/T Disconnect SW
(g) Go-around Switch
(h) Air/Ground System 1
(i) Air Data Computers (ADC) L,R with associated probes
(j) Engine EEC Discretes cards L,R with associated valves, switches, etc.
(k) Slat switch



Autopilot Flight Director System Bite Interfaces and Sensors
Figure 2

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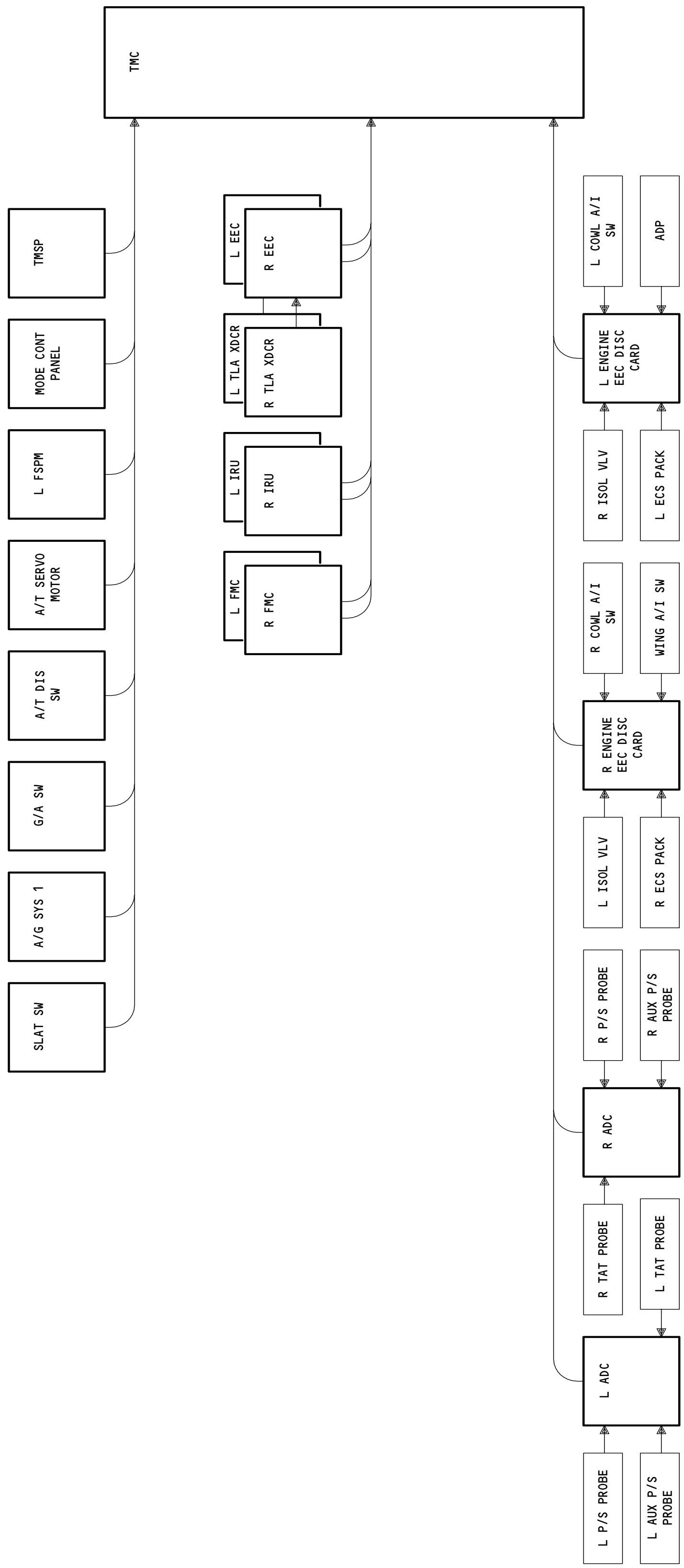


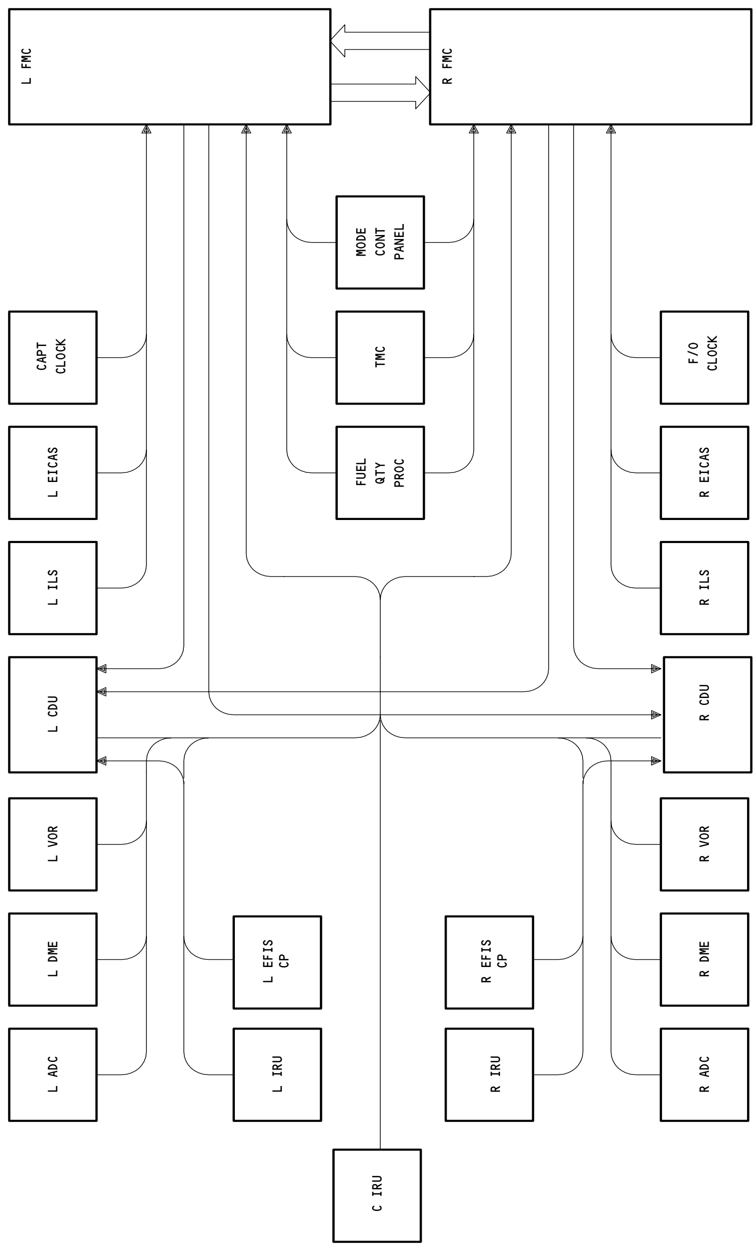
Thrust Management System Bite Interfaces and Sensors
Figure 3

D. Flight Management Computer System (Fig. 4)
   (1) The Flight Management Computer System consists of two FMCs and cross channel data buses, one MCP, and two FMC Control Display Units (CDUs).
   (2) The FMCs receive and report fault data from the following LRUs.
      (a) Captain's and First Officer's clock
      (b) EFIS control panel L,R
      (c) EICAS computer L,R
      (d) ILS receiver L,R
      (e) VOR receiver L,R
      (f) DME L,R
      (g) ADC L,R
      (h) IRU L,C,R
      (i) Fuel Quantity Processor
      (j) TMC

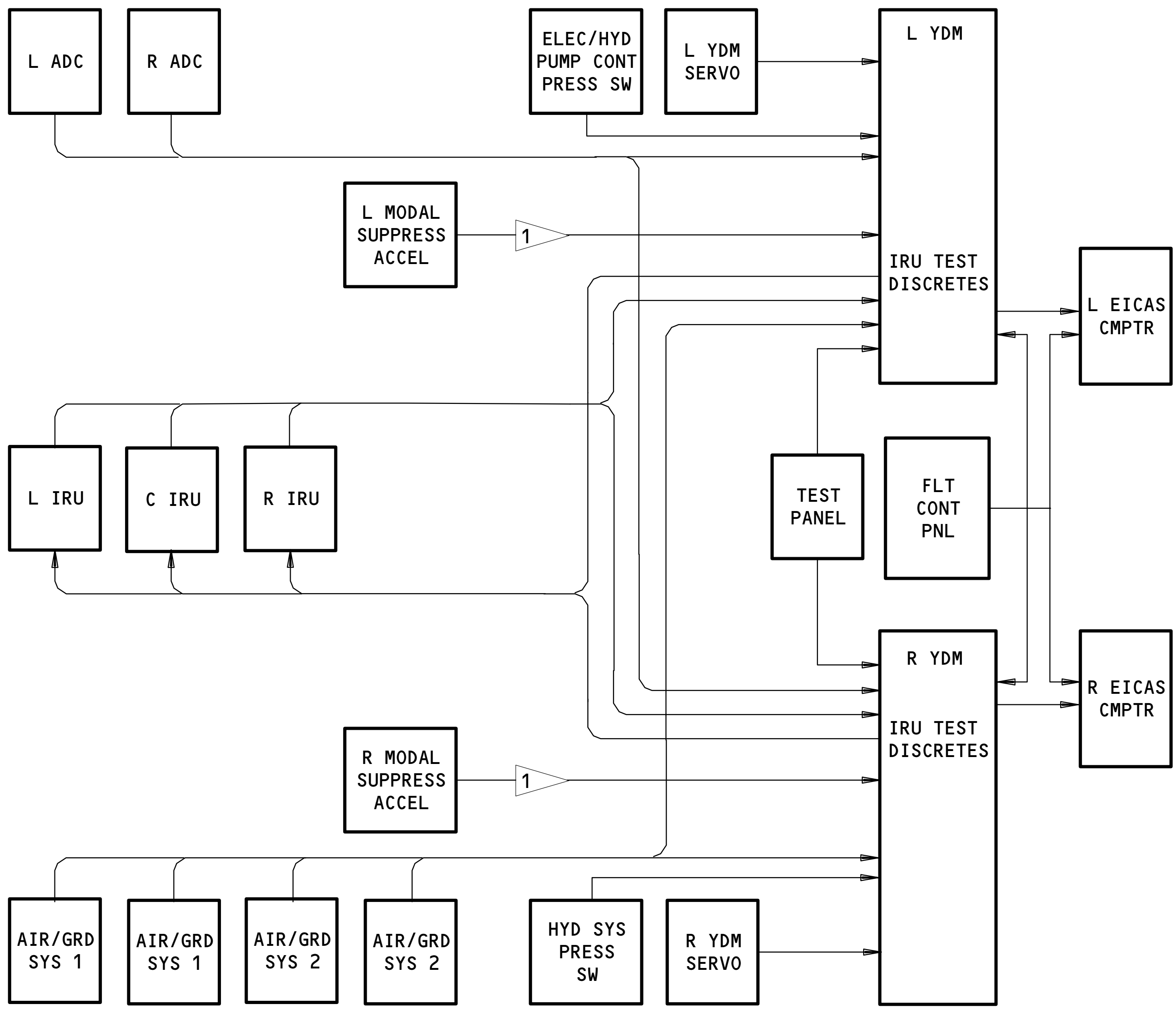
E. Yaw Damper System (Fig. 5)
   (1) The yaw damper system consists of two Yaw Damper Modules (YDM), a control panel, and a test switch.
   (2) The YDMs monitor the following LRUs.
      (a) YDM Servo L,R
      (b) Elect/Hydraulic Pump and Hydraulic System Pressure
      (c) IRU L,C,R
      (d) ADC L,R
      (e) Air/Ground System 1, 2
      (f) 767-300;
          Modal Suppression Accelerometer

3. <u>Operation</u>

A. Flight Faults
   (1) At takeoff the FCCs, FMCs, and TMC will purge previously-recorded fault data from memory. Throughout the flight each computer will monitor interfaces, interfacing LRUs, and its own circuitry for faulty operation. Any detected faults will be stored in memory for later use.
      (a) Faults detected in-flight can be stored in non-volatile memory and stored in volatile memory. Faults permanently stored in non-volatile memory are for shop use and can only be cleared in the shop. Faults temporarily stored in volatile memory are reported to the MCDP and cleared from memory. Data stored in the two types of memory generally will not be identical.
      (b) Faults reported to the MCDP are those that have a flight deck effect.
   (2) Transfer of fault data from the FCCs, FMCs, and TMC takes place at touchdown.
      (a) At touchdown, when the air/ground system transitions from air to ground and ground speed drops below 40 knots, the MCDP will be powered up automatically and undergo self test.



Flight Management Computer System Bite Interfaces and Sensors
Figure 4



1 767-300 AIRPLANES

Yaw Damper Bite Interface and Sensors
Figure 5

(b) Upon successful completion of power up and self test the MCDP will interrogate the FCCs, FMCs, and TMC for stored fault data, initiating automatic transfer.

(3) Data transferred to the MCDP at touchdown is stored in non-volatile memory. Two classifications of flight faults are Last Faults and Previous Faults.

(a) Last Flight Faults are those occurring during the last flight leg and represent faults that exist currently.

(b) Previous Flight Faults occurred before the last flight. The MCDP can keep in memory a mixture of the last 99 flights that had and did not have failures. If more than 10 of these flights had failures, then only the flights that occurred until there were 10 flights with failures will be kept in the memory of the MCDP. This data can be used historically, to look for failures that occurred more than once, or to keep the data for another use.

(c) Flight fault data stored in the MCDP can be erased only in the shop.

(4) Fault messages can be displayed on the MCDP or EICAS if a remote panel is used. The message is displayed in two lines.

(a) Top line of the flight faults messages displays the flight deck effect and historical flight tag. The historical flight tag identifies the flight leg in which the fault occurred. 00 is present fault and 01 thru 99 occurred 1 to 99 flights ago.

(b) Bottom line of the flight faults messages displays the faulty LRU.

(5) Diagnostic codes related to flight faults can be displayed by pressing the flight fault mode switch/light causing the flight deck effect displayed on the message top line to be replaced by the diagnostic.

B. Ground Tests

(1) Ground tests provide a means of verifying and amplifying on flight faults to aid in fault isolation. Ground test also provide fault indications when no flight faults were recorded.

(a) Tests 01 thru 16 are LRU tests oriented towards specific components.

(b) Tests 30 and 40 are system tests involving the entire autoflight system.

(c) Tests 51 thru 69 are supporting tests oriented towards specific function involving several components.

(2) 01 FCC (Left, Right, and Center)

(a) Displays any FCC self-test faults and FCC interface faults from the associated system interfaces.

(3) 02 TMC

(a) Displays any TMC self-test faults and TMC interface faults from associated system interfaces.

(4) 04 MCP
  (a) Displays any FCC and TMC self-test faults. Displays FCC and TMC interface faults related to MCP.

  NOTE: Failure of FCC L and FCC C, or FCC R will cause half-lighting of lower half of MCP switch/light indicators. Failure of TMC will prevent EPR (or N1 or THR) and SPD switch/lights from lighting.

  (b) Checks the status of the MCP program pin wiring for customer selected options.
  (c) Displays the interaction message for checking the MCP lightswitches.
  (d) Displays interaction messages for checking the ON and OFF position of the Captain's flight director switch.
  (e) Displays interaction messages for checking the ARM and OFF position of the autothrottle ARM switch.
  (f) Displays the interaction message for checking the command function of the indicated airspeed/mach select switch.
  (g) Displays the interaction message for checking the command function of heading select switch.
  (h) Displays the interaction message for checking the ON and OFF positions of the First Officer's flight director switch.
  (i) Displays interaction messages for checking the down and up position of the disengage bar.
  (j) Displays interaction messages for checking the command function for each position of the bank limit switch.
  (k) Displays interaction messages for checking the control function of the indicated airspeed/mach display rotary control.
  (l) Displays interaction messages for checking the control function of heading display rotary control.
  (m) Displays interaction messages for checking the control function of the vertical speed display rotary control.
  (n) Displays interaction messages for checking the control functions of the altitude select display rotary control.
(5) 05 TMSP
  (a) Displays TMC self-test faults.
  (b) Displays TMSP-to-TMC interface faults
  (c) Does a TMSP SW/MODE DISPLY test routine which checks the mode display and command function of continuous, climb, and cruise mode switching. Displays an interactive message at end of the test.
  (d) Does a TMSP TEMP SELECT test with interaction messages to check control function of temperature select control display.
(6) 06 ASA
  (a) Displays any FCC self-test faults.
  (b) Displays FCC interface faults related to ASA.
  (c) Checks all ASA displays and their reset function.
(7) 07 SERVO AIL
  (a) Displays any FCC self-test faults and FCC interface faults related to aileron servos.

(b) Provides interaction message to verify hydraulic pressure on (VFY HYD ON).
(c) Provides interaction message to verify aileron trim set to zero.
(d) Provide interaction message to push MCP CMD switches. This action automatically synchronizes the aileron and elevator servos to the present control surface position. Servos will not engage until servos synchronized.
(e) Wheel centers, then moves clockwise and then counterclockwise and back to center. Inboard ailerons are driven to a plus and minus 10 degree position. FCCs check that each position is within limits.

(8) 08 SERVO ELEV
(a) Displays any FCC self-test faults and FCC interface faults related to elevator servo.
(b) Provides interaction message to verify hydraulic pressure on (VFY HYD ON).
(c) Provides interaction message to verify elevator trim set to zero.
(d) Displays elevator feel position pressure for operator to verify pitot pressure is zero.
(e) Provide interaction message to push MCP CMD switches. This action automatically synchronizes the aileron and elevator servos to the present control surface position. Servos will not engage until servos synchronized.
(f) Column centers, then moves aft (trailing edge up), then forward (trailing edge down) and back to center. Elevators are driven to a plus 20 degrees (trailing edge up) and a minus 9 degrees (trailing edge down) and the FCCs check that each position is of the elevator servos and neutral shift tranducers (FEEL POS) are within limits.

(9) 09 SERVO RUD
(a) Displays any FCC self-test faults and FCC interface faults related to rudder servo.
(b) Provides interaction message to verify hydraulic pressure on (VFY HYD ON).
(c) Provides interaction message to verify rudder trim set to zero.
(d) Rudder centers, then moves rudder trailing edge right, then left and back to center. Rudder driven to a plus and minus 20 degrees and FCCs check that each position is within limits.

(10) 10 SERVO A/T
(a) Displays TMC self-test faults.
(b) Throttle is automatically initialized to aft position. The autothrottle is then driven forward for 8 seconds at 10 degrees/second and checked for a TLA of 84 ±4 degrees, and a tachometer signal feedback of 7 to 12 degrees/second. The autothrottle is driven aft for 8 seconds at 10 degrees/second, and checked for a tachometer feedback signal of 7 to 12 degrees/second.

(11) 11 SW A/P DISC
  (a) Displays any FCC self-test faults, and FCC interface faults related to A/P Disc switch.
  (b) Instructs the operator to close the Captain's and First Officer's autopilot disconnect switches. The switch output is monitored for autopilot disconnect command response.
(12) 12 SW A/T DISC
  (a) Displays any TMC self-test faults and TMC interface faults related to A/T disconnect switch.
  (b) Instructs operator to close the left and right autothrottle disconnect switches. Switch output is monitored for autothrottle disconnect command response.
(13) 13 SW G/A
  (a) Displays any FCC and TMC self-test faults FCC and TMC interface faults related to G/A switch.
  (b) Instructs the operator to close the Captain's and First Officer's Go-Around (G/A) switches. The G/A switch command response is monitored at each FCC and TMC.
(14) 14 XDCR COL L
  (a) This test does not apply to this airplane configuration. If test selected, MCDP displays 14 NO TEST THIS A/C CONFIG.
(15) 15 XDCR COL R
  (a) This test does not apply to this airplane configuration. If test selected, MCDP displays 15 NO TEST THIS A/C CONFIG.
(16) 16 XDCR WHL
  (a) This test does not apply to this airplane configuration. If test selected, MCDP displays 16 NO TEST THIS A/C CONFIG.
(17) 17 PVD
  (a) This test does not apply to this airplane configuration. If test selected, MCDP displays 17 NO TEST THIS A/C CONFIG.
(18) 30 CURRENT FAULT REPORT
  (a) Displays all LRU and interface faults detected and stored in the FCCs and TMC while the MCDP operates in the ground test mode. Ground fault messages and diagnostic codes are the same as flight fault messages except the flight deck effect and intermittent bit status are not displayed. Diagnostic codes are displayed by pressing the GRD TEST mode switch.
  (b) Provides interaction message to allow operator to skip Interface Fault messages.

(c) Provides interaction message to allow operator to select current Ground Fault display.

(19) 40 AUTOLAND
(a) Displays the current ground faults and interface faults related to the autoland system. The autoland related interfaces are automatically monitored. It automatically performs LRU and support tests related to the autoland system that do not require interaction with the operator.
(b) Performs automatically LRU and support tests related to the autoland system that require no manual preconditioning except for applying electrical power.
(c) Performs automatically LRU and support tests related to the autoland system that require manual preconditions but no hydraulic power.
(d) Performs manual interface checks related to the autoland system.
(e) Performs manual LRU and support tests related to the autoland system.
(f) Performs A/P servo and stab trim tests.
(g) Performs Autothrottle tests.

(20) 51 AIR/GRD RLY
(a) Displays any FCC and TMC self-test faults.
(b) Interaction test of the landing gear system 1 air/ground relay and its interface to the FCCs and TMC.
(c) Interaction test of the landing gear system 2 air/ground relay and its interface to the FCCs.

(21) 52 TMC RLY/SW
(a) Displays any TMC self-test faults and TMC interface faults related to Environmental Control System (ECS) cards, pneumatic system, anti-ice, and thrust reverser lever position.
(b) Tests for discrete output signals from the left and right Electronic Engine Controls (EEC) discrete cards, pneumatic shutoff and isolation valves, and anti-ice valves. Thrust reverser lever position is sent to the TMC from the EEC on the ARINC 429 buses.

(22) 56 FCC CONFIG/OPT
(a) Displays any FCC self-test faults and FCC interface faults related to parity.
(b) Checks the status of the FCC program pin wiring for airplane and hardware configurations and customer selected options.

(23) 57 TMC CONFIG/OPT
(a) Displays any TMC self-test faults and TMC interface faults related to parity.

(b) Checks status of TMC program strapping for engine and airplane configuration, autothrottle, and thrust limit customer selected options.

(24) 59 FCC INSTR

(a) Displays any FCC self-test faults.

(b) Sends test routine command and parameters to FCCs, and displays operator interaction message.

(c) Tests operation of Captain's and First Officer's MCP flight director switches and the ability of FCCs to move the pitch command bar 10 degrees. Checks each valid FCC separately.

(25) 60 TMC INSTR

(a) Displays any TMC self-test faults.

(b) Sends test routine number and parameters to TMC and displays operator action message.

(c) Tests the ability of the TMC to control the Captain's and First Officer's EPR instruments target index. EPR target index driven to a test value of 1.50

(26) 64 SPD BK/FLAP XDCR

(a) Displays any TMC and FCC self-test faults and FLAP XDCR related interface faults.

(b) Displays a VFY HYD ON message.

(c) Provides interaction messages to check the FCCs sensing of the speedbrakes down, armed, and up.

(d) Shows the messages that tell the person to set the flap lever to 25, 15, 1, and 0.

(e) Does a operational test of the flap position sensing transducer and the slat switch position for the FCC and TMC interfaces.

(27) 65 STAB TRIM

(a) Displays any FCC self-test faults and FCC interface faults related to stab trim.

(b) Provides action message VFY HYD ON.

(c) Tests the ability of the FCC to command the Stabilizer Trim and Elevator Asymmetry Limit Module (SAM) to drive the horizontal stabilizer UP and DOWN. It checks the FCC-to-SAM interface.

(28) 66 XDCR OUTPUTS

(a) Displays any FCC and TMC self-test faults, and transducer related interface faults.

(b) Displays position of horizontal stabilizer, aileron servo, aileron surface, elevator servo, elevator surface, rudder servo, rudder surface, and flaps to the FCC and TMC.

(29) 67 AIL SURF LIMIT

(a) Displays any FCC self-test faults and FCC interface faults related to aileron servo.

(b) Gives the operational messages VFY HYD ON, SET AIL TRIM = 0 - ADV, and A/P ENG LCR TO CMD (for only the FCCs that have good data) (LCR = Left, Center, Right).

(c) Drives aileron to its positive limit if operator responds YES/ADV to action message AIL POS LIM?

(d) Drives aileron to its negative limit if operator responds YES/ADV to action message AIL NEG LIM?

(e) Displays aileron surface position in degress.

AIL SURF DEG
±XX.X ±XX.XX ±XX.XX   where XX.X = decimal number

(30) 68 ELEV SURF LIM

(a) Displays any FCC self-test faults and FCC interface faults related to aileron servo.

(b) Give the operational messages VFY HYD ON, SET STAB TRIM = 6 - ADV, and A/P ENG LCR TO CMD (for only the FCCs that have good data) (LCR = Left, Center, Right).

(c) Drives elevator to its positive limit if operator responds YES/ADV to action message ELEV TO POS LIM?

(d) Displays elevator surface position in degress.

ELEV SURF DEG
±XX.X ±XX.X ±XX.X   where XX.X = decimal number

(e) Drives elevator to its negative limit if operator responds YES/ADV to action message ELEV TO NEG LIM?

(f) Displays elevator surface position in degress.

ELEV SURF DEG
±XX.X ±XX.X ±XX.X

(31) 69 RUD SURF LIM

(a) Displays any FCC self-test faults and FCC interface faults related to aileron servo.

(b) Give the operational messages VFY HYD ON, SET RUD TRIM = 0 - ADV, and A/P ENG LCR TO CMD (for only the FCCs that have good data) (LCR = Left, Center, Right).

(c) Drives rudder to its positive limit if operator responds YES/ADV to action message RUD TO POS LIM?

(d) Displays rudder surface position in degress.

    RUD SURF DEG
    ±XX.X ±XX.X ±XX.X

(e) Drives rudder to its negative limit if operator responds YES/ADV to action message RUD TO NEG LIM?

(f) Displays rudder surface position in degress.

    RUD SURF DEG
    ±XX.X ±XX.X ±XX.X

C. Fault Message Analysis

(1) Fault messages that the MCDP shows have the idea of what the flight deck effects are, and try to point to the cause of a failure. Hardware monitors and software monitors find problems with the unit. The monitors then give the data for the failure message and the data to keep the failure in memory.

(a) The common mode monitors, used to make sure of the LVDT operation, and the hardware detent comparators are the hardware monitors in the system. All the other monitors are software monitors.

(b) Software monitors test hardware functioning, interfaces, and data reasonableness. The various types of software monitors are:

1) Activity monitors.
2) Wrap-around test monitors.
3) ARINC 429 sign status matrix monitors (SSM).
4) Command/response monitors.
5) Discrete I/O monitors.
6) Detent comparators.
7) Operational monitors.
8) Signal Selection Fault Detection voting circuits (SSFD).

(2) Interface Messages

(a) Interfaces are tested at power up and during ground tests. Messages "NO INFC LRU/LRU" will not be reported as flight faults. Flight faults will report group faults which can be related to a ground test NO INFC message.

(b) If the MCDP generates a NO INFC message for FCC, TMC, or FMC on power up, the MCDP will not interrogate that LRU for flight faults and will not exercise ground tests for that LRU.

(c) The FCC and TMC will report NO INFC messages if an activity monitor fails to detect any change in data received over the interface for specified period of time.

(3) Mode Errors

(a) Mode errors are reported for flight deck effects caused by crew procedures. For most mode errors there is no fault and no need to replace hardware.

1) Typical crew caused mode errors include:

a) Failure to select TRUE heading when flying near the north pole.

b) Attempting to change ILS frequency after capturing glideslope.

c) Failure to disconnect a single channel engaged autopilot prior to flight and touchdown.

d) Pilot input via wheel/column that overrides a single channel engaged autopilot.

(b) Mode errors with diagnostic codes 247 or 143 may be hardware problems and may require corrective action.

(4) Sign Status Matrix (SSM) Faults

(a) LRUs transmitting ARINC 429 data provide SSM information in data. The SSM provides status of normal operation, functional test, no computed data, or failure warning.

(b) The FMCs and the TMC will usually keep any SSM, other than those that result from correct operation, in non-volatile memory as failure data. The FCCs will usually only keep any SSM that shutdown operation of the FCC in non-volatile memory as failure data. This data can be read in the shop.

(c) SSM data forwarded to the MCDP will consist normally of only failure warn. Three situations can occur that will result in SSM of no computed data to be forwarded to the MCDP.

1) Loss of data from IRU can result in diagnostic codes 57, 62, 75, 76, and 247 associated with NCD from IRU.

2) Loss of ILS data at or after capture for glide slope or localize can result in diagnostic codes 199, 200, 202 and 203 associated with NCD from ILS.

3) Loss of FMC data caused by NCD from IRU can result in diagnostic codes 120 and 121 associated with NCD from FMC.

(5) Signal Selection Fault Detection (SSFD) Faults

(a) Signal Selection Fault Detection is a procedure that is used to look at the quality of the data. SSFD is used when dual or triple data sources are available. Any one data source can be found to have bad data when compared with two other data sources of the same kind. A data source can also be found to have bad data when compared with a data valve that was calculated as the median.

(b) SSFC monitors have a process they go through to not let the autopilot and flight director use the bad data from a defective sensor to calculate the data they will use. SSFD monitors come on when the airspeed is more than 60 knots and an autopilot is in use, or if one or two of the flight directors are on. The monitors continue to compare data values independently of how many autopilot channels are in use. The diagnostic code intermittent bit indicates which of the three FCCs has bad data. This FCC, because of the bit, will not be permitted to have a say as to what data is good or bad. Without this bit, this FCC will possibly have that say and permit a data output from an FCC that has bad data.

(c) SSFD faults are inhibited on the ground and cannot be duplicated during ground test.

(d) SSFD faults can be generated for the following conditions.

1) ILS data fail during ILS capture; diagnostic codes 218, 219.
2) Localizer or glideslope deviation disagrees with data receiver on cross-channel; diagnostic codes 236, 237.
3) IRU data faulty or missing; diagnostic codes 114 thru 118, 122 thru 125, 132, 215, 216, 230, 231, 238, 240, 241, and 249.
4) LRRA data faulty; diagnostic codes 176, 221.
5) Elevator surface position data fault; diagnostic codes 142, 143.
6) Aileron surface position data fault; diagnostic codes 211, 214.
7) Rudder surface position data fault; diagnostic codes 212, 217.
8) Feel computer data fault; diagnostic codes 222, 223.
9) Stabilizer position data error; diagnostic codes 213, 229.
10) Flap position data error; diagnostic codes 210, 224.

D. Yaw Damper Module BITE

(1) The yaw damper module BITE consists of a number of monitors that check critical data, and of test routines designed into the module.

(a) Power-up testing occurs when power is applied to the module. Testing consists of fault disengage, servo loop monitor, command coincidence monitor, and memory read/write tests.

(b) Preflight testing occurs when the YDM test switch on the P61 panel is set to L or R. Testing causes the inertial reference system to self test, moves the rudder and checks rudder position, and check IRU inputs.

(c) Continuous inflight testing of data inputs and internal operation is performed by monitor circuits.

(2) Faults detected by yaw damper monitor circuits results in setting fault balls on the face of the module.

(a) The ADC input monitor checks ADC output bus 1 data for activity, parity, SSM, and data reasonableness. The monitor output sets the ADC fault ball.

(b) The IRU input monitor checks IRU data bus 3 data for activity, parity, SSM, and SSFD. The monitor output sets the IRU fault ball.

(c) The air ground input monitor compares data from three air ground relays. If any one input disagrees with the other two a fault is declared. The monitor output sets the AIR/GRD fault ball.

(d) The hydraulic pressure monitor checks hydraulic pressure switch high pressure and low pressure inputs. Monitor output sets the HYD PRESS fault ball.

(e) A common mode monitor checks LVDT voltage. Low voltage is declared a fault. Monitor output sets the LVDT fault ball.

(f) A command/response monitor compares yaw damper servo position to commanded position. A fault is declared if a disagree is detected. Monitor output sets the YD ACT fault ball.

(g) Monitor microprocessor output is compared to control microprocessor output. A disagree sets the YDM fault ball.

E. Troubleshooting Techniques

(1) Autoflight BITE can reduce troubleshooting time and eliminate unnecessary replacement of LRUs if used properly.

(a) Always check flight faults, record all faults and related diagnostic.

(b) If there are no current flight faults, perform ground test 30 and record all faults and associated diagnostic codes.

(c) If no flight fault exists for a particular flight squawk, perform an associated ground test.

(2) Interface faults should be corrected first, and associated ground tests rerun to ensure a faulty LRU is not being masked by another fault that renders the interface inoperative.
   (a) If an interface fault cannot be isolated to a specific LRU or wire, redundant LRUs can be swapped and the fault tracked to relate it to an LRU.
   (b) Fault balls and other fault indicators not related to autoflight BITE can be checked and individual LRU self-tests run.

(3) LRU faults detected by MCDP ground tests will remain in the MCDP even after the LRU is replaced. It is necessary to exit ground test to clear the faults from the MCDP.

(4) Faults that occur prior to take-off or after landing will not be reported as flight faults. These faults will be detected and reported by ground tests only.

AUTOFLIGHT BITE - MAINTENANCE PRACTICES

1. General
   A. This maintenance practice gives the data for the use of the Maintenance Control Display Panel (MCDP) for review of flight failures put into memory. The data is also here for the use of the MCDP to run tests and trouble shooting.
   B. Ground tests can be done by the MCDP for individual computers, line replaceable units (LRUs), and interfaces. The tests are shown on the MCDP front panel. The operation of each test is given in this section of maintenance practices.
   C. Procedures are given to energize the MCDP and do the self-test, to do a check for flight faults, and to do the ground tests.

TASK 22-00-02-712-432

2. Energize the MCDP and Do the Self-Test
   A. General
      (1) The MCDP is energized manually on the ground when flight faults are read or ground tests are done. The MCDP is energized when the MCDP ON/OFF switch is pushed.
      (2) When the MCDP is started, it does a sequence of self-tests. The FAIL light in the ON/OFF switch/light will come on when the MCDP is energized and remain on for 5 seconds. Also, the FLT FAULTS mode light and the GRD TEST mode light will come on. All areas of the display will come on. When the FAIL light goes off, the MCDP is prepared for use. The FAIL light will stay on if there is a failure during self-test.
      (3) Fault messages related to self-test failures are shown after the self-test is completed. These messages are not kept in non-volatile memory.
      (4) The MCDP will go off if power is off for more than 200 milliseconds. The MCDP will stay on if power is off for less than 200 msec. Messages related to no power are these:

MCDP FAIL RESTART	The MCDP had an error when the power came back on. Open and then close the MCDP circuit breakers to energize the MCDP.
PWR FAIL RST IN PROGRESS	This indication will show if the power goes off and the MCDP starts again during the ground test mode. The indication comes on for 3 seconds, then the ground test can be done.

   B. References
      (1) AMM 24-22-00/201, Electrical Power - Control

C. Access
   (1) Location Zones
         119   Main Equipment Center

   (2) Access Panels
         119BL   Main Equipment Center

D. Energize the MCDP and Do the Self-Test

   S 862-330
   (1) Close these circuit breakers on the overhead panel P11:
         11C29, LANDING GEAR POSITION AIR/GND SYS 2 ALTN
      (a) 11C29, LANDING GEAR POSITION AIR/GND SYS 2 ALTN
      (b) 11C30, LANDING GEAR POSITION AIR/GND SYS 1
      (c) 11U9, MAINT CONT DSPL
      (d) 11U15, AIR/GND SYS 1
      (e) 767-300 AIRPLANES;
          11U23, LANDING GEAR POSITION AIR/GND SYS 2
      (f) 767-200 AIRPLANES;
          11U24, LANDING GEAR POSITION AIR/GND SYS 2

   S 862-331
   (2) Supply electrical power (AMM 24-22-00/201).

   S 862-332
   (3) Put the MCDP ON/OFF switch to the ON position.
      (a) Make sure the FAIL light and the GRD TEST light come on for 5 seconds and then go off.
      (b) Make sure the FLT FAULT light comes on and stays on.

   S 752-333
   (4) To do a check for flight faults, do to the Flight Faults procedure. To do the ground tests, go to the Ground Test procedure.

TASK 22-00-02-742-001

3. Flight Faults

A. General
   (1) The Flight Faults Mode gives faults that occurred during a flight. Only the faults that cause a flight deck effect are shown as flight faults. The faults can be read after each flight when the MCDP is manually energized. After the MCDP has been energized and the self-tests done, the MCDP will automatically go into the flight faults mode.

B. References
   (1) AMM 24-22-00/201, Electrical Power - Control

C. Access

(1) Location Zones

119	Main Equipment Center
211/212	Flight Compartment

(2) Access Panels

119BL	Main Equipment Center

D. Show the Last Flight Faults

S 712-002

(1) Do the Energize the MCDP and Do the Self-Test procedure.

NOTE: Last Flight Faults shows the faults kept by the MCDP during the last time the MCDP was automatically turned on by touchdown. If the Flight Faults Mode is engaged on the ground, it causes the MCDP to show LAST FLT FAULT?. Push the YES/ADV button to show a fault that was found during the last flight. Each time the YES/ADV switch is pushed a different fault will show until all of the faults have been shown. The display will then show ALL LAST FLT FAULTS DSPLY.

S 982-003

(2) Push the NO/SKIP switch to go out of last Flight Faults.

NOTE: If no faults are found, the message that shows after the YES/ADV switch is pushed will be NO LAST FLT FAULT.

E. Show the Previous Flight Faults

S 712-004

(1) The Previous FLight Faults show the flight faults kept from flights before the last flight. When the MCDP shows the message ALL LAST FLT FAULT DSPLY or NO LAST FLT FAULT, push either YES/ADV or NO/SKIP to view the Previous Flight Faults.

NOTE: When you go into Previous Flight Faults, the message PREV FLT FAULT? shows. The MCDP will show faults from the flight before the last flight the same way as the Last Flight Fault messages were shown. The same messages for all previous faults shown and no previous faults are shown as applicable. The Previous flight Faults gives historical flight data. Faults from up to 99 flights are kept in the MCDP unless more than 10 flights have had faults. Faults for a specified flight are kept together and given a number (02 = two flights before, 05 = 5 flights before, etc.). Flight numbers are left off if the flight had no faults.

S 982-005

(2) To show Subsequent Previous Flight Faults push the YES/ADV switch.

TASK 22-00-02-742-006

4. Ground Test

A. General

(1) The Ground Test Mode gives a sequence of different tests to make sure specified parts of the Autopilot/Flight Director System (AFDS) and Thrust Management System (TMS) are satisfactory. Some of the tests are part of the AFDS Adjustment/Test (AMM 22-10-00) and the TMS Adjustment/Test (AMM 22-32-00). Others tests are specified by different Removal/Installation procedures and Maintenance Practices. The ground tests can also be used to make sure AFDS components are OK before takeoff. After the MCDP has been energized and the self-test done, the MCDP will automatically go into the flight faults mode. Push the GRD TEST switch to go into the Ground Test Mode.

(2) The tests are given by number sequence. The tests can be done by themselves or in a different sequence if the applicable steps to prepare for the test have been done.

(3) The MCDP can be put into GRD TEST MODE after the MCDP has been manually energized. The GRD TEST MODE program tells the FCC's and the TMC to do self-tests. The MCDP tells each computer to make sure its interfaces are OK. The results of the self-tests and interface tests are kept by the MCDP for future reference. The MCDP makes sure the self-tests and interface test results are OK every time a ground test is done. A computer that is not OK and is necessary to do the specified test is shown. The test can still be done, but not for the channel of the bad computer. The MCDP then shows interface faults kept by the MCDP that effect the specified test. Leave the GRD TEST MODE and then go into GRD TEST MODE again to bring the data up to date. When a GND TEST fault is corrected, you must go to the FLT FAULTS MODE and then go back into the GRD TEST MODE to remove that fault from the MCDP.

(4) For some steps it is necessary for a person to do an operation or reply. This may be to answer a question (YES or NO) or to do or not to do the next step (ADV or SKIP). The step can possibly tell the person to operate a switch or circuit breaker, supply or remove hydraulic power, or monitor a display or effect.

(5) Use the YES/ADV switch to start the test that shows, give a reply to MCDP indications or instructions, to show all of the failures for the test being done, and to continue with a test when a failure stops the test.

(6) The NO/SKIP switch usually causes the MCDP to stop the GRD test that is in operation and go to the subsequent GND TEST. The MCDP will not go to the subsequent GND TEST when the messages that follow show on the MCDP.

(a) 30 INFC FAULT? -
the MCDP will go to 30 CURRENT OP FAULT? when the NO/SKIP switch is pushed.

(b) 30 VFY THRST LVR SYST CLR-ADV -
the MCDP will go to 30 CURRENT OP FAULT? when the NO/SKIP switch is pushed.

(c) 40 A/L BUS ISLN TEST? -
the MCDP will go to 40 VFY ASA BLANK - RST ASA when the NO/SKIP switch is pushed.

(d) 40 A/L SW LIGHTS TEST? -
the MCDP will go to 40 A/L SRVO TEST? when the NO/SKIP switch is pushed.

(e) 40 A/L SRVO TEST? -
the MCDP will go to 40 A/L A/T TEST? when the NO/SKIP switch is pushed.

(7) Upon completion of a ground test, the MCDP display will read TEST COMPLETE with the test number before it.

NOTE: Test 40-AUTOLAND will not display 40 TEST COMPLETE if a test step is skipped or a fault message is displayed.

B. Equipment
(1) Nose Gear Steering Valve Lockpin - A09003-1
(2) Remote Control Unit, Maintenance Control Display Panel - A22001-22 (preferred), A22001-15 (optional)

C. References
(1) AMM 9-11-00/201, Towing
(2) AMM 22-41-01/401, Maintenance Monitor
(3) AMM 24-22-00/201, Electrical Power - Control
(4) AMM 27-21-00/501, Rudder and Rudder Trim Control System
(5) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
(6) AMM 31-41-00/501, Engine Indication and Crew Alerting System (EICAS)
(7) AMM 34-11-00/201, Pitot-Static System
(8) AMM 34-21-00/501, Inertial Reference System
(9) AMM 36-00-00/201, Pneumatic Power
(10) AMM 78-31-00/201, Thrust Reverser System (Deactivation/Activation)

D. Access
(1) Location Zones
119 Main Equipment Center

(2) Access Panels
119BL Main Equipment Center

E. Prepare to Test

S 862-008
(1) Open these circuit breakers on the overhead panel P11 to prevent unwanted aural warnings:
(a) 11B16, AURAL WARN SPKR
(b) 11H35, AURAL WARN SPKR R

S 862-009
(2) Supply electrical power (AMM 24-22-00/201).

S 862-011
(3) Do the Energize the MCDP and Do the Self-Test steps.

S 862-012
(4) If the remote MCDP is used, do the Energize the MCDP and Do the Self-Test on the main MCDP then do the steps that follow:
(a) Install the remote MCDP panel.
(b) Close the EICAS circuit breakers on P11 panel (31-41-00).
(c) On the EICAS maintenance panel (P61) push the CONF/MCDP switch to get the MCDP display on the EICAS lower display unit.

NOTE: The same messages shown on the MCDP in the main equipment center will be shown on the EICAS lower display unit. The indication MCDP FLT FAULTS or MCDP GRD TEST is shown to identify which mode the MCDP is in.

(d) Make sure the display shows the message MCDP OFF.
(e) Put the MCDP remote panel ON/OFF switch to the ON position.
(f) Operate the GRD TEST switch.

F. Supply Hydraulic Power (Necessary for Ground Tests 07, 08, 09, 13, 40, 52, 64, 65, 66, 67, 68, 69)

S 862-013
(1) Hydraulic pressure can be supplied by one of these two: the airplane electrically driven pumps or the ground hydraulic carts.

S 202-014

(2) Operation of hydraulic devices other than those necessary by these tests is to be prevented if possible. This makes sure that there is sufficient power when airplane electrically driven hydraulic pumps are used.

S 862-015

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(3) Do the deactivation procedure for the spoilers (27-61-00) or move all persons and equipment away from the spoilers.

S 862-016

(4) Supply pressure to the left, right, and center hydraulic systems and the hydraulic reservoirs (29-11-00) after the MCDP message VFY HYD ON is shown unless told differently by the procedure. Push the YES/ADV switch to show that hydraulics are on.

G. 01 FCC

S 862-017

(1) Do the Prepare to Test steps.

S 862-018

(2) If the test is to be done after an LRU replacement or a fault correction, make sure MCDP has been put out of GRD TEST MODE. Go into FLT FAULTS MODE or set MCDP to OFF and then go into GRD TEST MODE again.

NOTE: Before MCDP Test 01-FCC is done, the MCDP must be put out of GRD TEST MODE to remove, from the MCDP memory, faults that have been corrected.

S 742-019
(3) Do MCDP test 01.

NOTE: The MCDP shows FCC # FAIL if the FCC had a self-test failure when put into GRD TEST MODE. The MCDP shows FCC # if the FCC had a satisfactory self-test and there is an interface fault. The MCDP does not test LRUs or input signals. Each FCC is tested individually.

S 862-020
(4) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

H. 02 TMC

S 862-022

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Connect External Electrical Power (AMM 24-22-00/201).

S 862-476
(2) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoilers.

S 862-021

CAUTION: MAKE SURE THE ENGINES ARE NOT IN OPERATION. THIS TEST INCLUDES AUTOMATIC MOVEMENT OF THE THRUST LEVERS AND COULD CAUSE AIRPLANE MOVEMENT IF THE ENGINES ARE IN OPERATION. INJURY TO PERSONS COULD OCCUR.

(3) Make sure the engines are not in operation.

S 862-023
(4) Do the Prepare to Test steps.

S 862-024
(5) If the test is to be done after an LRU replacement or a fault correction, make sure MCDP has been put out of GRD TEST MODE. Go into FLT FAULTS MODE or set MCDP to OFF and then go into GRD TEST MODE again.

NOTE: Before MCDP Test 02-TMC is done, the MCDP must be put out of GRD TEST MODE to remove, from the MCDP memory, faults that have been corrected.

S 862-033
(6) Set the left and right EEC MAINT ENG POWER (EEC GROUND TEST) switches on the P61 panel to the TEST position. This will supply power to the EEC.

NOTE: Failure to power the EEC before the selection of the MCDP GRD TEST MODE will cause NO INFC TMC EEC L PRIM (SEC) and NO INFC TMC EEC R PRIM (SEC) messages to be shown during the test.

S 742-043
(7) Do MCDP test 02.

NOTE: The MCDP shows TMC FAIL if TMC had a self-test failure when put into GRD TEST MODE. If TMC FAIL message does not show, the MCDP shows messages to tell the person to clear the thrust lever system. The MCDP then tells the person to set the A/T switch to ARM. The MCDP then shows TMC interface failures related to the autothrottle servo. The TMC test does not make sure that the LRUs or input signals are correct.

S 862-044
(8) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

I. 04 MCP

S 862-045
(1) Do the Prepare to Test steps.

S 862-046
(2) Set the switches on the MCP as follows:
(a) A/T ARM to off
(b) DISENGAGE BAR up.
(c) The two F/D switches to off

S 742-049

(3) Do MCDP test 04.

NOTE: MCDP shows FCC # and TMC FAIL for computers that had a self-test failure when put into GRD TEST MODE. The MCDP shows FCC and TMC interface failures related to the MCP of FCCs and TMC that had a satisfactory self-test. The MCDP does a check to make sure the MCP program pins are connected correctly. Pins are read as a binary count and the decimal equivalent is shown on the MCDP. A manual test is then done to make sure switches, switch/lights, controls, and displays on the MCP operate correctly. It is necessary for a person to operate the controls.

S 862-050

(4) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

J. 05 TMSP

S 862-051

(1) Do the Prepare-to-Test steps.

S 742-052

(2) Do MCDP test 05.

NOTE: The MCDP shows TMC FAIL if TMC had a self-test failure when put into GRD TEST MODE. If TMC FAIL message does not show, MCDP shows TMC interface failures related to the TMSP. A manual test is done to make sure TMSP switches and temperature select control operate correctly. The switches to be pushed will show on the upper EICAS display during the 05 test. It is necessary for a person to operate the controls.

S 862-053
(3) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

K. 06 ASA

S 862-054
(1) Do the Prepare to Test steps.

S 742-055
(2) Do MCDP test 06.

NOTE: The MCDP shows FCC # FAIL for an FCC that had a self-test failure when put into GRD TEST MODE. The MCDP shows FCC interface failures related to the ASA. A person operates controls to do a check of the ASA reset function

S 862-056
(3) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

L. 07 SERVO AIL

S 862-057
(1) Do the Prepare to Test steps.

S 862-058

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(2) Do the deactivation procedure for the spoilers (AMM 27-61-00) or move all persons and equipment away from the spoilers.

S 862-059
(3) Do the Supply Hydraulic Power steps, when the MCDP message 07 VFY HYD ON - ADV is shown.

S 862-060
(4) Set the Flap handle to zero detent position.

S 742-062

(5) Do MCDP test 07.

NOTE: The MCDP shows FCC # FAIL for an FCC that had a self-test failure when put into GRD TEST MODE. The MCDP shows FCC interface failures related to aileron servos. The MCDP then tells the person to make sure hydraulic power is supplied. Other interaction messages tell the person to set aileron trim to zero, then engage the A/P in command. After servos engage, the MCDP tells the FCCs to move the servos to adjust the control wheel to center, rotate it clockwise, then counterclockwise, and back to center.

S 862-063

(6) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

M. 08 SERVO ELEV

S 862-433

(1) Do the Prepare to Test steps.

S 862-434

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(2) Do the deactivation procedure for the spoilers (AMM 27-61-00) or move all persons and equipment away from the spoilers.

S 862-064

(3) Do the Supply Hydraulic Power steps, when MCDP message 08 VFY HYD ON - ADV is shown.

S 862-065

(4) Set stabilizer to 6 units of trim.

S 742-067

(5) Do MCDP test 08.

NOTE: The MCDP shows FCC # FAIL for an FCC that had a self-test failure when put into GRD TEST MODE. The MCDP shows FCC interface failures related to elevator servos. The MCDP then tells the person to make sure hydraulic power is supplied. Other messages tell the person to set STAB TRIM to 6 units, set ELEV FEEL pitot-static pressure to zero. The person is then told to engage the A/P in command. After the servos engage, the MCDP gives the necessary parameters to the FCCs to drive the servos to adjust the control column to center, then move it forward, then move it aft, then back to center.

S 862-068

(6) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

N. 09 SERVO RUD

S 862-069

(1) Do the Prepare to Test steps.

S 862-070

WARNING: LOCK THE NOSE GEAR STEERING WHEN RUDDER MOVEMENT IS SCHEDULED TO PREVENT INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(2) Move the towing lever on nose gear metering valve module to the TOW position.

S 862-477

(3) Install the nose gear steering valve lockpin (AMM 09-11-00/201).

S 862-071

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(4) Do the deactivation procedure for the spoilers (AMM 27-61-00) or move all persons and equipment away from the spoilers.

S 862-072

(5) Do the Supply Hydraulic Power steps when the MCDP message 09 VFY HYD ON-ADV shows.

S 742-074
(6) Do MCDP test 09.

NOTE: The MCDP shows FCC # FAIL for an FCC that had a self-test failure when put into GRD TEST MODE. The MCDP shows FCC interface failures related to rudder servos. The MCDP then tells the person to make sure hydraulic power is supplied. Other messages tell the person to set rudder trim to zero. The MCDP gives the parameters to the FCCs to move the servos to set the rudder to center and then move it side to side. The person will monitor the movement of the rudder pedals.

S 862-075
(7) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

O. 10 SERVO A/T

S 862-435
(1) Connect External Electrical Power (AMM 24-22-00/201).

S 862-076

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(2) Do the deactivation procedure for the spoilers (AMM 27-61-00) or move all persons and equipment away from the spoilers.

S 862-483

CAUTION: MAKE SURE THE ENGINES ARE NOT IN OPERATION. THIS TEST INCLUDES AUTOMATIC MOVEMENT OF THE THRUST LEVERS AND COULD CAUSE AIRPLANE MOVEMENT IF THE ENGINES ARE IN OPERATION. INJURY TO PERSONS COULD OCCUR.

(3) Make sure the engines are not in operation.

S 862-077
(4) Do the Prepare to Test steps.

NOTE: This test does a check of the operation of the autothrottle servo loop which includes the servo, servo tachometer, and thrust lever angle transducers. This test cannot be done unless the two engines are installed (TLA interface).

S 862-081

(5) Set the left and right EEC MAINT ENG POWER (EEC GROUND TEST) switches on the P61 panel to the TEST position. This will supply power to the EEC.

NOTE: The failure to energize the EECs before the MCDP GRD TEST mode is set will cause NO INFC TMC EEC L PRIM (SEC) and NO INFC TMC EEC R PRIM (SEC) messages to be shown during the test.

S 862-095

(6) Make sure the Engine Control System is operational (AMM 76-11-00).

S 862-096

(7) Position A/T switch on MCP to OFF.

S 742-098

(8) Do MCDP test 10.

NOTE: The MCDP shows TMC FAIL for a TMC that had a self-test failure when put into GRD TEST MODE. If the TMC FAIL message does not show, MCDP shows TMC interface failures related to the EECs. The MCDP then tells the person to make sure that thrust levers and cables are clear, and then position the A/T on the MCP to ARM. Parameters sent to the TMC by the MCDP cause the throttles to start at the aft position. Then they are driven forward for 8 seconds to the forward position and back for 3 seconds.

S 862-112

(9) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

P. 11 SW A/P DISC

S 862-113

(1) Do the Prepare to Test steps.

S 742-114
(2) Do MCDP test 11.

NOTE: The MCDP shows FCC # FAIL for an FCC that had a self-test failure when put into GRD TEST MODE. The MCDP then shows FCC interface failures related to autopilot disconnect switch. The MCDP then tells the person to push and hold, then release, each (Capt, F/O) A/P disconnect switch. The switches are monitored for A/P disconnect command operation.

S 862-115
(3) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

Q. 12 SW A/T DISC

S 862-116
(1) Do the Prepare to Test steps.

S 742-117
(2) Do MCDP test 12.

NOTE: The MCDP shows TMC FAIL for a TMC that had a self-test failure when put into GRD TEST MODE. If the TMC FAIL message does not show, the MCDP shows TMC interface failures related to the autothrottle disconnect switch. The MCDP tells the person to push and hold, then release the two (L, R) autothrottle disconnect switches. The switches are monitored for autothrottle disconnect command operation.

S 862-118
(3) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

R. 13 SW G/A

S 862-119
(1) Do the Prepare to Test steps.

S 742-120
(2) Do MCDP test 13.

NOTE: The MCDP shows FCC # and TMC FAIL for an FCC and TMC that had a self-test failure when put into GRD TEST MODE. The MCDP shows FCC and TMC interface failures related to go-around switches. The MCDP then tells the person to push and hold, then release, the two (Capt, F/O) go-around switches. The G/A switch command operation is monitored by the FCCs and the TMC.

S 862-121
(3) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

S. 14 XDCR COL L

S 742-499
(1) This test is not used.

T. 15 XDCR COL R

S 742-503
(1) This test is not used.

U. 16 XDCR WHL

S 742-507
(1) This test is not used.

V. 17 PVD/PVDC

S 742-509
(1) This test is not used.

W. 30 CURRENT FAULT REPORT

S 862-436
(1) Align the Left, the Center, and the Right Inertial Reference Systems (AMM 34-21-00/201).

NOTE: It takes 10 minutes to align the IRS. Do the MCDP self-test during this time.

S 862-136

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(2) Do the deactivation procedure for the spoilers (AMM 27-61-00) or move all persons and equipment away from the spoilers.

S 842-487
(3) Make sure the engines are not in operation.

S 862-138
(4) Do the Prepare to Test steps.

S 862-474
(5) Set the A/T switch on the MCP to OFF.

S 862-174

(6) Set the left and right EEC MAINT ENG POWER (EEC GROUND TEST) switches on the P61 panel to the TEST position. This will supply power to the EEC.

NOTE: If you do not supply power to the EEC before you set the MCDP to the GROUND TEST MODE, these messages will show during the test:
- NO INFC TMC EEC L PRIM
- NO INFC TMC EEC L SEC
- NO INFC TMC EEC R PRIM
- NO INFC TMC EEC R SEC

NOTE: Because the engines are not in operation, these messages will show:

- CURRENT FAULT EEC L PRI
- CURRENT FAULT EEC L SEC
- CURRENT FAULT EEC R PRI
- CURRENT FAULT EEC R SEC

S 862-182

(7) Do MCDP test 30.

NOTE: The MCDP shows FCC # and TMC FAIL for each FCC and TMC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows FCC and TMC interface faults. The MCDP shows messages to make sure the thrust levers and cables are clear and then to set the A/T switch on the MCP to ARM. A check is done on the TMC interface with the A/T switch again and applicable faults show. All computers then transmit ground fault data to the MCDP to be shown. These fault messages will be the same as those that show as flight faults but do not include the Flight Deck Effect on the MCDP readout and the intermittent bit does not show in the diagnostic code.

(a) MCDP shows FCC (L, C, R) FAIL and TMC FAIL for each FCC and TMC that had a self-test failure when it went into GRD TEST MODE.
(b) MCDP shows 30 INFC FAULT?.
(c) Push YES/ADV to see interface faults or push NO/SKIP to go to 30 CURRENT OP FAULT?.
(d) The MCDP shows 30 CURRENT OP FAULT?.
(e) If a check of autopilot engage problems is to be done, do these steps before YES/ADV is pushed:
    1) Do the Supply Hydraulic Power steps. Supply hydraulic power without waiting for a VFY HYD ON message on the MCDP.
    2) Engage the applicable autopilot channel.
    3) Push the YES/ADV switch.

NOTE: The autopilot and autothrottle must be disengaged before YES/ADV is pushed to go to 30 CURRENT OP FAULT?. If the MCDP goes to CURRENT OP FAULT with the autopilot and/or autothrottle engaged, the MCDP will show to display SRVO ON - NO GRD TEST (FCC [L, C, R] or TMC).

If SRVO ON - NO GRD TEST shows, set FLT FAULTS MODE, disengage the autopilot and/or the autothrottle, and then set GRD TEST MODE.

S 862-183

(8) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are necessary.

NOTE: The MCDP must be put out of ground test mode before MCDP ground test 02 is done or MCDP ground test 30 is done again. If this is not done many "NO INFC TMC" messages can appear.

X. 40 AUTOLAND

S 862-185

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Do the deactivation procedure for the spoilers (AMM 27-61-00) or move all persons and equipment away from the spoilers.

S 862-186

CAUTION: MAKE SURE THE ENGINES ARE NOT IN OPERATION. THIS TEST INCLUDES AUTOMATIC MOVEMENT OF THE THRUST LEVERS AND COULD CAUSE AIRPLANE MOVEMENT IF THE ENGINES ARE IN OPERATION. INJURY TO PERSONS COULD OCCUR.

(2) Make sure the engines are not in operation.

S 862-187

(3) Do the Prepare to Test steps.

S 862-188

WARNING: LOCK THE NOSE GEAR STEERING WHEN RUDDER MOVEMENT IS SCHEDULED TO PREVENT INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(4) Move the towing lever on the nose gear metering valve module to the towing position.

S 862-189

(5) Install the nose gear steering valve lockpin (AMM 09-11-00/201).

S 862-190

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(6) Do the Supply Hydraulic Power steps. Supply the hydraulic power when MCDP message 40 VFY HYD ON - ADV shows.

NOTE: Because of the high electrical power necessary for the electric hydraulic pumps, do the ground test on one autopilot channel at a time. Ignore ground test fault messages that show for the autopilot channels without hydraulic pressure. If an autopilot channel fails the test with only one hydraulic system pressurized, then do the test again with a minimum of two hydraulic systems before a procedure is done to correct the failure.

S 862-191

(7) On the P10 panel, set the L and C STAB TRIM CUTOUT switches to NORM.

S 862-192

(8) On the MCP, set switches as follows:
   (a) Set the A/T ARM switch to ARM.
   (b) Set the DISENGAGE BAR to the UP position.

S 862-196

(9) Set the left and right EEC MAINT ENG POWER (EEC GROUND TEST) switches on the P61 panel to the TEST position. This will supply power to the EEC.

S 742-484

(10) Do MCDP test 40.

NOTE: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when it went into GRD TEST MODE. The MCDP shows FCC interface faults related to autoland. The ILS Tune and RA Test inhibit functions are tested for a "not inhibited" condition and for an "inhibited" condition when inhibit is asked for. Fault messages for either failed condition will show. If the FCC R/MCDP interface is inactive, the MCDP shows that the bus isolation test cannot be done. The MCDP then tells the person to make sure the standby power is in the AUTO position, the battery switch is on, and the left bus tie is in the AUTO position. An advisory message shows that the EICAS display will not be on during the bus isolation test and the person pushes the YES/ADV switch. The remaining tests use parts of other ground tests to do a check of the autoland related functions. Autoland autothrottle test is done after the autoland servo test. After the start of the autoland autothrottle test, the MCDP shows TMC FAIL if the TMC had a self-test failure when it went into GRD TEST MODE. If no TMC FAIL message shows, the MCDP shows all TMC interface faults. The MCDP then does the autothrottle servo test and the go-around switch test.

NOTE: AIRPLANES WITH -133 FCCs AND AN OPTION GROUP 1 VALUE OF 6; If the RUD SERVO X FAIL (X = L, C, R) message shows intermittenly when you do this ground test, ignore this message unless the RUD SERVO X FAIL (X = L, C, R) message is shown in the last flight faults or previous flight faults with the diagnostic code 206.

S 862-184

(11) Do these steps to clear the EICAS maintenance messages(s) FLT CONT ELEC, STAB TRIM, and/or YAW DAMPER that can show after the Autoland Bus Isolaton Test is done:

WARNING: STAY AWAY FROM THE SPOILERS WHEN THESE CIRCUIT BREAKERS ARE OPENED. ACCIDENTAL SPOILER MOVEMENT CAN OCCUR WHEN THESE CIRCUIT BREAKERS ARE OPENED. INJURY TO PERSONS CAN OCCUR.

(a) Open these circuit breakers on the P11 panel then close them in less than 25 seconds:
   1) 11C6, FLT CONT ELEC 1L AC

2) 11C8, FLT CONT ELEC 2L AC
3) 11G17, FLT CONT ELEC 1R AC
4) 11G26, FLT CONT ELEC 2R AC

(b) If a faultball stays set on a Stabilizer Trim/Aileron Lockout Module (SAM) and/or a Yaw Damper Module (YDM) after the above circuit breakers have been opened and closed, push the reset button on the front of the applicable SAM or YDM.

S 862-475

(12) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are necessary.

Y. 51 AIR/GRD RLY

S 862-438

(1) Do the Prepare to Test steps.

S 862-437

WARNING: PREPARE THE SAFETY-SENSITIVE SYSTEMS FOR THE AIR MODE BEFORE YOU OPEN THE AIR/GROUND CIRCUIT BREAKERS. IN THE AIR MODE, MANY OF THE AIRPLANE SYSTEMS CAN OPERATE AND CAUSE INJURIES TO PERSONS AND DAMAGE TO EQUIPMENT.

(2) Prepare the safety-sensitive systems for air mode simulation (AMM 32-09-02/201).

S 742-205

(3) Do MCDP test 51.

NOTE: The MCDP shows FCC # and TMC FAIL for each FCC and TMC that had a self-test failure when it went into GRD TEST MODE. The MCDP then tells a person to open and close the air/ground circuit breakers to do a check of the FCC and TMC reply to air/ground relay operation. MCDP messages C/B LAND GEAR SYST 1 and C/B LAND GEAR SYST 2 refer to 11U15, AIR/GND SYS 1, 11C29, POSITION AIR/GND SYS 2 ALT, and 11U23 or 11U24, POSITION AIR/GND SYS 2 C/Bs on the P11 panel.

S 862-482
(4) Put the safety-sensitive systems back to their initial conditions (AMM 32-09-02/201).

S 862-207
(5) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are necessary.

Z. 52 TMC RLY/SW

S 862-439
(1) Do the Prepare to Test steps.

S 862-208

WARNING: MAKE SURE THE AREA AFT OF THE THRUST REVERSER IS CLEAR OF PERSONS AND EQUIPMENT BEFORE THRUST REVERSER OPERATION. THE THRUST REVERSER WILL EXTEND WHEN THE REVERSE THRUST LEVERS ARE MOVED AFT TO THE REVERSE IDLE POSITION. THIS CAN CAUSE INJURY TO PERSONS OR DAMAGE TO THE AIRPLANE.

(2) Do this procedure to prepare the thrust reverser system for operation: THRUST REVERSER OPERATION - DEPLOY/STOW (POWER METHOD) (AMM 78-31-00).

S 862-485

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(3) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoilers.

S 862-209
(4) On the P5 panel, set switches as follows:
(a) Set the L and R PACK switch/lights to OFF.
(b) Set the ANTI-ICE WING, L and R ENGINE switch/light to OFF.

(c) Set the BLEED AIR PANEL ISOLATION switch/light open (switch pushed in).
(d) Set the BLEED AIR PANEL L ENG and R ENG switch/lights to OFF.

S 862-210
(5) Make sure the reverse thrust levers are not set for operation (stowed position).

S 742-211
(6) Do MCDP test 52.

NOTE: The MCDP shows TMC FAIL if the TMC had a self-test failure when it went into GRD TEST MODE. If no TMC FAIL message shows, the MCDP shows TMC interface faults related to the ECS relay. This test makes sure all TMC analog input discrete signal paths are OK. The MCDP tells a person to operate controls during this test.

S 862-219
(7) Remove pneumatic power if it is not necessary.

S 862-220
(8) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

AA. 56 FCC CONFIG/OPT

S 742-335
(1) Do the Prepare to Test steps.

S 742-221
(2) Do MCDP test 56.

NOTE: The MCDP shows FCC # FAIL for an FCC that had a self-test failure when put into GRD TEST MODE. The MCDP shows FCC interface failures related to parity. A check is done to make sure the FCC program pin wiring has the correct configuration and customer options. The pins are read as a binary count and the decimal equivalent shows on the MCDP. All three channels show at the same time.

S 862-222
(3) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

AB. 57 TMC CONFIG/OPT

S 742-336
(1) Do the Prepare to Test steps.

S 742-223
(2) Do the MCDP test 57.

NOTE: The MCDP shows TMC FAIL if the TMC had a self-test failure when put into GRD TEST MODE. The MCDP shows TMC interface failures related to parity. A check is done to make sure the TMC program pin wiring has the correct configuration and customer options. The pins are read as a binary count and the decimal equivalent shows on the MCDP.

S 862-224
(3) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

AC. 59 FCC INSTR

S 862-225
(1) Do the Prepare to Test steps.

S 742-327
(2) Do MCDP test 59.

NOTE: The MCDP shows FCC # FAIL for an FCC that had a self-test failure when put into GRD TEST MODE. Tests of the operation of the Captain's and First Officer's flight director switches, and instrument source select switches are done.

S 862-227

(3) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

AD. 60 TMC INSTR

S 862-228

(1) Do the Prepare to Test steps.

S 862-229

(2) Make sure the thrust reference control knob is pushed in.

S 742-230

(3) Do MCDP test 60.

NOTE: The MCDP shows TMC FAIL if the TMC had a self-test failure when put into GRD TEST MODE. The MCDP sends test parameters to the TMC to control the Captain's and First Officer's EPR/N1 instruments target index.

S 862-236

(4) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

AE. 64 SPD BK/FLAP XDCR

S 862-237

(1) Do the Prepare to Test steps.

S 862-238

WARNING: DO NOT SUPPLY HYDRAULIC POWER DURING SPEEDBRAKE TEST. THE SPEEDBRAKES WILL MOVE IF HYDRAULIC POWER IS SUPPLIED.

(2) If the speedbrake handle is put in the ARM position, the hydraulics are on, and throttle levers set at idle, the speedbrakes will deploy and the speedbrake handle will move to UP position. This will cause MCDP message 64 SPD BK ARM FCC (L,C,R) FAIL message to show.

S 862-240

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(3) Do the deactivation procedure for the spoilers (AMM 27-61-00) or move all persons and equipment away from the spoilers.

S 862-241

(4) Do the Supply Hydraulic Power steps, when the MCDP message 64 VFY HYD is shown.

NOTE: Because of the high electrical power necessary for the electrical hydraulic pumps, do the ground test on one autopilot channel at a time. Ignore ground test fault messages that show for the autopilot channels without hydraulic pressure. If an autopilot channel fails the test with only one hydraulic system pressurized, then do the test again with a minimum of two hydraulic systems before a procedure is done to correct the failure.

S 862-486

WARNING: MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS MOVE TO THEIR COMMANDED POSITION IN LESS THAN 1 SECOND AND CAUSE INJURY TO PERSONS AND DAMAGE TO EQUIPMENT

(5) Move all persons and equipment away from the spoilers.

S 742-328

(6) Do MCDP test 64.

NOTE: The MCDP shows FCC # and TMC FAIL for an FCC and TMC that had a self-test failure when put into GRD TEST MODE. The MCDP shows FCC and TMC interface failures related to the slat switch and flap transducer. The MCDP then tells the person to make sure hydraulic power is supplied. Tests of speedbrake and flap lever are done when the MCDP tells the person to set the handle or lever to a specified position. The MCDP then does a check for proper position.

S 862-244
(7) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

AF. 65 STAB TRIM

S 862-245
(1) Do the Prepare to Test steps.

S 862-246

<u>WARNING</u>: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(2) Do the deactivation procedure for the spoilers (AMM 27-61-00) or move all persons and equipment away from the spoilers.

S 862-247
(3) Do the Supply Hydraulic Power steps when the MCDP message 65 VFY HYD ON - ADV is shown.

<u>NOTE</u>: Because of the high electrical power necessary for the electrical hydraulic pumps, do the ground test on one autopilot channel at a time. Ignore ground test fault messages that show for the autopilot channels without hydraulic pressure. If an autopilot channel fails the test with only one hydraulic system pressurized, then do the test again with a minimum of two hydraulic systems before a procedure is done to correct the failure.

S 742-249
(4) Do MCDP test 65.

<u>NOTE</u>: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when put into GRD TEST MODE. The MCDP shows FCC interface failures related to stab trim. The person is told to make sure the SAM is on and the hydraulic power supplied, and then the A/P engaged in command. The MCDP then sends parameters to the FCCs which tell the Stabilizer Trim and Aileron Lockout Module (SAM) to move the stabilizer up and down. One FCC is tested at a time.

S 862-250
(5) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

AG. 66 XDCR OUTPUTS

S 862-251

(1) Do the steps shown to prepare to test each of these components.
   (a) Aileron Autopilot Servos
      1) Set aileron trim to 0 units of trim.
      2) Set flaps to 0
   (b) Elevator Autopilot Servo
      1) Set stabilizer to 6 units of trim.
      2) Pressurize the elevator feel pitot system to 350 knots (34-11-00).
   (c) Rudder Autopilot Servo
      1) Make sure the test switches on the L and R Rudder Ratio Modules in the Main E/E Compt are set to NORM (AMM 27-21-00).
      2) Set rudder trim to 0 units of trim.
      3) Make sure the rudder is at neutral (AMM 27-21-00).

S 862-252

(2) Do the Prepare to Test steps.

S 862-253

WARNING: LOCK THE NOSE GEAR STEERING WHEN ANY RUDDER MOVEMENT IS SCHEDULED TO PREVENT INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(3) Move towing lever on nose gear metering valve module to the TOW POSITION.

S 492-254

(4) Install nose gear steering valve lockpin (AMM 09-11-00/201).

S 862-255

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(5) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoilers.

S 862-256

(6) Do the Supply Hydraulic Power steps.

NOTE: Because of the high electrical power necessary for the electrical hydraulic pumps, do the ground test on one autopilot channel at a time. Ignore ground test fault messages that show for the autopilot channels without hydraulic pressure. If an autopilot channel fails the test with only one hydraulic system pressurized, then do the test again with a minimum of two hydraulic systems before a procedure is done to correct the failure.

S 742-275

(7) Do MCDP test 66.

NOTE: The MCDP shows FCC # and TMC FAIL for a FCC and TMC that had a self-test failure when put into GRD TEST MODE. The MCDP shows FCC and TMC interface failures related to the transducers. The MCDP does tests and shows the position of the horizontal stabilizer, aileron servo, aileron surface, elevator servo, elevator surface, rudder servo, rudder surface and flaps. Test 66 also does a test and shows the percentage of the speedbrake handle. See Fig. 202 for the rigging parameters.

S 862-277

(8) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

AH. 67 AIL SURF LIM

S 862-278

(1) Do the Prepare to Test steps.

S 862-279

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(2) Do the deactivation procedure for the spoilers (AMM 27-61-00) or move all persons and equipment away from the spoilers.

Not Used
Figure 201

MCDP DISPLAY	RIG VALUE	MAX CH TRACKING DIFFERENCE	
		NULL	MAX INPUT
66 STAB DEG/LCR (STAB TRIM AT 6 UNITS)	-5 ±0.3° [3] -4 ±0.3° [4]	0.3°	0.6°
66 AIL SRVO DEG	[1]		
66 AIL SURF DEG	0 ±0.5°	0.6°	1.2°
66 ELEV SRVO DEG	[1]		
66 ELEV SURF DEG (STAB TRIM AT 6 UNITS)	0 ±0.2° [2]	0.4°	0.8°
66 RUD SRVO DEG	[1]		
66 RUD SURF DEG	0 ±0.6°	1.2°	2.4°
66 SPD BK PCT (ARM DETENT)	15.4 ±5%	N/A	N/A
66 FLAP DEG/FCC 66 FLAP DEG/TMC } FLAPS AT 1 DETENT AND SPD BK HANDLE AT FULL DOWN	0 ±2°	0.6°	1.2°

[1] THE SERVO POSITIONS WERE SET BY THE CENTERING SPRINGS THE LAST TIME HYDRAULIC POWER WAS SET TO OFF. NO TOLERANCES ARE SPECIFIED. ONLY USE THE DATA TO MAKE SURE THE SERVO IS ALMOST IN THE CENTER AND IS NOT AT THE POSITIVE OR NEGATIVE LIMIT.

[2] IF THE NULL (RIG) VALUE IS NOT IN TOLERANCE, PRESSURIZE THE ELEVATOR FEEL PITOT SYSTEM TO 350 KNOTS (REF 34-11-00).

[3] SAS 050-149

[4] ALL MTH AIRPLANES, SAS 150-999

Rigging Tolerances and Position Limits - Tests 66,67,68,69
Figure 202 (Sheet 1)

MCDP DISPLAY	POSITIVE LIMIT	NEGATIVE LIMIT
67 AIL SURF DEG (SINGLE CHANNEL ENGAGED)	+15 ±2°	−15 ±2°
67 AIL SURF DEG (DUAL OR TRIPLE CHANNEL ENGAGED)	20° MINIMUM	−20° MINIMUM
68 ELEV SURF DEG (SINGLE CHANNEL ENGAGED)	+7.5° MINIMUM	−7.5° MINIMUM
68 ELEV SURF DEG (SINGLE CHANNEL ENGAGED WITH ELEVATOR FEEL PITOT SYSTEM PRESSURIZED TO 190± 2 KNOTS)(REF 34-11-00/201)	+5.2 TO 8.8°	−2.7 TO −8.9°
68 ELEV SURF DEG (DUAL OR TRIPLE CHANNEL ENGAGED)	18° MINIMUM	−25° MINIMUM
69 RUD SURF DEG (DUAL OR TRIPLE CHANNEL ENGAGED)	22.8° MINIMUM	−22.8° MINIMUM

Rigging Tolerances and Position Limits - Tests 66,67,68,69
Figure 202 (Sheet 2)

S 862-280

(3) Do the Supply Hydraulic Power steps when the MCDP message 67 VFY HYD ON - ADV is shown.

NOTE: Because of the high electrical power necessary for the electrical hydraulic pumps, do the ground test on one autopilot channel at a time. Ignore ground test fault messages that show for the autopilot channels without hydraulic pressure. If an autopilot channel fails the test with only one hydraulic system pressurized, then do the test again with a minimum of two hydraulic systems before a procedure is done to correct the failure.

S 742-282

(4) Do MCDP test 67.

NOTE: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when put into GRD TEST MODE. The MCDP shows FCC interface failures related to the aileron autopilot servos. The MCDP tells the person to make sure hydraulic power is supplied. The MCDP tell the person to set the aileron trim to zero and engage all three A/P channels in command. The MCDP then moves the aileron servo to the positive limit and shows the aileron surface position in degrees for each channel. The same check is done for the negative limit. This does a test of the autopilot aileron servo limits. See Fig. 202 for limit parameters.

S 862-283

(5) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

AI. 68 ELEV SURF LIM

S 862-284

(1) Do the Prepare to Test steps.

S 862-285

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(2) Do the deactivation procedure for the spoilers (AMM 27-61-00) or move all persons and equipment away from the spoilers.

S 862-286

(3) Do the Supply Hydraulic Power steps when MCDP message 68 VFY HYD ON - ADV is shown.

NOTE: Because of the high electrical power necessary for the electrical hydraulic pumps, do the ground test on one autopilot channel at a time. Ignore ground test fault messages that show for the autopilot channels without hydraulic pressure. If an autopilot channel fails the test with only one hydraulic system pressurized, then do the test again with a minimum of two hydraulic systems before a procedure is done to correct the failure.

S 742-288

(4) Do MCDP test 68.

NOTE: The MCDP shows FCC # FAIL for a FCC that had a self-test failure when put into GRD TEST MODE. The MCDP shows FCC interface failures related to elevator autopilot servos. The MCDP then tells the person to make sure hydraulic power is supplied. The MCDP tells the person to set the stab aileron trim to 6 units and engage all three A/P channels in command (you can do MCDP test 68 with one, two, or all three A/P channels in command) . The MCDP then moves the elevator servo to the positive limit and shows the elevator surface position in degrees for each channel. The same check is done for the negative limit. This does a test of the elevator autopilot servo limits. See Fig. 202 for limit parameters.

S 862-329

(5) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

AJ. 69 RUD SURF LIM

S 862-289

(1) Make sure the test switches on the L and R Rudder Ratio Modules in the Main E/E Compt are set to NORM (AMM 27-21-00).

S 862-290

(2) Set rudder trim to 0 units of trim.

S 862-291

(3) Make sure the rudder is at neutral (AMM 27-21-00).

NOTE: The Rudder Ratio Changer must be in the Low Speed position and the rudder in neutral for this test. The rudder is at neutral when the rudder trailing edge is aligned within the rudder index plate groove.

S 862-292

(4) Do the Prepare to Test steps.

S 862-293

WARNING: LOCK THE NOSE GEAR STEERING WHEN ANY MOVEMENT IS SCHEDULED TO PREVENT INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(5) Move the towing lever on nose gear metering valve module to the TOW POSITION.

S 492-294

(6) Install the nose gear steering valve lockpin (AMM 09-11-00/201).

S 862-295

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(7) Do the deactivation procedure for the spoilers (AMM 27-61-00) or move all persons and equipment away from the spoilers.

S 862-296

(8) Do the Supply Hydraulic Power steps when MCDP message 69 VFY HYD ON-ADV is shown.

NOTE: Because of the high electrical power necessary for the electrical hydraulic pumps, do the ground test on one autopilot channel at a time. Ignore ground test fault messages that show for the autopilot channels without hydraulic pressure. If an autopilot channel fails the test with only one hydraulic system pressurized, then do the test again with a minimum of two hydraulic systems before a procedure is done to correct the failure.

S 742-298

(9) Do MCDP test 69.

NOTE: Th MCDP shows FCC # FAIL for an FCC that had a self-test failure when put into GRD TEST MODE. The MCDP shows FCC interface failures related to rudder autopilot servos. The MCDP then tells the person to make sure hydraulic power is supplied. The MCDP tells the person to set the rudder trim to zero and engage all three A/P channels in command. The MCDP then moves the rudder servo to the positive limit and shows the rudder surface position in degrees for each channel. The same check is done for the negative limit. This does a test of the autopilot rudder servo limits. At least two autopilot channels must be in operation for this test. See Fig. 202 for limit parameters.

S 862-299

(10) Do the Put the Airplane Back to Its Usual Condition steps if no other ground tests are to be done.

AK. Put the Airplane Back to Its Usual Condition

S 862-300

WARNING: STAY AWAY FROM THE NOSE GEAR WHEELS WHEN THE LOCKPIN IS REMOVED THE NOSE WHEEL CAN MOVE QUICKLY TO THE CENTER POSITION. THIS CAN CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Make sure the nose gear wheels are centered.

S 032-301

(2) Remove the nose gear steering valve lockpin.

S 752-481

(3) Make sure the towing lever goes to the NORMAL POSITION.

S 862-480

(4) Set the MCDP to OFF.

S 862-392

(5) Set the left and right MAINT ENG POWER (EEC GROUND TEST) switches on the P61 panel to the NORM position.

S 862-313

(6) Remove the remote MCDP control unit if it is installed.

S 862-320

(7) Remove hydraulic power if it is not necessary (AMM 29-11-00/201).

S 862-321

(8) Remove electrical power if it is not necessary (AMM 24-22-00/201).

AUTOPILOT (FLIGHT CONTROL) - DESCRIPTION AND OPERATION

1. General
   A. The autopilot (A/P) provides automatic control of the roll, pitch, and yaw axes through all phases of flight except takeoff (T/O). The Flight Director (F/D) provides guidance commands for all phases of flight including takeoff using manual control. The AFDS provides display data to the Electronic Horizontal Situation Indicators (EHSI) and Electronic Attitude Director Indicators (EADI) of the Electronic Flight Instrument System (Ref 34-22-00).
   B. Configuration
      (1) MCP

	-206	-221
ALL	BASIC	SB 22A92

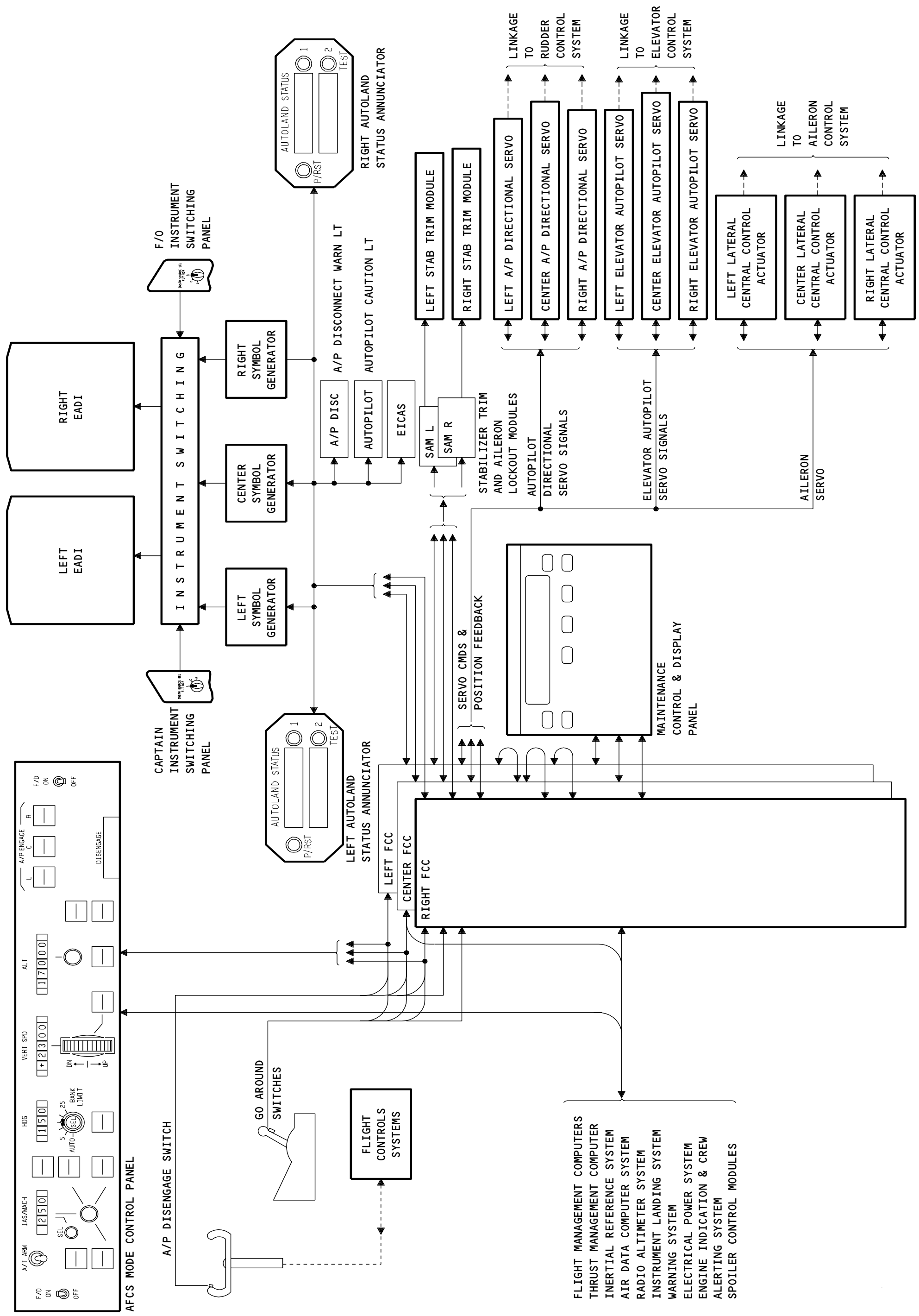
      (2) FCC

	-108	-109	-133	-135	-136
SAS 050,051, 150-155, 162-165	BASIC	SB 22-3			
SAS 156,157, 166,167		BASIC			
SAS 052-099, 158-161, 168-999					BASIC
MTH 275,276	BASIC	SB 22-38			
MTH 277-280, 282-999			BASIC		
MTH 281				BASIC	
MTH 282-999					BASIC

   C. The Autopilot (Flight Control) provides an overview of the Autopilot/Flight Director System. Additional details of the Autopilot/Flight Director System are provided by the following sections:
      (1) Autopilot/Flight Director Power (MM 22-11-00) covers the component detail and operation of the FCC's, MCP, Go-Around Switches, and Autopilot Disengage Switches. It provides details on the autopilot system interfaces and BITE.
      (2) Autopilot/Flight Director Pitch Channel (MM 22-12-00) covers the component detail and operation of the Elevator Autopilot Servos. It provides details on the pitch channel Arm and Engage Logic and signal flow.

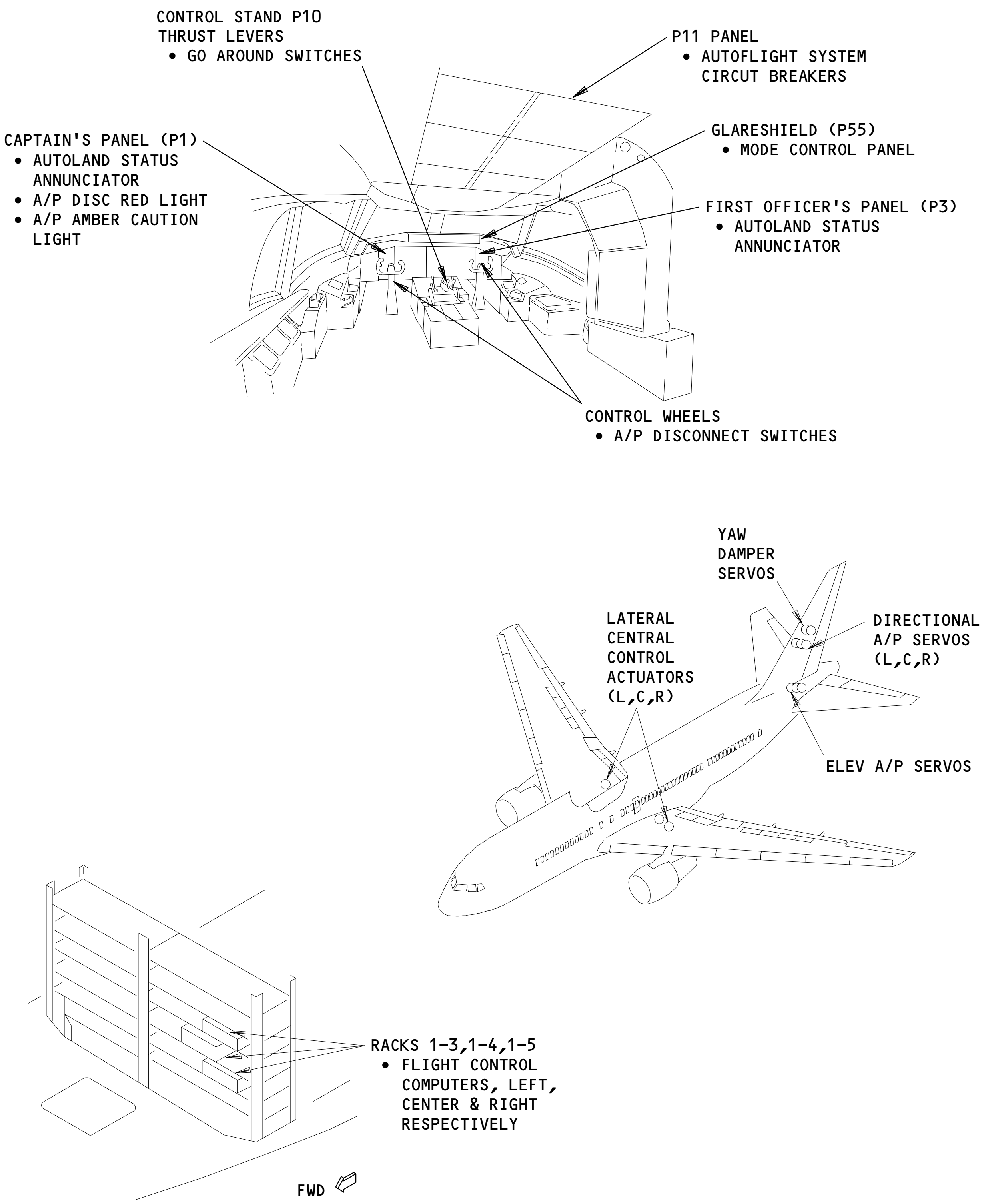
(3) Autopilot/Flight Director Roll and Yaw Channel (MM 22-13-00) covers the component detail and operation of the Lateral Central Control Actuator and Directional Autopilot Servo. It provides details on the roll and yaw channel Arm and Engage Logic and signal flow.
(4) Autopilot/Flight Director Warning and Annunciation (AMM 22-14-00) covers the component detail of the Autoland Status Annunciators, Autopilot CAUTION Light, and A/P DISC light. It provides details on the Warning and annunciation logic and signal flow.
(5) Autopilot/Flight Director Interchannel Data (AMM 22-15-00) covers the exchange of data between Flight Control Computers on the ARINC 429 Data Bus and Analog Discretes.

D. Components (Fig. 1)
(1) The AFDS has three digital Flight Control Computers (FCCs) and one Autoflight Control Systems (AFCS) Mode Control Panel (MCP). The system is capable of providing autopilot/flight director guidance and control for Autoland Category III operations.
(2) The left, center, and right (L, C, and R) FCCs contain all logic and signal handling circuitry for pitch, roll and yaw axis control. Yaw control is only during a multichannel autoland.
(3) Each FCC provides Autopilot (A/P) command output signals which drive hydraulic servos that in turn provide commands to the flight control actuators. Each FCC also provides Flight Director (F/D) output signals for steering commands and mode annunciations displayed on the Electronic Attitude Director Indicators (EADIS).
(4) The digital interface between the AFDS and associated sensor systems use the ARINC 429 Digital Information Transfer System (DITS).
(5) The Autopilot/Flight Director System (Fig. 2) consists of the following components:
(a) Mode Control Panel - located on the center glareshield, P55 panel
(b) Autoland Status Annunciator - Captain's P1-3 panel and First Officer's P3-1 panel.
(c) A/P DISC red warning light - Captain's panel, P1-3.
(d) AUTOPILOT amber caution light - Captain's panel P1-3.
(e) A/P disengage switches - Inboard side of outboard horn on each control wheel.
(f) AFDS go-around switches - One switch on each thrust lever for each FCC.
(g) F/D source select switches - Captain's P1-1 panel; First Officer's P3-3 panel.
(h) FCCs (3) - on main Electrical/Electronics (E/E) racks (E1-3, E1-4, and E1-5).
(i) MCDP - on main E/E racks (E1-2).
(j) SAM (2) modules - on main E/E racks (E1-1 and E2-1).
(k) Lateral Central Control Actuators (LCCA) - Left and Right in the left wing root, Center in the right wing root.



Autopilot/Flight Director System
Figure 1

586613



Autopilot/Flight Director System - Component Locations
Figure 2

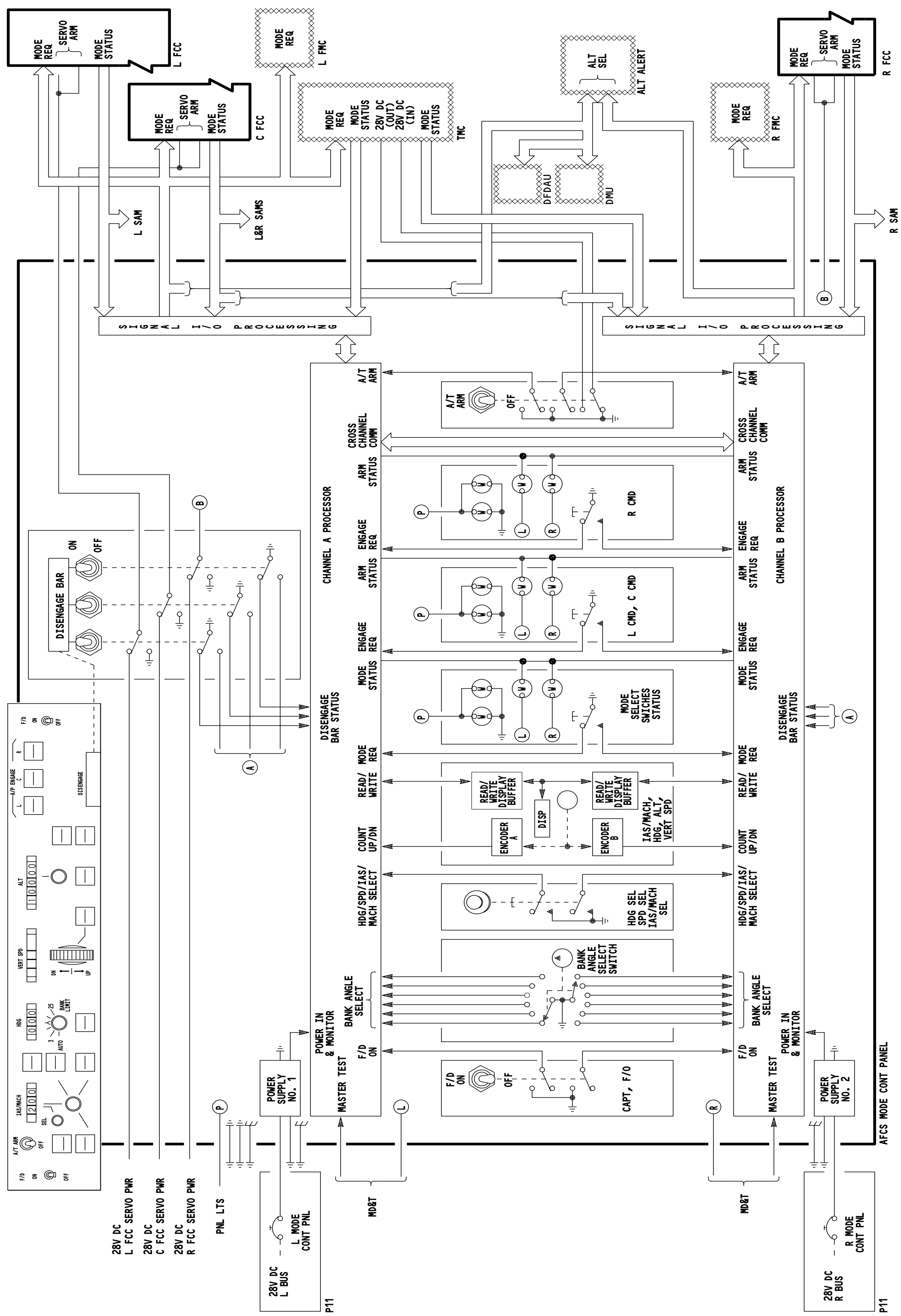
(l) Elevator Autopilot Servos (EAS) - Left, Center, and Right located aft of the horizontal stabilizer.
(m) Directional Autopilot servos (DAS) - Left, Center, and Right in the lower area of the vertical stabilizer.

E. Indications
(1) The Autopilot/Flight Director System condition and status is indicated by the following:
(a) The mode select switch/lights on the MCP have identifying legends in the upper half of the switch cover. The legends light glareshield instrument panel lights are on. A dot-bar matrix on the lower half lights when the FCC has accepted the associated mode request. Digital displays on the MCP indicate airspeed, vertical speed, heading, and altitude. Panel legends identify the status of the F/D, auto throttle (A/T), and bank angle (B/A) limit switches.
(b) The Electronic Attitude Director Indicator (EADI) provides F/D displays and Flight Mode Annunciation (FMA) of roll and pitch modes which are armed or engaged.
(c) A red autopilot disconnect (A/P DISC) light and an amber AUTOPILOT caution light are on the Captain's instrument panel (P1-3).
(d) The master CAUTION annunciators and aural are controlled by the EICAS (AMM 31-41-00/501). The master WARNING annunciators and aural are controlled by the Warning System (AMM 31-51-00/501).

F. Mode Control Panel/Flight Control Computer Interfaces Schematic (Fig. 3)
(1) Autopilot (A/P) and flight director (F/D) engage requests are generated at the Mode Control Panel (MCP) and are supplied to the Flight Control Computers (FCCs). Once accepted, the requests can result in the arming and engaging of the A/P servos.
(2) Mode requests for both A/P and F/D are generated at the MCP and transmitted by digital data bus to the FCCs.
(3) Flight director engage requests will be automatically generated by the FCC's in the go-around mode. This option supports windshear capabilities and is enabled by grounding the FCC program pin F/D AUTOMATIC ON.
(4) The A/P disengage bar logic tells the FCC if the bar is up or down. This allows the FCC to enable or disable the A/P servos.
(5) The FCCs provide engage and mode status to the MCP. Mode status is indicated by the dot-bar matrix on the A/P engage and mode select switch/lights. Mode status is also indicated on the EADIs (AMM 22-14-00).

G. FCC Input Discretes From Remote Switches
(1) Two A/P disengage switches (one on each control wheel) provide discrete signals to the FCCs which disengage the autopilot. The go-around switches (one on each thrust lever) provide discretes to the FCCs to start the Go-Around (GA) mode.

H. AFDS Input Power
(1) The FCC's use 115v ac 400 Hz for power. The 115v ac is also reduced to 26v ac for excitation power to the autopilot servos position LVDT's. The FCC's receive 28v dc power which is used for the Servo ARM and ENGAGE Solenoids.



Autopilot/Flight Director System Mode Control Panel Interfaces Schematic
Figure 3

586624

(2) The MCP is powered by two separate 28 vdc power sources.

I. Autopilot/Flight Director System Mode Control Panel Operation

(1) MCP Functional Partitioning

(a) The MCP contains channels A and B which use a common display element. This prevents a single failure from causing a total loss of the MCP function. Each channel receives inputs from panel switches and encoders and processes the data independently.

(b) The A channel receives ARINC 429 data from the L and C FCCs, L FMC, and TMC. Channel A transmits ARINC 429 data to the L and C FCCs, L FMC, TMC, and pilot's instruments. The B channel receives ARINC 429 data from the R and C FCC, R FMC, and TMC. Channel B transmits ARINC 429 data to the R FCC, R FMC, and pilot's instruments.

(c) In the event of a failure of one MCP channel, the output bus from that side switches to the other channel and continues to provide all ARINC 429 outputs.

(2) MCP Autopilot Engage Switch/lights

(a) Pressing any one of the CMD switch/lights (L, C, or R) generates an engage request signal. The L and C switch/light requests are processed by channel A, and the R request is processed by channel B. Illumination of the dot-bar matrices on the switch/lights is controlled by status logic generated in the appropriate FCC. The matrix lights when the request is accepted.

(b) The dot-bar matrix lights on the mode select switch/lights are checked by the Master Dim and Test System (AMM 33-16-00/501).

(3) MCP Autopilot Mode Select Switch/lights

(a) The mode select switch/lights provide mode request signals to both A and B channels of the MCP. The request also sets a bit in a digital word which is transmitted to the FCC or TMC. The illumination of the dot-bar matrices of these switches is controlled by mode status logic generated in the engaged FCC(s) or TMC. No illumination is provided for the IAS/MACH SEL switch or the HDG SEL switch.

(4) MCP Autopilot Reference Data Controls and Displays

(a) Reference values for altitude, heading, vertical speed, and airspeed or mach number are entered using the related select knob or thumbwheel on the MCP. These reference values are optically encoded and supplied to both the A and B channels and routed to the appropriate MCP window for display. The reference values are also sent to the FCC, FMC, and TMC.

(5) Vertical Speed Mode

(a) When the vertical speed (V/S) mode select switch/light is pressed, a V/S mode request discrete is initiated. This discrete generates a manual ON signal which arms the vertical speed mode if the FCC is engaged in CMD or the F/D is on and the airplane is airborne. The vertical speed mode is automatically initiated by an engaged FCC (CMD or F/D) if no other pitch modes are selected or engaged (AMM 22-12-00).

(b) The dot-bar matrix annunciator in the V/S switch/light illuminates when the MCP receives V/S OPER from the FCC's. V/S OPER is also sent to the EFIS symbol generators for mode display on the EADIs. Once V/S mode is engaged, it can be disabled if CMD is disabled and the two F/Ds are turned off. The V/S mode will also be disabled by the selection of the following pitch modes: ALT HLD select, FLCH select, VNAV select, APP select, or GA select.

(c) The V/S pushbutton can also be illuminated using the TEST switch on the master dim and test panel. This test input checks MCP circuits (data management) as well as the dot-bar matrices.

(6) Heading Hold Mode

(a) Pressing the HDG HOLD mode select switch/light initiates a heading hold mode request discrete. This discrete generates a manual ON signal which arms the heading hold mode if the FCC is engaged in CMD or the F/D is on and the airplane is airborne. The heading hold mode is automatically initiated by an engaged FCC (CMD or F/D) if no other roll modes are selected or engaged (AMM 22-13-00).

(b) The dot-bar matrix annunciator in the HDG HOLD switch/light lights when the MCP receives HDG HOLD OPER from the FCC's. HDG HOLD OPER is also sent to the EFIS symbol generators for mode display on the EADIs. Once HDG HOLD mode is engaged, it can be disabled if CMD is disabled and the two F/Ds are turned off. Heading hold is also disabled by selection of the following modes: HDG select, LNAV select, LOC override, or GA select.

(7) Altitude Hold Mode

(a) Pressing the ALT HOLD mode select switch/light initiates an altitude hold mode request discrete. This discrete generates a manual ON signal which arms the altitude hold mode if the FCC is engaged in CMD or the F/D is on and the airplane is airborne. Altitude hold can also be automatically initiated by an engaged FCC (CMD or F/D) during altitude capture (Ref 22-12-00).

(b) The dot-bar matrix in the ALT HOLD switch/light lights when the MCP receives ALT MODE OPER from the FCC(s). ALT HOLD OPER is also sent to the EFIS symbol generators for mode display on the EADIs. Once altitude hold mode is engaged, it can be disabled if CMD is disabled and the two F/Ds are turned off. Altitude hold will also be disabled by selection of the following modes: V/S select, FLCH select, VNAV select, APP select, or GA select.

(8) Thrust Mode

(a) Pressing the EPR (or N1, or THR) mode select switch/light initiates a thrust mode request discrete. This discrete results in a thrust mode select signal if the autothrottle (A/T) is armed and operating with no faults while the airplane is either on the ground, in the air, or in the takeoff mode.

(b) The dot-bar matrix in the thrust mode select switch/light lights when the MCP receives THRUST MODE OPER from the TMC. Once the thrust mode is engaged, it can be disabled if the A/T is disengaged or another mode is selected (SPD select, FLCH select, or VNAV select).

(9) Speed Mode

(a) Pressing SPD switch/light initiates a speed mode request discrete. This discrete generates a speed mode select signal if the A/T is enabled and the airplane is airborne. The speed mode can also be generated by FLCH mode or VNAV mode disengagement.

(b) The dot-bar matrix in the SPD switch/light lights when the MCP receives SPEED MODE OPER from the TMC. Once SPEED MODE is initiated, it can be disabled if the A/T is disabled, A/T retard engaged, or GA mode selected.

(10) Flight Level Change Mode

(a) Pressing the Flight Level Change (FLCH) mode switch/light initiates a flight level change discrete. This discrete generates a flight level change select signal if the FCC is engaged in CMD or the F/D is on. The FLCH select signal arms the flight level change mode.

(b) The dot-bar matrix in the FLCH switch/light lights when the MCP receives FLIGHT LEVEL CHANGE OPER from the FCCs. FLIGHT LEVEL CHANGE OPER is also sent to the EFIS symbol generators for mode display on the EADIs. Once flight level change is engaged it can be disabled if the following modes are selected: ALT HOLD select, VS select, VNAV select, or GA select.

(11) Lateral Navigation Mode

(a) Pressing the Lateral Navigation (LNAV) mode select switch/light initiates an LNAV mode request discrete. This discrete generates an LNAV SELECT signal if the FCC is engaged in CMD or the F/D is on. The LNAV SELECT signal arms the LNAV mode. The LNAV mode can be manually disarmed by pressing the LNAV switch/light a second time.

(b) The dot-bar matrix in the LNAV switch/light lights when the MCP receives LNAV OPER or LNAV ARM from the FCCs. The LNAV OPER signal is also sent to the EFIS symbol generators for mode display on the EADIs. Once LNAV mode is engaged, it can be disabled if any one of the following modes are selected:

1) HDG HLD (Heading Hold)
2) HDG (Heading)
3) GA (Go-Around)

(c) The same signal that lights the LNAV switch/light is provided, by the MCP, to the Flight Management System (FMS) to enable LNAV steering commands from the FMC (AMM 34-61-00/501).

(12) Vertical Navigation Mode

(a) Pressing the Vertical Navigation (VNAV) mode select switch/light initiates a VNAV mode request discrete. This discrete generates a VNAV SELECT signal if the FCC is engaged in CMD or the F/D is on. The VNAV SELECT signal arms the vertical navigation mode. The VNAV mode can be manually disarmed by pressing the VNAV switch/light a second time.

(b) The dot-bar matrix in the VNAV switch/light lights when the MCP receives VNAV OPER from the FCCs. The VNAV OPER signal is also sent to the EFIS symbol generators for mode display on the EADIs. Once VNAV MODE is engaged it will be disabled if any one of the following modes is selected:

1) ALT HLD (Altitude Hold)
2) V/S (Vertical Speed)
3) FLCH (Flight Level Change)

(c) The same signal that lights the VNAV switch/light is provided, by the MCP, to the FMS to enable VNAV steering commands from the FMC (AMM 34-61-00/501).

(13) Localizer Mode

(a) Pressing the localizer (LOC) mode select switch/light initiates a localizer mode request discrete. This discrete generates a LOC MODE request signal if the FCC is engaged in CMD or the F/D is on. The LOC MODE signal arms the localizer mode. This signal can be manually disarmed by pressing the LOC mode switch/light a second time.

(b) The dot-bar matrix in the LOC switch/light lights when the MCP receives LOC MODE ARM/OPER. LOC MODE OPER is also sent to the EFIS symbol generators for mode display on the EADIs. Once the localizer mode is engaged it can be disabled if the following modes are selected: HDG HLD select, HDG select, or LNAV select.

(14) Approach Mode

(a) Pressing the Approach (APP) mode select switch/light initiates an approach mode request discrete. This discrete generates an APP MODE request signal if the FCC is engaged in CMD or the F/D is on. The APPROACH MODE request signal arms the approach mode. This signal can be manually disarmed by pressing the APP switch/light a second time.

(b) The dot-bar matrix in the APP switch/light lights when the MCP receives APPROACH MODE ARM/OPER from the FCCs. Once the approach mode is engaged, it can be disabled when the following modes are selected: HDG HLD Select, HDG select, or LNAV select.

(15) ALT, VERT SPD, & HDG Reference Displays/Controls
- (a) Encoders and Displays
  - 1) Each numerical magnetic wheel display has an optical encoder. This is controlled by a knob or thumbwheel which is used for modifying the display. Each encoder has dual circuits on a common shaft for interfacing with channels A and B of the MCP.
  - 2) Each display has initial, maximum, and minimum values and an incremental value as previously defined in the component details section.
  - 3) The V/S and HDG displays can be initialized/synchronized by the FCCs or manually set by the crew.

(16) IAS/MACH Reference Displays/Controls
- (a) IAS/MACH Display States and Transitions
  - 1) The IAS/MACH display has four valid states.
    - a) OFF
    - b) Airspeed being displayed: Airspeed (IAS) display is entered automatically when MCP power is turned on or when the SPD knob is pressed. Airspeed is also displayed if the SEL switch is pressed when MACH is being displayed, or if the display is blank when SPD or FLCH modes are initiated.
    - c) MACH being displayed: this state is entered if the SEL switch is pressed when airspeed is being displayed and the airplane's speed is between 0.40 and 0.95 mach.
    - d) Speed display blank: this state is entered from either IAS or MACH display states if the VNAV mode is initiated or the SPD knob is pressed with the AFDS not in the FLCH, SPD, T/O or G/A modes. (These are airspeed-being-displayed only modes.)
- (b) Encoders and Displays
  - 1) The numerical magnetic wheel display is modified by an optical encoder which is controlled by the rotary knob. The encoder has dual circuits on a common shaft for interfacing with channels A and B of the MCP. The display shows airspeed from 100 to 399 knots in increments of one knot, and Mach number from 0.40 to 0.95 in increments of 0.01 Mach. If the reference airspeed from the FCC is invalid, the display shows 200 knots.
  - 2) The IAS/MACH display can be initialized/synchronized by the FCCs or manually set by the crew.

(17) AFDS MCP Disengage Bar - Operation

(a) The DISENGAGE bar disengages all three autopilots when the bar is down. The DISENGAGE bar assembly consists of three DPDT toggle switches (one for each FCC) tied together with a bar across the bat handles. When the bar is up, the servos may be engaged; when down, 28 Vdc is removed from the servos, preventing engagement. When the bar is down, a dayglow orange strip is visable, annunciating the disengaged position of the switch.

J. Autopilot/Flight Director System Interfaces (Fig. 4)

(1) The integrated Flight Management Systems (FMS) consists of the L and R FMCs, the L, R, and C AFDSs, the Thrust Management System (TMS), and the Maintenance Control Display Panel (MCDP).

(a) The AFDS can function with the A/P off, and the F/D on; or with single channel A/P engaged in CMD, with the F/D on. The configuration for autoland operation is normally three channels, but can be two autopilot channels in CMD and F/D off or on.

(b) With the CMD mode engaged, the AFDS can provide flight path guidance and control using raw data input from interfacing sensors. In an FMS coupled mode, it receives computed VNAV and LNAV pitch and roll commands from the FMC. The AFDS also provides the commands for automatic stabilizer trim.

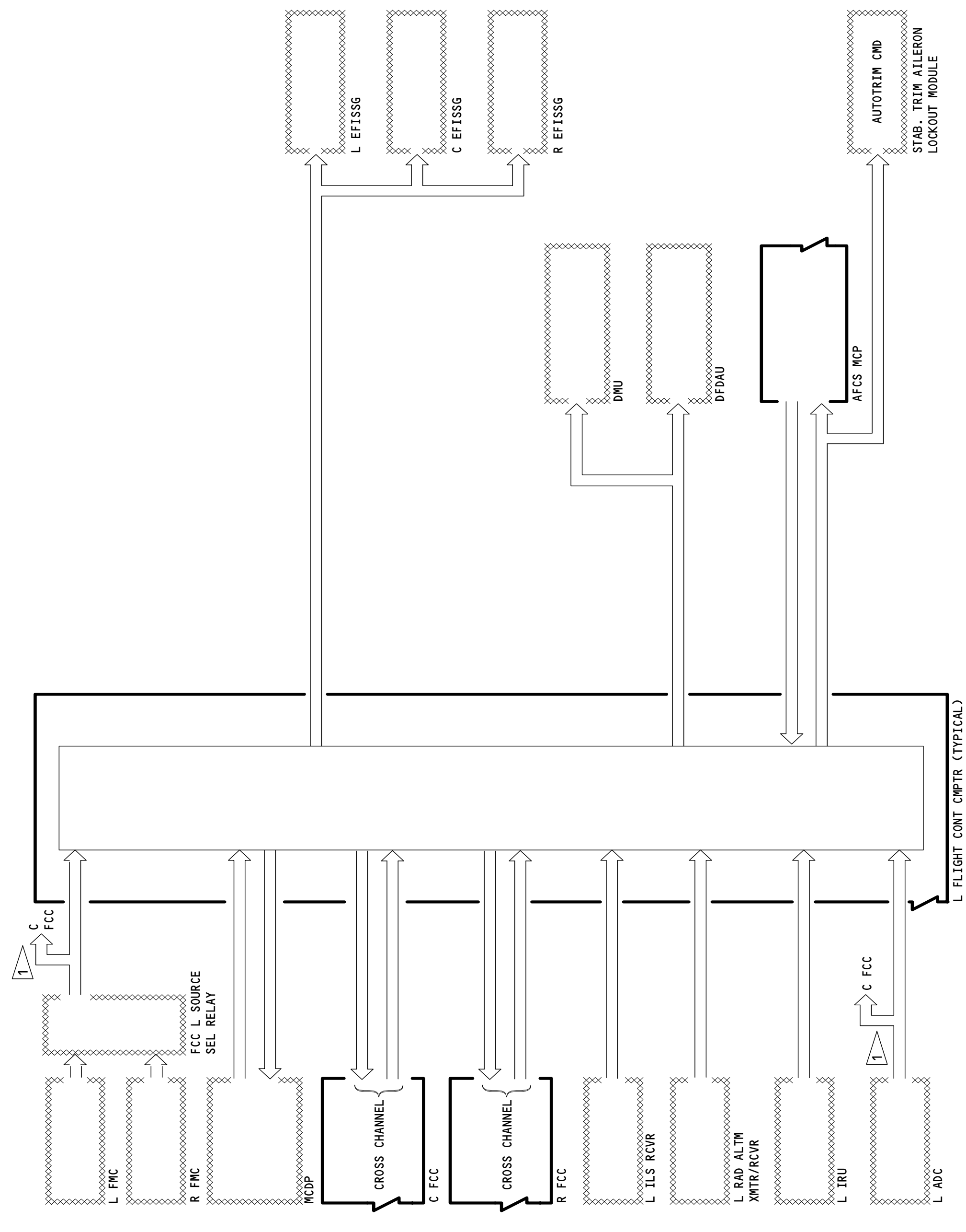
(c) Full system monitoring and Built-In-Test Equipment (BITE) detect system faults in flight. This data is stored for recall by the MCDP after landing.

(d) Digital interfaces between the AFDS and associated systems use ARINC 429 DITS buses.

(2) AFDS Interfaces

(a) Each FCC has two-way interfaces with all A/P servos (LCCAs, DASs, and EASs), the MCP, the SAMs, the MCDP, and the other two FCCs.

(b) Each FCC receives data from critical sensors (dedicated sources for AUTOLAND signals) and non-critical sensors (non-dedicated sources and switched inputs). Each FCC also uses signals from the A/P disconnect switches, go-around switches, and FCC program pins.



1 R SYSTEM TO R FCC

DIGITAL INPUT AND OUTPUT

Autopilot/Flight Director System Interfaces Schematic
Figure 4 (Sheet 1)

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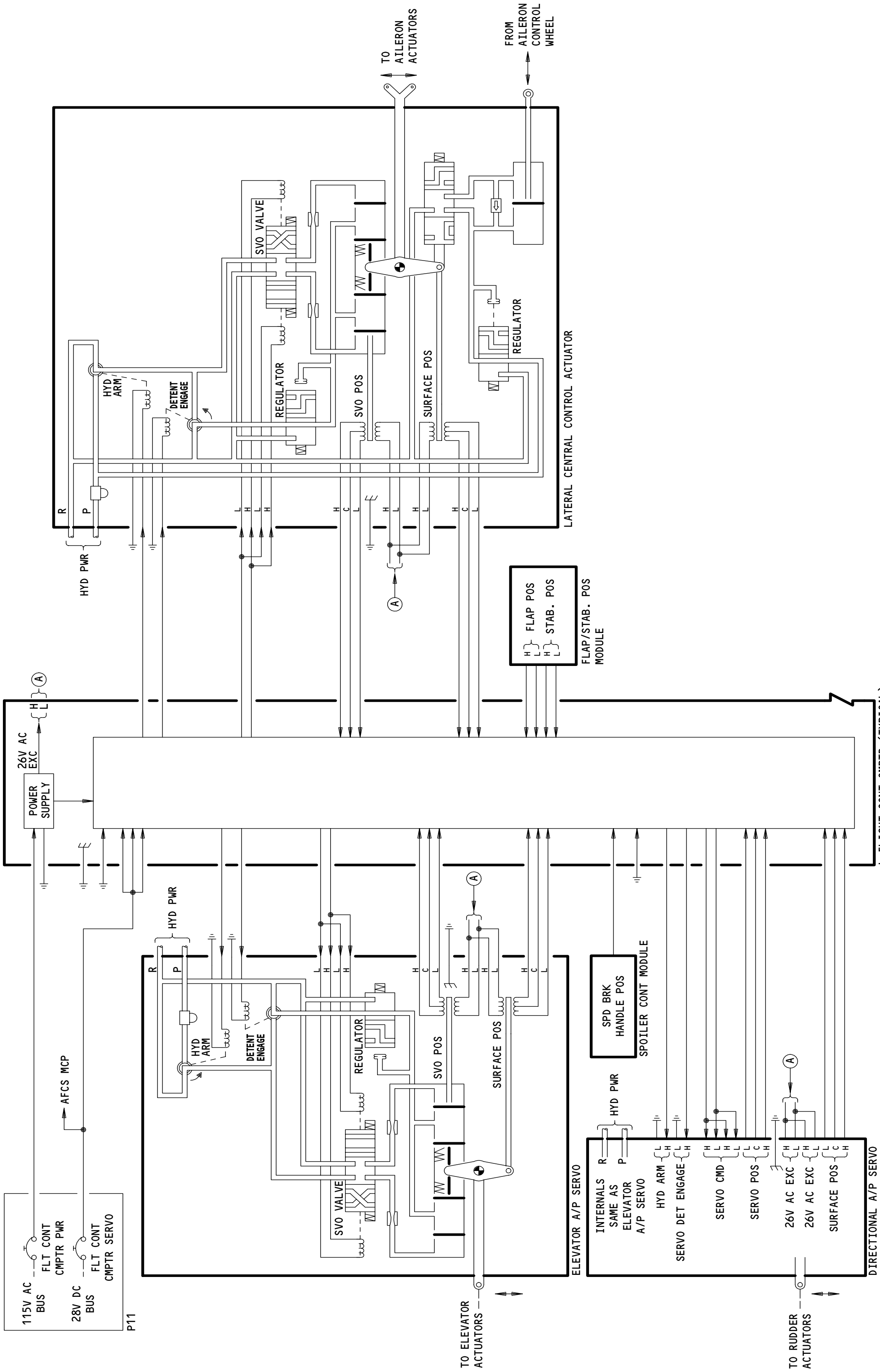

ANALOG INPUT AND OUTPUT

Autopilot/Flight Director System Interfaces Schematic
Figure 4 (Sheet 2)

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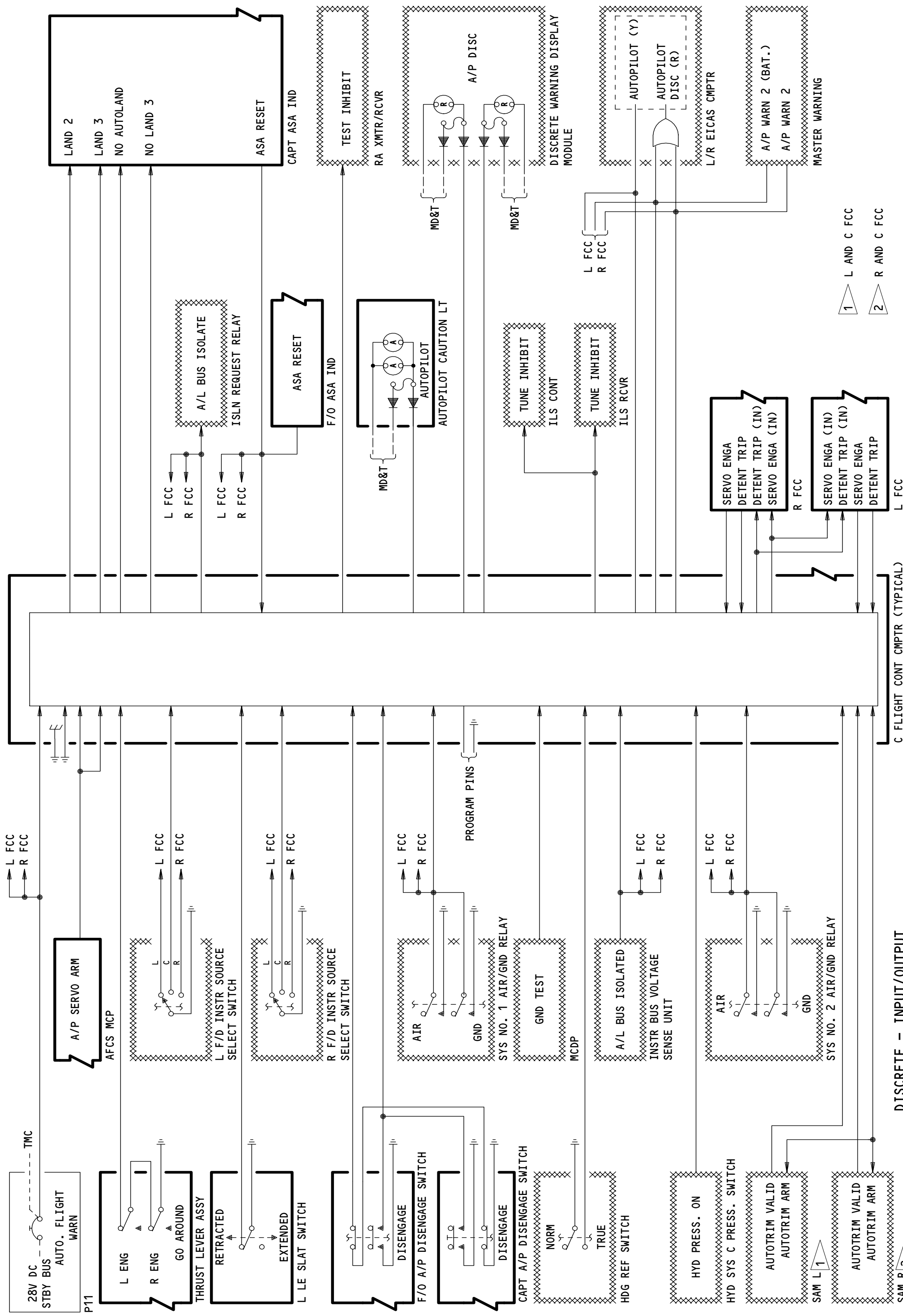


DISCRETE - INPUT/OUTPUT

Autopilot/Flight Director System Interfaces Schematic
Figure 4 (Sheet 3)

(c) Each FCC sends data to the Autoland Status Annunciator (ASA), the Engine Indication and Crew Alerting system (AMM 31-41-00/501), and symbol generators (SGs) of the Electronic Flight Instrument System (EFIS) (AMM 34-22-00/501, Flight Instrument System).

(3) The MCP interfaces with the Warning Electronics Unit (WEU) (Chapter 31), each FMC (Chapter 34) and the TMC (Chapter 22).

(a) The MCP has a two-way analog and digital data link with the left and center FCCs, and a separate two-way link with the right FCC. Each FCC has a two-way link with the other FCCs and the MCDP.

(b) The MCP interfaces with the following units via ARINC 429 data links:

1) Digital Flight Data Acquisition Unit (DFDAU) (AMM 31-31-00) and Data Management Unit (DMU) (AMM 31-35-00)
2) Left, center, and right FCCs (AMM 22-11-00).
3) Thrust Management Computer (AMM 22-31-00).
4) Altitude alert warning module in the Warning Electronic Unit (WEU) (AMM 31-51-00, Caution and Warning System).

(4) The A/P disengage switches on the control wheels interface with all three FCCs.

(5) The go-around switches on the thrust levers interface with the TMC and the FCC.

(6) Each (2) FLT DIR (Flight Director) source select switch selects the L, C, or R FCC as the data source for F/D displays. The left switch, (P1-1 panel) controls the captain's EADI. The right switch (P3-3 panel) controls the first officer's EADI.

(7) The EFI (Electronic Flight Instrument) switch found on the left instrument source select panel (P1-1 panel) selects the left or Center EFIS SG (Symbol Generator) as the display processor for the left Electronic Horizontal Situation Indicator (EHSI) and EADI. The EFI switch found on the right instrument source select panel (P3-3 panel) selects the right or center (ALTN) symbol generator for the right EHSI and EADI.

(8) The FMC switch found on the left instrument source select panel (P1-1 panel) selects the left (normal) or right (alternate) FMC as the source of data for the left and center FCCs and captain's EHSI, EADI. The FMC switch found on the right instrument source select panel (P3-3 panel) selects the right (normal) or left (alternate) FMC as the source of data for the right FCC (through the Capt & F/O Both On SG-C Relay) and F/O's EHSI, EADI. If the two EFI switches found on the left and right instrument source select panels are in the ALTN position, the Capt & F/O Both On SG-C Relay is energized. The FMC switch found on the left instrument source select panel then determines which FMC is the source for the right FCC.

(9) If the two EFI switches found on the left and right instrument source select panels are selected to the same source a LEVEL G AURAL sounds and an amber INSTR SW message is displayed on EICAS. Instrument switching is covered in detail in 34-22-00, Flight Instrument System.

K. Sensors

(1) Critical sensors are those which provide data to the FCC's during Autoland. Each FCC receives a dedicated input from a critical sensor. Autoland critical sensors are the Instrument Landing System (ILS) receiver, Inertial Reference Unit (IRU), Radio Altimeter (RA), and the Electrical Power (Control) System (EPCS).

(2) Non-critical sensors are those which do not provide raw data during Autoland. The FCCs receive non-critical sensor inputs through instrument switching or common data links. Non-critical sensors are the Flight Management Computer System (FMCS), Air Data Computing System (ADCS), Proximity Switch Electronic Unit (PSEU), spoiler control modules, and the flap and slat position sensors.

L. AFDS Outputs

(1) The AFDS produces the following outputs:
- (a) Autopilot servo commands.
- (b) Flight director displays.
- (c) Mode status and Autoland annunciation.
- (d) Caution and Warning outputs.
- (e) Flight data acquisition signals.

(2) Interchannel data permits data sharing and total system verification during Autoland (Ref 22-15-00).

(3) Each FCC provides analog servo commands to its own LCCA EAS, and DAS Electrohydraulic Servo Valves (EHSV). Each FCC also provides analog discretes to the other two FCCs when its pitch (EAS) and roll (LCCA) A/P servos are engaged.

(4) Output buses one, two, and three from each FCC supply the MCP, MCDP, and EFIS symbol generators respectively. Output bus four from each FCC supplies the Data Management Unit (DMU) (AMM 31-31-00) and from the left FCC supplies the Digital Flight Data Acquisition Unit (DFDAU) (AMM 31-31-00).

(5) FCC - Analog Discrete Outputs
- (a) Pin A6 supplies a ground to the electrical power (control) System when the AUTOLAND BUS ISOLATE REQUEST discrete is produced after APP selection (AMM 24-22-00/201).
- (b) Pin F9 normally supplies 28 vdc, but changes to ground if the FCC detects any detent trip condition. The signal is fed to the other two FCCs.
- (c) Pin G10 supplies 28 vdc to the other two FCCs when A/P servos are engaged. A ground is supplied when they are disengaged.
- (d) Pin K2 supplies an A/P CAUTION - 1 ground which illuminates the AUTOPILOT amber caution light on the P1-3 panel.
- (e) Pin K3 supplies an A/P CAUTION - 2 ground to the Engine Indication and Crew Alerting computer (Ref 31-41-00).
- (f) Pin E12 (A/P WARN 2 - BAT) and pin E12 (A/P WARN 2) both supply grounds in the A/P WARN condition to two independent inputs of the aural warning module in the WEU (AMM 31-51-00).

(g) Pin G12 (A/P WARN 1 - BAT) and pin G12 (A/P WARN 1) both supply grounds in the A/P WARN condition to two pairs of bulbs in the red A/P DISC warning light on the P1-3 panel.

M. AFDS Inputs

(1) Each FCC has analog inputs as shown from its own LCCA, elevator autopilot servo, directional autopilot servo, left aileron CFT, right and left elevator CFTs, and from the respective (L, C, and R) flap and stabilizer position modules. Each FCC has provisions for signal inputs from a right aileron CFT if it is installed.

(2) FCC Digital Inputs and Cross Channel Lines

(a) Cross-channel data is encoded in vendor unique code; all other data is transmitted in standard ARINC 429 format. Critical sensor data is transmitted on dedicated data links while non-critical data is stored or switched.

(b) The MCP has two identical micro-processors, A and B, which perform identical functions. The A processor interfaces with the left and center FCCs; the B processor, with the right FCC.

(c) Each FCC has independent data links with the MCDP.

(3) FCC Analog Discrete Inputs

(a) Pin H15 receives 28 vdc from the MCP when the A/P servos are armed. Pin C6 receives 28 vdc from the hydraulic system low pressure light, on the P10 panel, when the light is off.

(b) Pin C7 changes from ground to open after the electrical power control system has completed Autoland power bus isolation.

(c) Pin A3 receives a ground from the MCDP when the ground test mode is selected.

(d) Pins C13, F3, or F1 receive a ground from the air/ground relay when the airplane is on the ground. Two of these pins for each FCC are wired to the Air/Ground Systems 1 and 2.

(e) Pin H2 receives a ground or open when the MAG/TRUE switch is in the TRUE or MAG position respectively.

(f) Pin H11 receives a ground whenever the A/P DISC switch is pressed.

(g) Pin F15 (connector B) normally has a ground through both A/P DISC switches (in series) and receives an open whenever either switch is pressed.

(h) Pin E3 or G1 receives 28 volts DETENT TRIP from another FCC (from pin F9).

(i) Pin K1 or K3 receives 28 volts SERVO ENG DC from another FCC (from pin G10 output).

(j) Pins J10 and K10 receive ground whenever the RESET switch is pressed on either ASA.

(k) Pin G9 on both connectors receive 28 vdc autotrim valid from the L and R Trim and Aileron Lockout Module (SAM). The center FCC receives signals from the left and right SAMs. The right and left FCCs interface only with the applicable left or right SAM.

(l) Pin F10 or F9 receives a ground when one of the two FCC switches found on the left and right instrument source select panels selects the FCC.

(m) Pin E9 receives an open or a ground when the leading edge slat position switch is in the DEPLOYED or STOWED position respectively.

(n) Pin F15 (connector A) normally receives a ground when both go-around (GA) switches are relaxed and an open when either switch is pressed.

N. Switching (Fig. 4A)

(1) Source Select Switches

(a) Each FLT DIR switch found on the left and right instrument source select panels connects the selected FCC computed F/D commands and selected data to the associated EADI.

(b) Each FMCS NAV switch found on the left and right instrument source select panels controls FMC data to the FCCs with inputs supplied to the FCC source select relays.

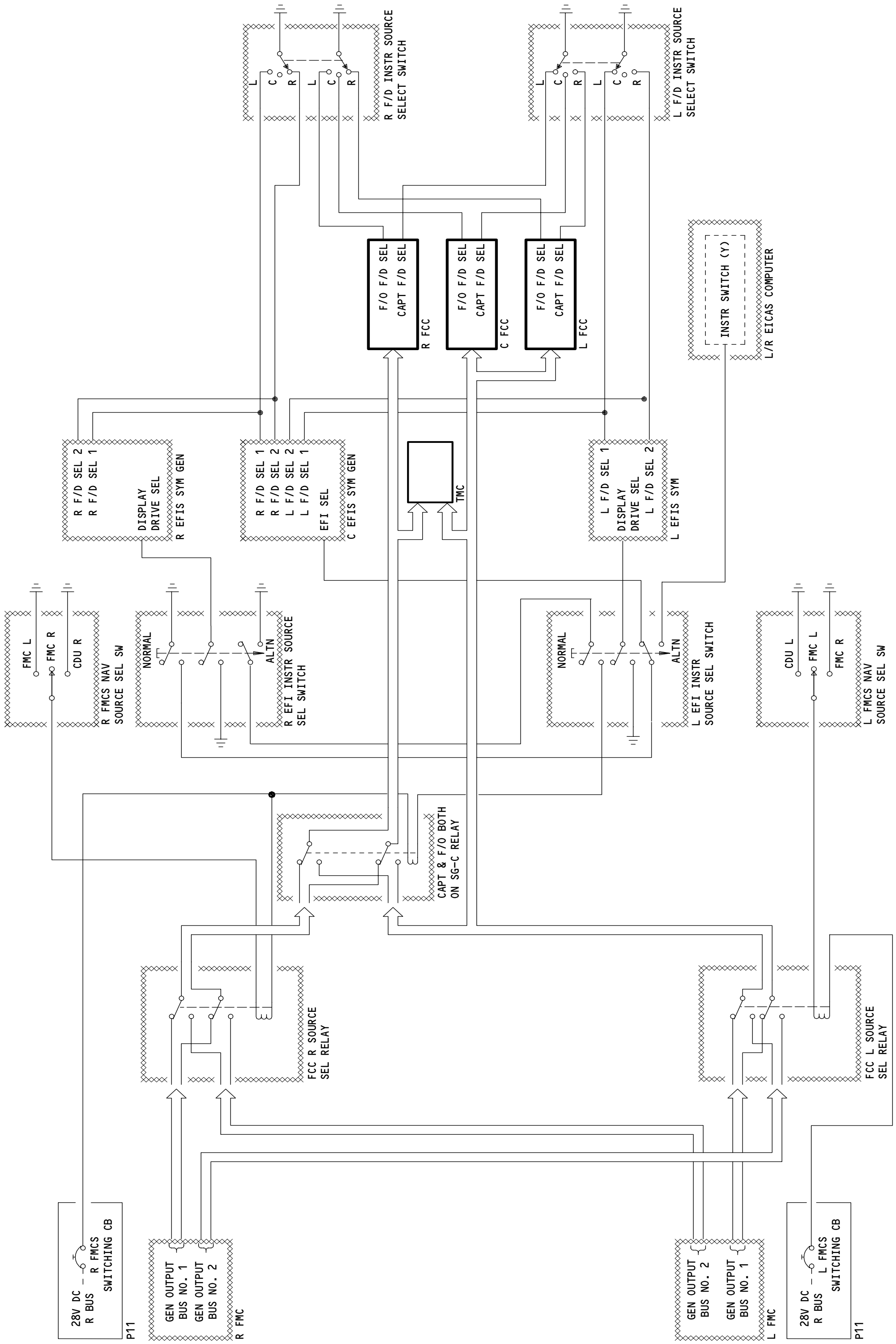
(c) Each EFI switch found on the left and right instrument source select panels enables the NORMAL or ALTERNATE symbol generator to process incoming AFDS and FMCS data. The EFI source select switches also supply a ground to the Capt & F/O both on SG-C relay which controls FMC data to the R FCC.

(d) The IRS switches found on the left and right instrument source select panels do not control IRU inputs to FCCs but they do control attitude display signals.

(e) The air data switches found on the left and right instrument source select panels do not affect the ADC data to the FCCs. The left ADC supplies data to the left and center FCCs and the right ADC supplies data to the right FCC.

(2) F/D Switching

(a) Positioning the captain's or first officer's F/D switch to ON permits the FCCs to compute F/D commands and activates the EADI FMA. These commands can be generated simultaneously by an FCC engaged in CMD mode.



AFDS Instrument Switching Schematic
Figure 4A

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(b) Each FLT DIR switch found on the left and right instrument source select panel is a three-position, rotary switch which provides a ground to the left, center or right FCC. This enables that FCC to supply computed F/D signals to the selected EFIS Symbol Generators (SGs). The FLT DIR switch found on the left instrument source select panel supplies a ground to the left and center EFIS symbol generators when in the L or R position. These grounds are identified as L F/D SEL 1 and L F/D SEL 2 respectively.

(c) The FLT DIR switch found on the right instrument source select panel supplies a ground to the right and center EFIS SGs when in the R or L position. These grounds are identified as R F/D SEL 2 and R F/D SEL 1 respectively.

(d) The FLT DIR switch grounds for the EFIS SGs tell the SG which F/D data input to use for the L or R F/D display (EADI). When one of the two FLT DIR switches are in the C position, no F/D source ground is supplied.

(3) EFI Switching

(a) The two EFI switches found on the left and right instrument source select panels are alternate action pushbutton switches. The legend ALTN (alternate) is illuminated when the associated switch is pressed to select another Symbol Generator (SG). The EFI switches control the DISPLAY DRIVE SELECT inputs to the left and right EFIS SGs and the EFI select to the center EFIS SG. Each switch in the NORMAL position supplies an open to the DISPLAY DRIVE SELECT input. This enables the left or right SG to select the source of video for data processing. If either EFI switch is in the ALTN position, the normal SG for that switch is disabled.

(b) The POWER DOWN input to the left EFIS SG is an open when the captain's switch is normal, but grounded when in ALTN. This ground disables the left SG. The POWER DOWN input to the center SG is grounded when both EFI source select switches are normal. Thus, the center SG is disabled. If either EFI switch is in ALTN, the center SG is powered up. The right SG has a permanently open POWER DOWN input.

(c) The EFI SELECT input controls which data source is used by the center SG. This signal is routed through the two EFI switches found on the left and right instrument source select panels. When open, the center SG selects the left data source. This signal input is an open under all combinations of EFI switch positions except when the captain's switch is in normal and the F/O's is in ALTN. If both switches are set at ALTN, the captain has priority use of the center SG. The master CAUTION light illuminates, a level B aural sounds, and EICAS message INSTR SWITCH is displayed.

(4) FMC Data Switching
 (a) The FMCS NAV switches found on the left and right instrument source select panels control the source of FMC data to the FCCs. Each switch is a three-position rotary switch which supplies a ground to the FCC L or FCC R source select relay. Each FMC supplies the same data to each relay. The FMCS NAV switch found on the left instrument source select panel switch sets the FMC source for the left and center FCCs. The FMCS NAV switch found on the right instrument source select panel selects the FMC source for the right FCC when the two EFI switches found on the left and right instrument source select panels are not in the ALTN position. If the two EFI switches are in ALTN, the Capt & F/O both on SG-C relay energizes and the FMC data source is set by the left FMCS NAV switch.

(5) IRU Switching
 (a) Each IRU supplies dedicated data to its respective FCC from output bus one, and to its respective SG from output bus two. The left and right IRUs supply data to the center SG on output bus 3. The center IRU supplies data to the left and right SGs on output bus 3. In each IRU, bus one data and bus three data are identical.
 (b) When the IRS switch found on the left instrument source select panel is in NORMAL, the left and center SGs receive a left IRS SEL discrete which enables both SGs to process data from the left IRU. If the left IRS SEL line is grounded (the left IRS switch in ALTN), the left and center SGs select center IRU data. Operation of the IRS switch found on the right instrument source select panel is similar. This arrangement permits the source of data for the attitude display (as selcted by the IRS switches) to be different from the source of data for the F/D commands on the same EADI. This is done because IRU data to each FCC is dedicated.

(6) FCC Program Pin Connections (Fig. 5)
 (a) Program pins are permanently wired at the shelf connector. They provide inputs to the FCC which control certain operational characteristics. Program pins enable the following functions and options to be selected.
  1) Boeing options
  2) Airplane configuration options
  3) Channel identification
  4) Configuration interlocks
  5) Customer options
  6) Program pin parity

OPTION DESCRIPTION	BARRIER GROUND CONNECTIONS INSERT-PIN TO INSERT-PIN		1 = CONN TO GRD 0 = OPEN	MCDP TEST 56 READOUT
FCC CONFIGURATION - PIN 1	A-A12	A-A11	0	
PIN 2	-B12	-B11	0	
PIN 3	-C12	-C11	1	04 4
PIN 4	B-A11	B-G8	0	
PIN 5	-B11	-H8	0	
FCC INTERLOCK - PIN 1	B-C11	B-J8	0	
PIN 2	-F11	-K9	0	
PIN 3	-D12	-D15	0	00 4
PIN 4	-F12	-K7	0	
PIN 5	-H12	-J15	0	
FCC OPTION 1				
CUSTOMER OPTION PIN 1	B-A12	B-B15	0	
CUSTOMER OPTION PIN 2	-B12	-D15	1	
CUSTOMER OPTION PIN 3	-C12	-E5	1	06 4
CUSTOMER OPTION PIN 4	-A13	-A14	0	
CUSTOMER OPTION PIN 5	-B13	-B14	0	
FCC OPTION 2				
A/P MODE ENGAGE	B-A3	B-E4	1	
G/S CAP INHIBIT	-H13	-J14	0	
FULL TIME NO LAND	-K13	-K14	1	37 4
METHOD 2	-G13	-H14	0	
AUTOLAND STATUS ANNUNCIATOR	-C8	-E8	0	
LATERAL COMMAND ENGAGE	-C10	-C14	1	
FCC OPTION 3				
CWS INHIBIT	B-J2	B-J4	1	
FULL TIME F/D	-D3	-K4	1	
F/D AUTOMATIC ON	-A4	A-D4	1	31 4
ILS ANOM DLY	-D4	B-H4	1	
A/P ENGA TO/GA OPT	-G1	A-G2	1	
SYS ARCHITECTURE	-H1	-H2	0	
FCC OPTION 4 3				
GS RELAY STATUS	B-G2	A-G4	0	
MAG/TRUE IN	-H2	1	0 OR 1	16 (MAG)
LOC RELAY STATUS	-D7	B-G4	0	18 (TRUE)
EO A/L INHIBIT	-G10	A-G8	0	4
SINGLE SOURCE A/P	-J10	-B10	1	
PARITY (ODD) THE SUM OF THE GROUNDED PINS (TOGETHER WITH THE PARITY PIN) MUST BE EQUAL TO AN ODD NUMBER. YOU MUST GROUND THE PARITY PIN OR KEEP IT OPEN TO GET ODD PARITY.	B-C13	B-F4	L = 0 OR 1 C = 1 OR 0 R = 0 OR 1	2 4

FCC PIN CONFIGURATION (EXAMPLE) 4

FCC	CH IDENT 1	CH IDENT 2
LEFT	GROUND	OPEN
CENTER	GROUND	GROUND
RIGHT	OPEN	GROUND

CHANNEL IDENTIFICATION TABLE

1 CONNECTED TO MAG/TRUE HDG SWITCH ON P3-1 PANEL

2 THE PARITY SHOWS ON THE DISPLAY. IF THE PARITY IS INCORRECT, THE STATUS WILL BE SHOWN AS "ADD" OR "DEL" FOR THE ASSOCIATED CHANNEL.

3 FCC OPTION 4 GROUP NOT INCLUDED IN PARITY CHECK

4 ALL PIN CONNECTIONS AND "MCDP TEST 56 READOUT" VALUES ARE SHOWN AS AN EXAMPLE. FOR SPECIFIC VALUES FOR EACH AIRPLANE, REFER TO MCDP GROUND TEST 56 (FIM 22-00-04/101, FIG. 103).

FCC Program Pin Connections
Figure 5

(b) Five pin connections at the shelf identify the aircraft type. The FCC reads these pins as a binary code; there are 32 possible combinations.
  1) SAS 050-149;
     FCC configuration pins B12 and C12 on connector insert A are grounded.
  2) ALL MTH AIRPLANES; SAS 150-999;
     FCC configuration pin C12 on connector insert A is grounded.

(c) Two pin connections of the shelf identify the FCC as L, C, or R channel. The binary code used is shown in the table of Fig. 5.

(d) The five configuration interlock binary coded pins provide interlock protection during exchange of FCCs with different part numbers. For example, if FCCs -101, -102, and -103 are available and FCC capability increases with each dash number, a -103 FCC can function in lieu of a -101 or -102, but a -101 or -102 can not function in lieu of a -103. Presently the configuration interlock pins are used only for the -109 FCC's. Interlock protection to make sure other FCC dash numbers are not mixed is done in the FCC Option 1.

(e) FCC Option 1
  1) Customer Option 1 (pin P1B-A12 grounded) - When this pin is grounded, the -103 FCC cannot be used with subsequent dash number FCCs. Also, Customer Option 1 includes these functions:
     a) CHANGED BANK ANGLE LIMIT SPEED SCHEDULE: The bank angle limit speed schedule in the AUTO position is changed to give increased turn operation in the terminal area with airspeed between 200 and 250 knots.
     b) CHANGED LOGIC FOR ENGINE-OUT INPUT: The operation of the engine-out logic during electrical power transfers is changed to prevent the command of nuisance V2 airspeed during F/D takeoff mode.
     c) CHANGED TAKEOFF AND GO-AROUND CONTROL LAW: The takeoff and go-around control law is changed to give better airplane guidance when there is wind shear.
     d) CHANGED HEADING SELECT CONTROL LAW: The heading select control law is changed to make the turn rate of the airplane better.
     e) CHANGED FLIGHT LEVEL CHANGE (FLCH) CONTROL LAW: The FLCH control law is changed to make the climb performance during flap retraction better.
     f) ADDED GLIDESLOPE ANOMALY ANNUNCIATION AND MAINTENANCE REPORTING: A glideslope anomaly detector and associated annunciation and maintenance reporting are added to decrease the risk of the autopilot track a disturbed glideslope beam.

g) CHANGED LNAV AND VNAV MODE ARMING LOGIC: The mode select logic is changed so that LNAV and VNAV are not armed while in the go-around mode below 400 feed radio altitude so that pitch or roll errors do not add up before LNAV or VNAV engagement.

h) CHANGED ALTITUDE CAPTURE LOGIC DURING TAKEOFF AND GO-AROUND: The altitude capture logic is changed to prevent altitude capture below 800 feet radio altitude during takeoff or go-around to prevent accidental altitude capture during wind shear.

i) CHANGED NOSE LET-DOWN LOGIC AT TOUCHDOWN: The nose let-down logic is changed to keep pitch-up at touchdown to a minimum when the speed brakes are extended.

j) CHANGED FLIGHT MODE ANNUNCIATION: The flight mode annunciation is changed to remove ALPHA annunciations during FLCH and go-around modes.

k) CHANGED GO-AROUND LOGIC: The go-around logic is changed to allow a transition from takeoff mode to go-around mode.

l) ADDED F/D AUTO ON OPTION: A program pin is available which supplies flight director guidance by pilot operation of the takeoff/go-around switches if the flight director was not engaged before.

NOTE: This feature is part of the wind shear protection system and the F/D AUTO ON program pin must be grounded for this feature to operate.

2) Customer Option 2 (pin P1B-B12 grounded) - When this pin is grounded, the -105 FCC cannot be used with subsequent dash number FCCs. The functions listed for customer option 1 are still applicable. Customer Option 2 is required for 767-300 airplanes, and includes this function:

a) CHANGED FLARE INITIATION HEIGHT: The flare initiation height is changed from 45 feet to 50 feet.

3) Customer Option 3 (pin P1B-A12 grounded, and pin P1B-B12 grounded) - When these pins are grounded, the -105 FCC cannot be used with subsequent dash number FCCs. The functions listed for customer option 1 and customer option 2 are still applicable. Also, Customer Option 3 includes these functions:

a) CHANGED BACKCOURSE (BCRS) MODE LOGIC: Mode logic is changed to permit a change from the flight director takeoff mode directly to the flight director BCRS mode. This will make BCRS departures easier at airports where it is necessary to use these procedures.

b) NUISANCE GLIDESLOPE ANOMALY TRIPPING DECREASED: The delay on the glideslope anomaly detector annunciation is increased from 10 seconds to 15 seconds to decrease the risk of nuisance glideslope beam error fault indications. Also, the glideslope anomaly detector is disconnected for flight director only approaches.
c) CHANGE F/D AUTO-ON LOGIC: The F/D auto on feature is prevented below 80 knots indicated airspeed.
d) RUDDER VALVE TEST PREVENTED FOR ENGINE-OUT: The rudder valve test is prevented if one engine has failed before approach to prevent nuisance NO LAND 3 annunciations because of rudder trim.
e) CHANGED STABILIZER TRIM LOGIC: Nose up stabilizer mistrim at 330 feet AGL is prevented during LAND 3 operation.
f) CHANGED HEADING SELECT LOGIC: The heading select logic is changed to prevent a wrong-way turn flight director command after a go-around.
g) CHANGED MULTI-CHANNEL A/P ENGAGE LOGIC: Multi-channel autopilot engagement is prevented if there is an autopilot caution. This will prevent LAND 3 annunciation with the glideslope anomaly detector open.
h) CHANGED ADC SELECT LOGIC: A latch delay is added to the Air Data Computer (ADC) select logic prevent inadvertent transfers to cross-channel ADC inputs.

4) Customer Option 6 (pin P1B-B12 grounded, and pin P1B-C12 grounded) - When these pins are grounded the -133 FCC cannot be used with lower dash number FCCs. The functions listed for Customer Option 1 and Customer Option 3 are still applicable. Also, Customer Option 6 includes these functions:
a) BETTER ALTITUDE CAPTURE MANEUVERING: The altitude capture maneuvering is changed to give speed protection during noise abatement takeoff operations and go-around maneuvers. This also gives better Autopilot/Flight Director operations when used with deep thrust cutbacks during climb.
b) BETTER TAKEOFF PERFORMANCE: The takeoff performance is changed to enable speed tracking immediately after takeoff.
c) BETTER FLIGHT DIRECTOR AND GO-AROUND MODES: Flight director takeoff operations and speed/attitude transitions are better.
d) REMOVED POSSIBLE AUTOPILOT DISCONNECT: A possible autopilot disconnect during the climb of a fail Passive Dual Channel Go-Around in moderately turbulent weather is removed.
e) BETTER AUTOLAND FLARE CONTROL LAW: The Autoland flare control law is changed to limit the maximum pitch attitude during the flare maneuver more than before.

f) ADDED AN OPTION FOR SINGLE SOURCE AUTOPILOT OPERATION: This allows engagement of the autopilot with only a single FCC, IRU or Stab Position that operates.

NOTE: A -104 or higher dash number IRUs also necessary for single source autopilot operation. When Option Pins 2 and 3 are selected the SINGLE SOURCE A/P pin is wired also.

g) DECREASED TIME DELAY FOR AN ILS TRANSMITTER MODE FAILURE ANNUNCIATION TO COME ON:
The time delay for an ILS transmitter mode failure annunciation to come on when you are below 200 feet is decreased from 15 seconds to 4 seconds.

h) BETTER MAINTENANCE GROUND TEST: The maintenance ground test is changed to let you find soft rudder servo detents when the plane is on the ground.

(f) FCC OPTION 2

1) Autopilot Mode Engage Option

a) The Autopilot Mode Engage Option is enabled when the option pin is grounded.

b) Baseline configuration: The A/P engages into HDG HOLD and synchronizes to V/S regardless of F/D status.

c) Option: On initial CMDENG, the A/P engages into HDG HOLD and synchronizes to Vertical Speed (V/S) if both F/Ds are off, or either F/D is in the Take Off (T/O) or Go-Around (GA) mode. If either F/D is on in any other mode than T/O or GA, the A/P engages into the current F/D mode. Glideslope (G/S) active or localizer (LOC) active modes transition to V/S and Heading (HDG) HOLD. Then GS and LOC are rearmed.

2) Autopilot Disengage Warning, Double/Single Push Reset

a) When an A/P disengage warning occurs due to a control wheel disengage switch action, the warning reset procedure depends on grounding of the option pin.

b) Baseline configuration: The control wheel A/P disengage switch must be pressed and released a second time to reset the warning.

c) Option: The warning is reset automatically after 2 seconds.

d) Reset of the A/P disengage warning due to any other means of disconnect is achieved by pressing and releasing either control wheel A/P disengage switch. The warning continues until the switch is released.

3) Full-Time No Land 3 Option

a) This option controls the display capability of each ASA.

b) Baseline configuration: The autoland capability is not displayed until the Approach (APP) mode is operative.

c) Option: The autoland capability is displayed full-time.

4) Method 1/Method 2 Approach Mode Channel Selection
   a) At the selection of the APP mode, this option determines how off-line A/P channels are armed.
   b) Baseline Configuration - Method 1: All off-line channels automatically arm for multichannel operation and dot-bar matrices on the CMD ENGAGE switch/lights illuminate.
   c) Option - Method 2: Multichannel arm requires that each CMD ENGAGE switch/light be individually pressed.

5) The ASA option permits the use of an alternate type of ASA.

6) Lateral Command Engage Option (LCEO)
   a) When the A/P is initially engaged in CMD or a F/D is turned on, the roll channel response depends upon this option.
   b) Baseline configuration: When the A/P is engaged in CMD, the airplane is rolled to wings level and the A/P engages into HDG HOLD mode regardless of the bank angle (B/A).
   c) Option: If the B/A is less than 5 degrees, A/P action is the same as in the baseline configuration. If the B/A is between 5 and 30 degrees at CMDENG, the A/P engages into Roll Attitude Hold. When the B/A is greater than 30 degrees at CMDENG, the A/P rolls the airplane back to 30 degrees and then engages into Roll Attitude Hold.

(g) FCC OPTION 3

1) CWS Inhibit Option
   a) This option inhibits CWS related maintenance messages on airplanes without CWS mode.
   b) Baseline configuration: Column-force and wheel-force monitors are active.
   c) Option: Column-force and wheel-force monitors are not active.

2) Full Time Flight Director Option
   a) This option controls the flight director command displays of each EADI.
   b) Baseline configuration: The flight director bars associated with the engaged channel will bias out of view.
   c) Option: Flight director bars are displayed for the engaged channel.

3) F/D Automatic On Option
   a) Flight director engage requests will be automatically generated by the FCC's in the go-around mode. This option supports wind shear capabilities.

4) ILS ANOM DLY
   a) This option finds the time delay between when glideslope and localizer anomalies occur and when they show.

b) Baseline configuration: The time delay for a detected glideslope anomaly is 15 seconds. The time delay for a detected localizer anomaly is 20 seconds.
c) Option: During a multi-channel approach below 200 feet, the time delay for a detected glideslope anomaly or a detected localizer anomaly is 4 seconds.

5) A/P ENGA TO/GA OPT
a) When this pin is selected, engagement of the autopilot into the F/D TO/GA mode is allowed INAIR

6) SYS ARCHITECTURE
a) This pin is not used on the 767.

(h) FCC OPTION 4
1) GS RELAY STATUS
a) This pin is not used on the 767.
2) MAG/TRUE IN
a) The HSI HDG REF switch on the P3-1 panel indicates a magnetic or true heading.
3) LOC RELAY STATUS
a) This pin is not used on the 767.
4) E O A/L INHIBIT
a) When this pin is selected, the Autopilot will still engage with one FCC, IRU or STAB position that operates.

(i) The parity pin provides odd parity for the total of program pin ground connections to meet ARINC 429 standards.

2. Component Details
A. The flight control computer, AFCS mode control panel, AFDS go-around switch, and A/P disengage switch are described in Autopilot/Flight Director Power - Description and Operation (22-11-00).
B. The elevator autopilot servos are described in Autopilot/Flight Director Pitch Channel - Description and Operation (AMM 22-12-00).
C. The directional autopilot servos are described in Autopilot/Flight Director Roll and Yaw Channel - Description and Operation (AMM 22-13-00).
D. The autoland status annunciator, autopilot caution light and A/P disconnect warning light are described in Autopilot/Flight Director Warning and Annunciation - Description and Operation (AMM 22-14-00).

3. Operation
A. Functional Description
(1) General (Fig. 6)
(a) These AFDS configurations define, in parentheses, the type of guidance or control provided. Transition into and out of the ACTIVE configuration is possible as defined in the table. In multichannel engage operation, each FCC may be in a different configuration before all FCCs become CMD (ROLLOUT).

(2) Single Channel Mode Functions (Fig. 7)
(a) The table shows the basic control functions for modes which can be active during single channel operation. Simultaneous pitch and roll modes are those in which both pitch and roll control functions are enabled at mode selection. Pitch and roll axes control laws are not variable by pilot action.
(b) Pitch and roll command and F/D modes may be independently selected and allow pilot control interface actions.

(3) Flight Director Only Block Diagram (Fig. 8)
(a) In the flight director only mode, either or both F/Ds are on and all A/P channels are disconnected. The F/D can use stand-alone reference inputs (uncoupled FMC) or FMC steering signals (coupled FMC) derived from a selected flight plan.
(b) The FCC generates F/D steering commands that are displayed on the EADI. The F/D mode annunciation is also indicated on the EADI. The pilot controls the airplane attitude to follow F/D commands with control wheel and column movement.
(c) In the F/D only mode, surface position (aileron, elevator, rudder) feedback is used to synchronize the servo commands to null. All other signal loops are synchronized in the FCC. Hydraulic power is always required to move control surfaces. In the F/D only mode, solenoids in the actuators permit mechanical input signals only to control surface position. Autopilot servo commands are not used.

ENTRY CONFIGURATION	ACTIVE CONFIGURATION	EXIT CONFIGURATION
ARM, CMD	OFF (PITCH & ROLL)	ARM
OFF	ARM (PITCH & ROLL)	OFF, CMD
ARM	CMD (PITCH & ROLL)	OFF
ARM (ROLLOUT), CMD (ROLLOUT)	OFF (ROLLOUT)	ARM (ROLLOUT)
OFF	ARM (ROLLOUT)	OFF, CMD (ROLLOUT)
ARM (ROLLOUT	CMD (ROLLOUT)	OFF

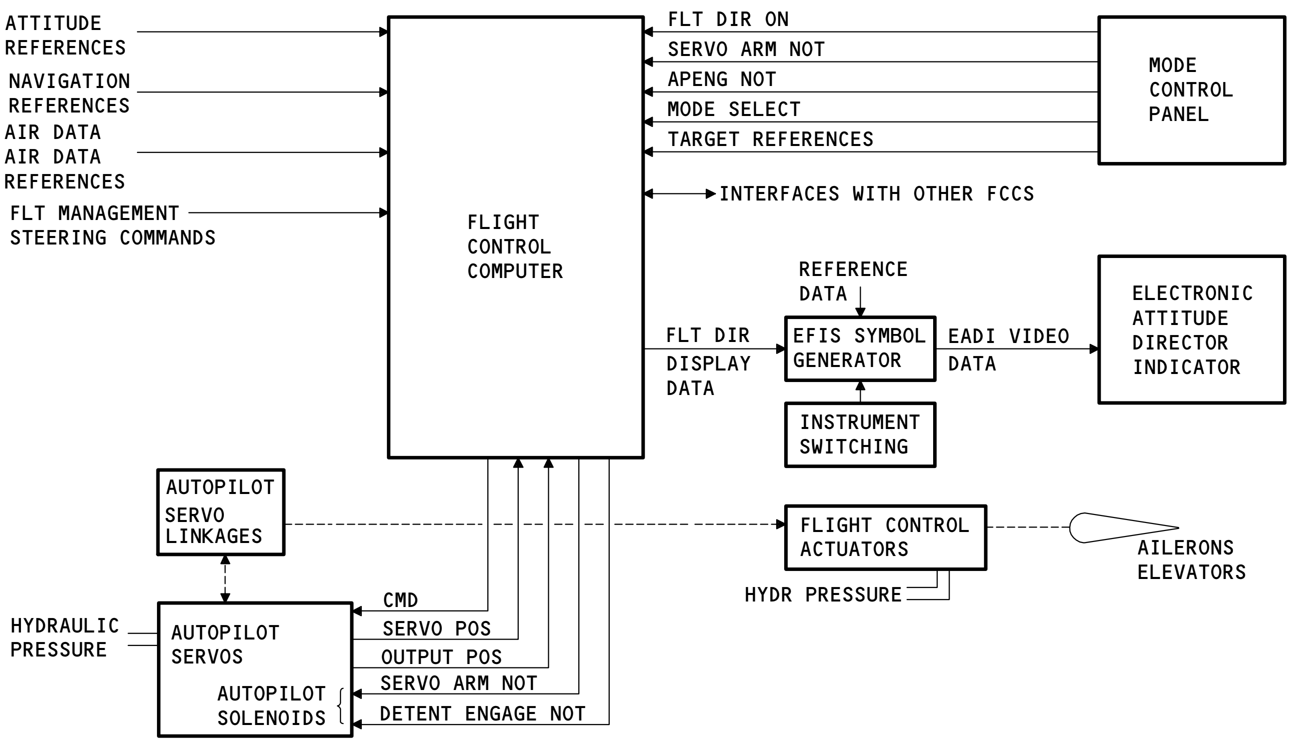
PITCH & ROLL CONFIGURATIONS ARE APPLICABLE TO SINGLE CHANNEL COMMAND ENGAGE OPERATIONS. ROLLOUT CONFIGURATIONS ARE APPLICABLE TO MULTI-CHANNEL COMMAND ENGAGE OPERATIONS, PROVIDING PITCH, ROLL & YAW CONTROL/GUIDANCE OUTPUTS FOR AUTOLAND APPROACHES.

AFDS Configurations
Figure 6

MODE CLASSIFICA-TION	MODES	FUNCTIONS	ANNUNCIATED ON FLIGHT MODE ANNUNCIATOR
SIMULTANEOUS PITCH AND ROLL MODES	CONTROL INACTIVE	NO SERVO OR F/D COMMANDS. ALL CHANGES SYNCHRONIZE ALL CHANNELS MONITORED FOR MODE ENGAGEMENT CAPABILITY	___
	TAKEOFF	F/D ONLY MODE, AUTOMATICALLY SELECTED ON GROUND. ROLL F/D CMDS FOR WINGS LEVEL UNTIL LIFTOFF, THEN TRACK HOLD PITCH F/D CMDS FOR ROTATION ATTITUDE, THEN SAFE CLIMBOUT SPEED	 YES YES
F/D OR CMD	GO AROUND	ROLL CMDS GROUND TRACK HOLD PITCH CMDS FOR SAFE CLIMB RATE USING SPEED SCHEDULE	YES YES
PITCH MODES (F/D OR CMD)	VERTICAL SPEED	BASIC F/D OR A/P CMD SELECTION MODE CONTROLS TO SELECTED V/S	 YES
	ALTITUDE ACTIVE	ALTITUDE HOLD & ALTITUDE SELECT	YES
	FLIGHT LEVEL CHANGE	CONTROLS TO IAS OR MACH SELECTED, DURING CLIMB OR DESCENT TO SELECTED ALTITUDE. PROVIDES FLAP PLACARD AND MINIMUM SPEED PROTECTION	YES
	VNAV MODE	CONTROLS TO VERTICAL STEERING COMMANDS FROM FMC	YES
	GLIDESLOPE ACTIVE	GLIDESLOPE CAPTURE & TRACK	YES
ROLL MODES (F/D OR CMD)	HEADING HOLD	BASIC F/D OR A/P COMMAND SELECTION MODE [1]	YES
	HEADING SELECT	CONTROLS TO SELECTED HEADING	YES
	LNAV MODE	CONTROLS TO HORIZONTAL (LATERAL) STEERING CMDS FROM FMC	YES
	LOC ACTIVE	CAPTURE & TRACK LOCALIZER CENTERLINE	YES

[1] FUNCTION MAY INCLUDE ROLL ATTITUDE HOLD DEPENDING ON LATERAL CMD ENGAGE OPTION

Single Channel Mode Functions
Figure 7



Flight Director Only Block Diagram
Figure 8

(d) The AFDS is used in this condition for takeoff. Only wings level, and rotation target angle guidance is available.

(4) Single Channel Command Engage (Fig. 10)

(a) In the single channel CMD ENG mode, the A/P receives inputs from the MCP, FMC steering commands or pilot-selected navigational commands.

(b) Reference inputs come from the sensor system, FMC or MCP selected target references, position feedback from the A/P servos, and mechanical control feedback to the control wheel and column.

(c) The FCC outputs in CMD ENG include servo commands to A/P servos, stabilizer trim commands, annunciation discretes to the EFIS, and caution and warning discretes.

(d) The FCC in CMD can provide simultaneous Flight Director commands to the EADI if the F/D switch on the MCP is set to ON and F/D source select switch is set to that FCC.

(e) The CMD ENG climb, cruise, and descent control. It can be used for single channel ILS approach control except for the runway alignment, flare, landing, and rollout control.

(5) Multichannel Engage (Autoland) (Fig. 11)

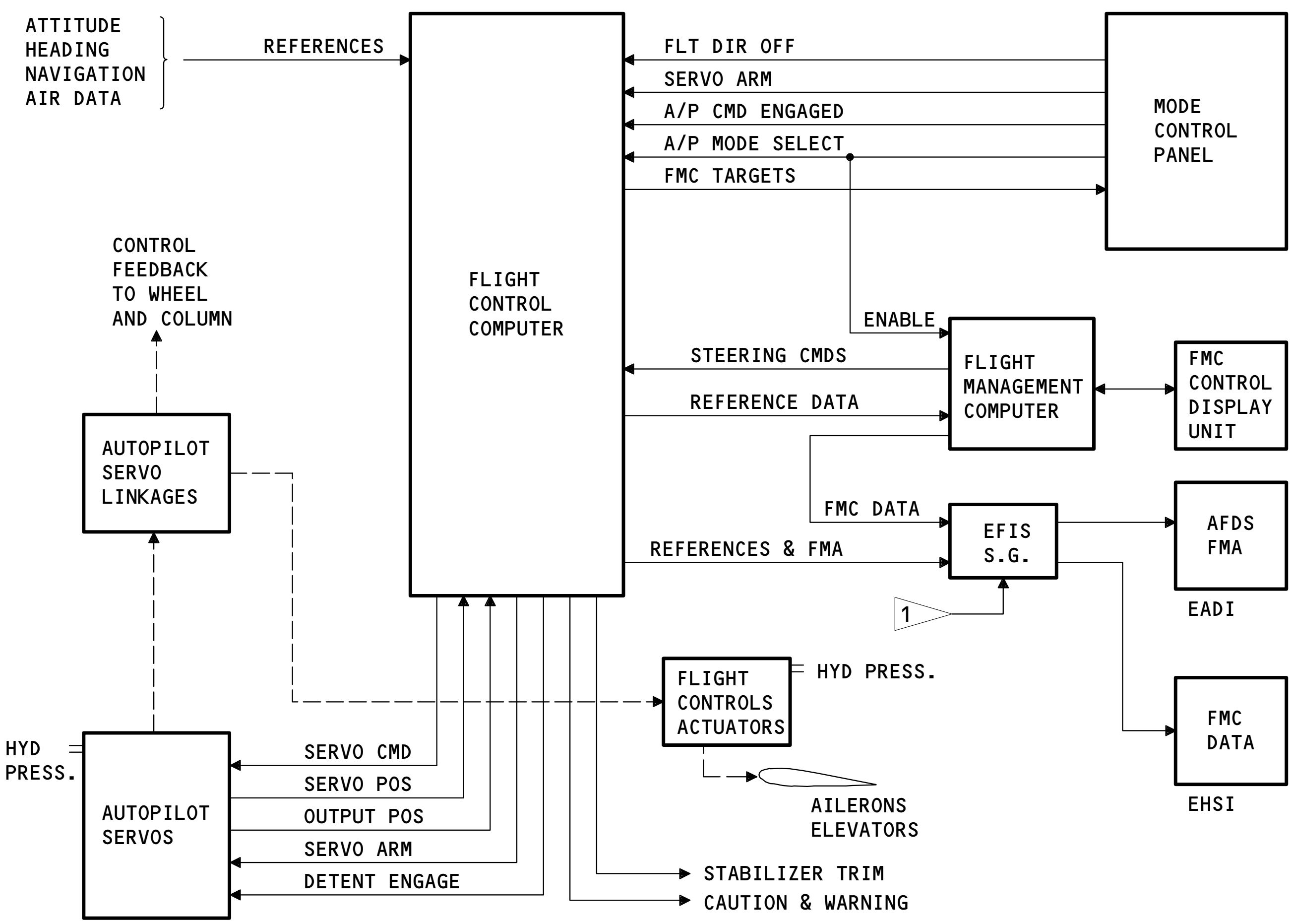
(a) Each FCC has dedicated interfaces with A/P reference sensors, A/P servos (LCCA, DAS, EAS), EFIS symbol generators, the caution and warning system, and an electrical power source. Each FCC receives data from the AFCS MCP and each of the other FCCs.

(b) All A/P servos of one channel are powered from that channel's hydraulic system and provide mechanical inputs to the respective flight control actuator.

(c) Each FCC supplies data to the engine indication and crew alerting system (EICAS), the A/P DISC (red) warning light, the AUTOPILOT (amber) caution light, and the automatic stabilizer trim control system.

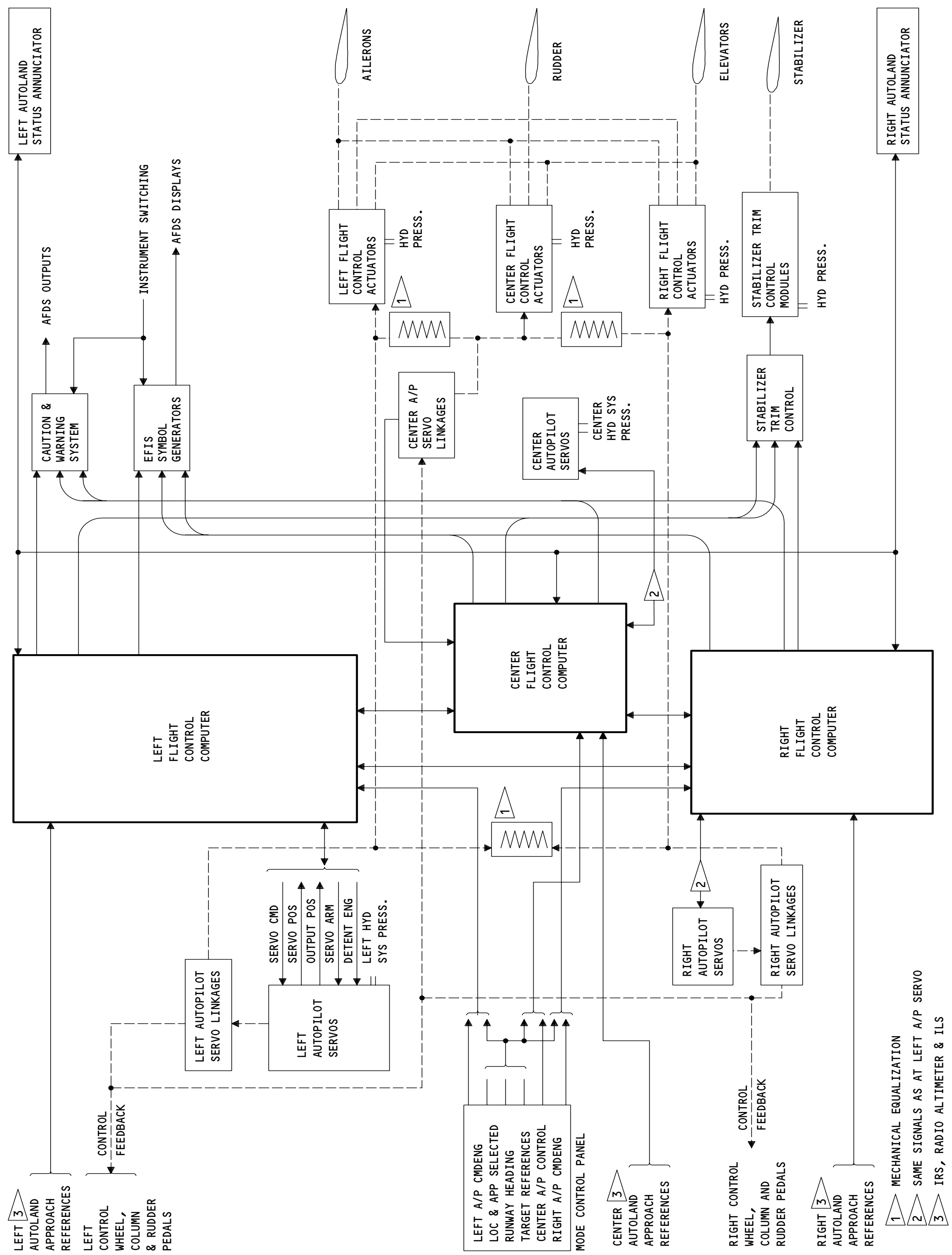
(d) Autoland requires 2 or 3 channels in CMD. Autoland status is indicated on the Autoland Status Annunciator (ASA) as LAND 3 (green), LAND 2 (green), NO LAND 3 (amber), and NO AUTOLAND (amber).

Not Used
Figure 9



1 F/D CMDS AVAILABLE FROM ANY FCC

Single Channel Command Engage Block Diagram
Figure 10



Multi-Channel Engage (Autoland) Block Diagram
Figure 11

(e) The AFDS status is displayed on the EADIs. The AFDS flight mode annunciation (FMA) display indicates roll and pitch modes that are armed or engaged.

(6) Autoland Categories and Configurations (Fig. 12)

(a) The AFDS is capable of controlling the airplane in Category III approaches, depending on the ground facilities available. A typical arrangement is as follows.

1) The G/S transmitter is approximately 900 feet from the runway threshold and transmits a beam with a centerline inclined at a minimum of 2 1/2 degrees to the horizontal.

2) The AFDS controls the airplane to touchdown 500 feet beyond the G/S transmitter. Lateral and landing rollout control is guided by the localizer transmitter at the far end of the runway.

(b) A Category III approach is divided into three sub-categories (A, B, C) according to the runway visual range (RVR) (see Fig. 12).

(c) The Fail Operational configuration has three A/P channels hydraulically engaged in roll, pitch, and yaw. All three use independent data and independent electrical and hydraulic power systems. The ASA shows LAND 3 (green).

(d) The Fail Passive configuration has 2 or 3 channels engaged using two independent sources of data and electrical power, and three hydraulic systems. The ASA shows LAND 2 (green).

(e) Autoland configurations are not active until each FCC recognizes LOC and G/S capture; the airplane is below 1500 feet radio altitude; and other autoland logic requirements are met. Autoland configurations provide control for runway alignment, flare, landing, and rollout.

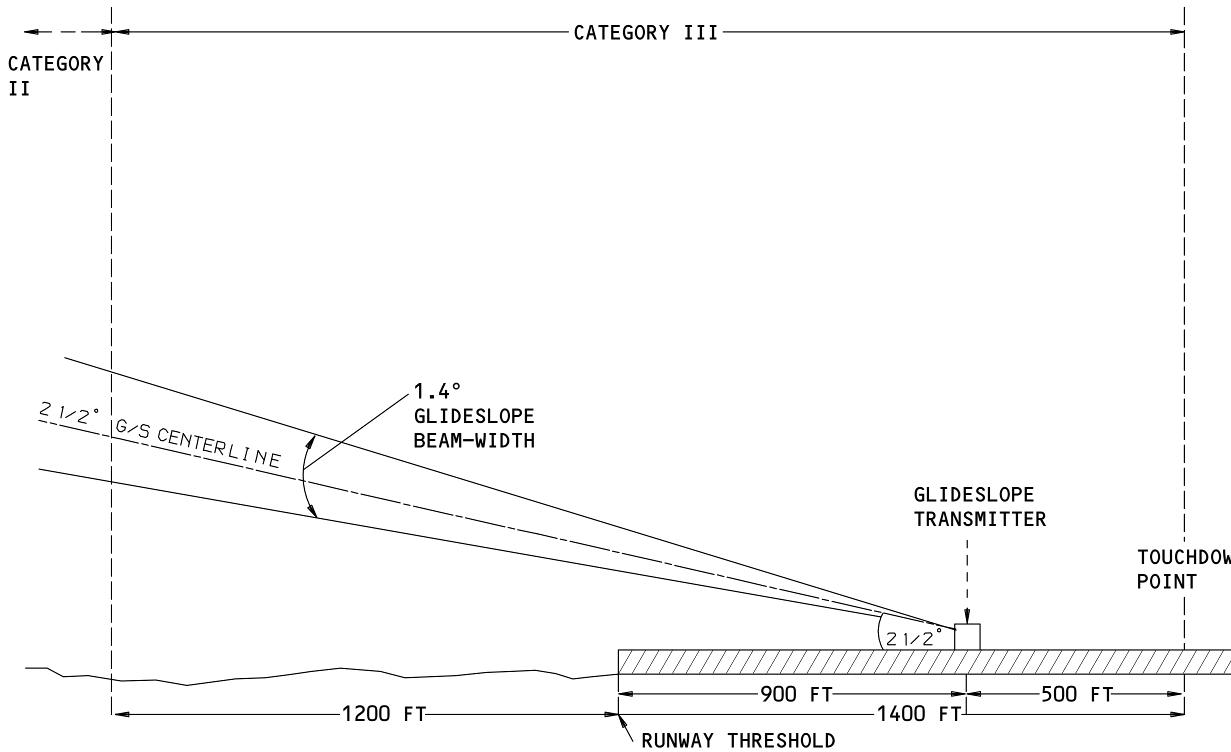
(7) Autoland Functional Sequence (Fig. 13)

(a) The left (L) A/P is CMDENG (pitch and roll) on ILS approach. The LOC has been captured and the A/P is in ALT HOLD prior to glideslope intercept. The center (C) and right (R) FCCs supply captain's and F/O's F/D displays.

(b) The APP mode enables multichannel engage (MCHENG). This automatically arms all A/Ps for MCHENG and makes the L FCC the first channel in CMD. The FMA on the EADI now displays roll and pitch ARM modes.

(c) Multichannel engage occurs when all A/Ps have detected LOC and G/S capture and 1500 feet from a valid radio altimeter (RA). The first channel in command must receive acceptable CMDENG criteria from the other two FCCs. All A/Ps engage pitch, roll, and yaw channels. The ASA displays LAND 3 (Fail Operational) or LAND 2 (Fail Passive). All phases of autoland are possible with LAND 3 or LAND 2 displayed.

(d) If the current first-channel-in-command cannot provide autostabilizer trim control, automatic selection of a new first-in-command is made by a prescribed sequence (i.e., L, R, C, L...).



AUTOLAND CATEGORY	DH	RVR
I	200	≥1800
II	100	≥1200
IIIA	0	≥700
IIIB	0	≥150
IIIC	0	≥0

DH= DECISION HEIGHT
RVR= RUNWAY VISUAL RANGE

Autoland Categories and Configurations
Figure 12

REQUIREMENTS/CONDITIONS FOR FAIL-OPERATIONAL AUTOLAND	LEFT AUTOPILOT			CENTER AUTOPILOT			RIGHT AUTOPILOT		
	ROLL CHANNEL FUNCTIONS	PITCH CHANNEL FUNCTIONS	YAW CHANNEL FUNCTIONS	ROLL CHANNEL FUNCTIONS	PITCH CHANNEL FUNCTIONS	YAW CHANNEL FUNCTIONS	ROLL CHANNEL FUNCTIONS	PITCH CHANNEL FUNCTIONS	YAW CHANNEL FUNCTIONS
L A/P CMDENG • LOC CAPT • [1] GSENG.HR>1500•ALT HOLD. CAPT F/D ON USING CFCC, F/O F/D ON USING RFCC [2]	CONTROLS LEFT LCCA FOR LOCALIZER TRACKING	CONTROLS LEFT A/P ELEV SERVO FOR BARO ALT HOLD	SYNCHRONIZING TO RUDDER POSITION	A/P CH SYNCS TO AIL POSN. SUPPLIES F/D CMDS FOR LOCALIZER TRACKING	A/P CH SYNCS TO ELEVATOR POSN. SUPPLIES F/D CMDS FOR ALT HOLD	SYNCS TO RUDDER POSITION	AS FOR CENTER A/P	AS FOR CENTER A/P	AS FOR CENTER A/P
AS ABOVE, PILOT PRESSES APP SWITCH (METHOD 1 OPTION)	AS ABOVE - L FCC REMAINS CMDENG BUT ALSO IS MULTI-CHANNEL ARM.			AS ABOVE - CENTER A/P AUTOMATICALLY BECOMES MULTI-CHANNEL ARM			AS ABOVE - RIGHT A/P AUTOMATICALLY BECOMES MULTI-CHANNEL ARM		
AS ABOVE, BUT NOW GSENG AND ALT HOLD.	AS ABOVE	CONTROLS LEFT A/P ELEV SERVO FOR GLIDESLOPE CAPTURE & TRACK	AS ABOVE	AS ABOVE	A/P CH SYNCS F/D CMDS FOR G/S CAPT & TRACK	AS ABOVE	AS ABOVE	AS FOR CENTER A/P	AS ABOVE
AS ABOVE, BUT HR < 1500FT. ALL VALID A/PS ARE MCHENG. ROLL & YAW CH ENTER ROLLOUT ARM PITCH CH ENTERS FLARE ARM. ASA SHOWS LAND 3 (GREEN)	AS ABOVE, PLUS LIMITED BANK CMD FOR RUNWAY ALIGNMENT. ENTER ROLLOUT ARM MODE	AS ABOVE, ENTERS FLARE ARM MODE. ADJUSTS PITCH ATTITUDE DURING APPROACH SPEED AND CONFIGURATION CHANGES. REMAINS IN CONTROL OF AUTO STAB TRIM. FOLLOWS C FCC AS FIRST IN CMD	CONTROLS A/P DIRECTIONAL SERVO DURING RUNWAY ALIGNMEMT, PROVIDING YAW IF BANK CANNOT MAINTAIN LOC CENTERLINE	ENGAGES TO CONTROL CENTER LCCA, USING CENTER A/P APPROACH REFERENCES IN ROLLOUT ARM MODE.	ENGAGES TO CONTROL CENTER A/P ELEVATOR SERVO, USING APPROACH REFERENCES IN FLARE ARM MODE. FOLLOWS RIGHT FCC AS FIRST IN CMD FOR AUTO STAB TRIM CONTROL IF RIGHT A/P DISCONNECTS	ENGAGES TO CONTROL CENTER A/P DIRECTIONAL SERVO DURING ROLLOUT ARM MODE.	ENGAGES TO CONTROL RIGHT LCCA USING RIGHT A/P APPROACH REFERENCES IN ROLLOUT ARM MODE.	ENGAGES TO CONTROL RIGHT ELEVATOR SERVO, USING RIGHT A/P REFERENCES IN CMD FOR STAB TRIM CONTROL, IF LEFT A/P DISCONNECTS	ENGAGES TO CONTROL RIGHT A/P DIRECTIONAL SERVO DURING ROLLOUT ARM MODE.
AS ABOVE, BUT WHEN HR<50FT PITCH CH ENTERS FLARE ENGAGE AND GSENG.	AS ABOVE, BUT AILERON COMMANDS REDUCE AS ALTITUDE REDUCES	CONTROLS PITCH ATTITUDE TO REDUCE DESCENT RATE TO 2 FT/SEC AT TOUCHDOWN. TD POINT TARGET IS 1400 FT FROM RUNWAY THRESHOLD APPROX.	AS ABOVE		AS FOR LEFT A/P			AS FOR LEFT A/P	
TOUCHDOWN ROLL CH ENTERS ROLLOUT ENGAGE. PITCH CH DISENGAGES FLARE MODE	COMMANDS WINGS LEVEL	APPLIES NOSE DOWN PITCH CMD SO NOSEWHEEL STEERING CAN BE EFFECTIVE	MAINTAINS RUDDER CONTROL DURING HIGH SPEED SECTOR OF LANDING		AS FOR LEFT A/P			AS FOR LEFT A/P	
LANDING ROLL - CONTROL APPLIED UNTIL AUTOPILOT DISCONNECT	AS ABOVE	AS ABOVE	PROVIDES NOSEWHEEL STEERING CMDS TO MAINTAIN RUNWAY CENTERLINE		AS FOR LEFT A/P			AS FOR LEFT A/P	
GO-AROUND REMAINS MCHENG UNTIL EXIT CONFIGURATION IS COMPLETE	CONTROLS TO HOLD GROUND TRACK	CONTROLS FOR SAFE CLIMB RATE USING SPEED/FLAPS REF	CONTROLS TO HOLD GROUND TRACK		AS FOR LEFT A/P			AS FOR LEFT A/P	

[1] L A/P ASSUMED TO BE CMDENG INITIALLY. C OR R A/P COULD ALSO BE CMD INITIALLY.

[2] ANY FCC CAN BE USED TO SUPPLY F/D DATA

Autoland Functional Sequence
Figure 13

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(e) If a critical sensor fails during a Fail-Operational approach, the ASA shows NO LAND 3, and the FMA displays a mode fault annunciation. When the pilot presses the ASA reset, the AFDS reverts to a FAIL-PASSIVE approach and the ASA displays LAND 2.

(f) The A/P channel having its own dedicated sensor invalid selects a valid equivalent sensor input from the cross-channel data bus and the A/P remains multichannel engaged. If this channel is FIRST IN CMD, it remains FIRST IN CMD. Loss of two similar sensor signals disables autoland capability. The ASA goes blank if radio altitude is above the decision height (DH). The FMA displays fault modes.

(g) Single channel A/P disconnect causes the autoland configuration to revert to FAIL-PASSIVE.

(h) In single channel ILS approach, no runway alignment, flare, rollout control, or rollout guidance is available.

(i) The Autoland configuration is normally disconnected by pressing the A/P disengage switch on either control wheel just before runway turnoff. It is automatically disconnected by an ENGAGE REMOVAL FAULT.

(j) When disconnect warnings occur, they must be reset. The A/P disengage warning reset depends on whether the one-push or two-push option is active. The autopilot remains in multi-channel configuration if Go-Around (GA) is selected. When exit from GA occurs, MCHENG reverts to single channel CMDENG (FIRST IN CMD).

(8) Control and Display

(a) Control

1) One A/P ENGAGE mode, CMD, is available for selection on the Mode Control Panel. Automatic navigation is available from selectable reference sources in single channel CMD engage for climbout, cruise, descent, and ILS approach.

2) Autoland approaches are possible when two or three A/P channels are engaged in CMD.

3) The autopilot is not certified for takeoff. The takeoff mode is a Flight Director only mode.

4) In A/P stand-alone (no F/D is on) operation, Lateral Navigation (LNAV) is done through these modes:

a) Heading Hold

b) Heading Select

c) Localizer Only

d) Approach Guidance and Rollout

5) When coupled with the Flight Management Computers (FMCs), the LNAV mode uses these references:

a) VOR

b) IRS (Inertial Reference System)

c) Stored Flight Plan Information

6) In A/P stand-alone operation, Vertical Navigation (VNAV) is accomplished with the following modes:

a) Pitch altitude hold

b) Autostabilizer trim - through a Stabilizer Trim/Aileron Lockout Module (SAM).

c) Vertical speed select and hold
d) Airspeed/Mach select and hold
e) Go-around
f) Approach guidance, flare, and landing
g) Glideslope

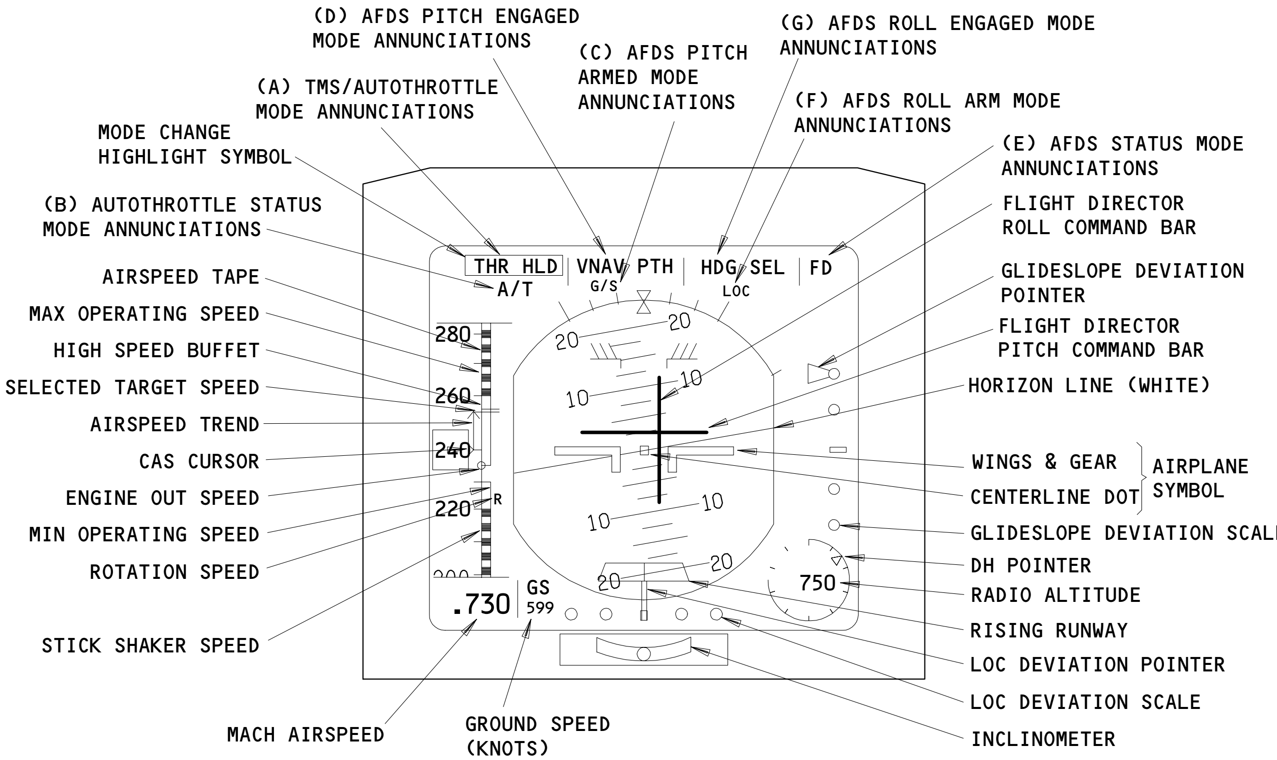
7) When coupled to the FMCS, the VNAV mode uses the following references:
a) Stored flight plan incorporating Standard Terminal Arrival Routes (STARs) and Standard Instrument Departures (SIDs).
b) Computed flight plan depending on the IAS/MACH mode selected.
c) Computed flight plan depending on the FMCs selected mode.
d) IRS.

8) The FCC roll modes do not use any functions associated with Stabilizer Trim and Aileron Lockout Modules (SAM).

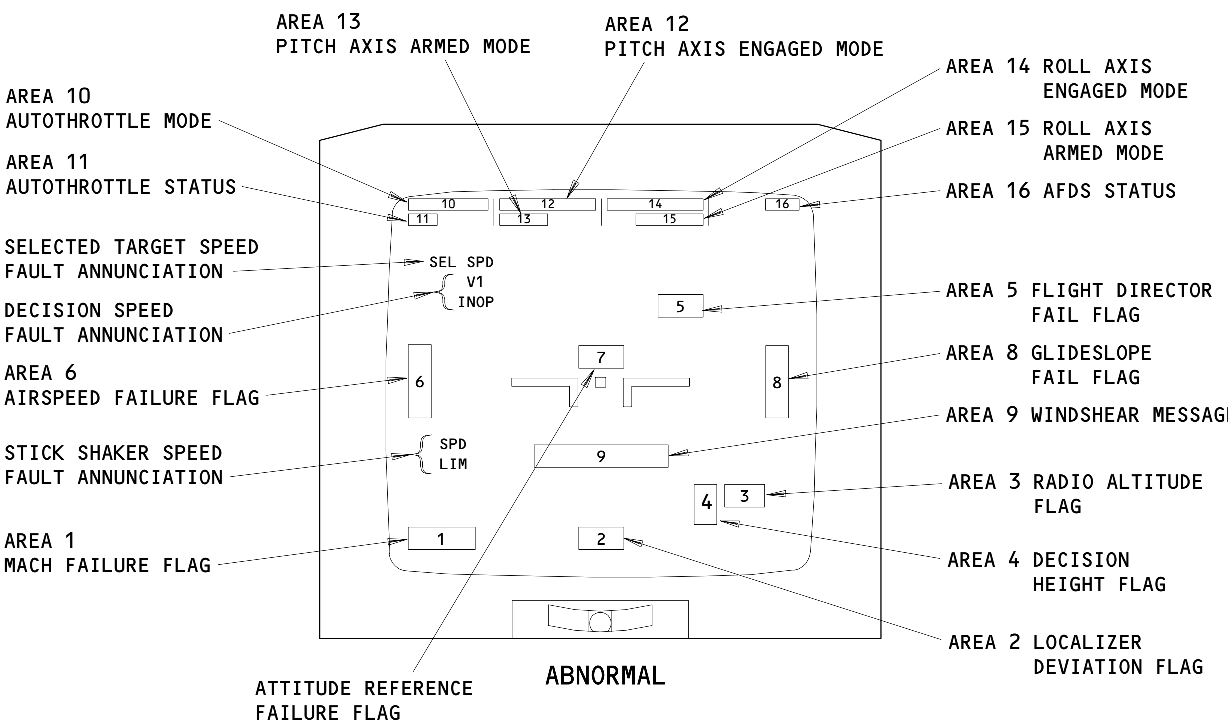
9) The FCC pitch modes use functions associated with SAMs in the following systems:
a) Horizontal Stabilizer Trim System (AMM 27-41-00/501).
b) Automatic Stabilizer Trim System (AMM 22-22-00)

(b) Displays

1) EADI - AFDS Displays - Normal (Fig. 14)
a) The airplane is represented on the EADI as a fixed, white symbol consisting of a center dot flanked by two L-shaped bars. The vertical lines represent the airplane wheels, the horizontal lines, the wings.
b) The roll command bar is a magenta vertical line capable of lateral movement within ± 0.625 inch from the center dot. The roll command scale factor is 30 degrees per inch.
c) The pitch command bar is a magenta horizontal line capable of vertical movement within ± 0.625 inch from the center dot. The pitch command scale factor is 14.3 degrees per inch.
d) If the commanded fly-to point (CFP) is outside the normal deflection range for the command bars, both bars deflect together. They remain intersected at 0.125 inch (minimum) from the bar end to indicate the CFP.
e) Up to 3 alphanumeric characters, 0.167 inch high, identify one of the following AFDS status annunciations: TST, CMD, or FD. These annunciations are green and appear in area (E).
f) Up to 5 alphanumeric characters, 0.123 inch high, identify one of the following armed pitch mode annunciations: G/S, FLARE or VNAV. These annunciations are white and appear in area (C).



NORMAL



ABNORMAL

EADI - AFDS Displays
Figure 14

g) Up to 8 alphanumeric characters, 0.167 inch high, identify one of the following engaged pitch mode annunciations: TO, ALT HOLD, ALT CAP, V/S, VNAV, SPD, SPD LIM, G/S, FLARE, GA, FLAP LIM, VNAV PTH or VNAV SPD. These annunciations are green and appear in area (D).

h) Up to 8 alphanumeric characters, 0.123 inch high, identify one of the following armed roll mode annunciations: LOC, ROLLOUT, LNAV or LOC/LNAV. These annunciations are white and appear in area (F).

i) Up to 8 alphanumeric characters, 0.167 inch high, identify one of the following engaged roll mode annunciations: HDG HOLD, HDG SEL, TRK HOLD, LNAV, LOC, ROLLOUT, ATT, TO or GA. These annunciations are green and appear in area (G).

j) Up to 7 alphanumeric characters, 0.167 inch high, identify one of the following TMS autothrottle operational mode annunciations: N1 (or EPR), SPD, SPD LIM, IDLE, FLP LIM, ALPHA, TEST, THR HOLD, FLCH or GA. These annunciations are green and appear in area (A).

k) Up to 3 alphanumeric characters, 0.167 inch high, identify one of the following TMS status mode annunciations: A/T or F/S. These annunciations are green and appear in area (B).

l) The airspeed tape is a display that appears on the left side of the EADI. It consists of a white calibrated airspeed tape moving against a fixed green computed airspeed (CAS) pointer which represents current CAS. White numerics are placed against the 10 knot indices every 20 knots. Pointers, bugs, and barber poles are displayed against the scale to indicate certain special speeds, as shown below:

MINIMUM FLAP RETRACTION SPEED	GREEN
MINIMUM OPERATING SPEED	YELLOW
STICK SHAKER SPEED	RED/BLACK
MAXIMUM OPERATING SPEED	RED/WHITE
V1 SPEED	GREEN
ENGINE OUT SPEED	CYAN
SELECTED TARGET SPEED	MAGENTA
AIRSPEED TREND	GREEN

m) The localizer deviation scale is at the bottom center of the EADI in white symbols. The standard scale is ±2 dots when A/P or F/D is in LOC ARM ARM mode. The outer dot represents ±0.155 DDM (Difference in Depth of Modulation). An expanded scale of ±0.5 dot is used when the A/P or F/D is in LOC ENG. The outer dot represents 0.388 DDM.

n) The localizer deviation pointer is magenta and appears whenever the LOC deviation scale is in view. It is rectangular and is capable of ±0.73 inch deflection left and right of center. Maximum deviation represents ±0.17 DDM (2.2 dots) against the standard scale and ±0.0485 (-.625 dot) against the expanded scale.

o) The G/S deviation scale is a vertical scale on the right side of the EADI. The symbols are white. The standard scale is ±2 dots when the A/P or F/D is in G/S ARM or G/S ENG. The outer dot represents ±0.175 DDM.

p) The G/S deviation pointer is a magenta colored, truncated triangle capable of vertical movement inside the scale. Maximum deflection is ±0.193 DDM (2.2 dots).

2) EADI - AFDS Displays - Abnormal (Fig. 14)

a) If pitch or roll data is invalid the following display elements are removed: horizon line, pitch attitude scale lines, roll pointer, sky/ground shading and pitch reference line. A yellow, rectangular flag with black letters ATT appears in area 7. A No Computed Data (NCD) condition removes the display elements, but the flag does not appear.

b) Invalid data or the loss of the F/D-on bit, removes the command bars and shows a rectangular FD failure flag (yellow with black letters) in area 5. Command bars are removed if an NCD condition exists; no flag is displayed.

c) Invalid data removes the speed tape and cursor and draws a rectangular SPD failure flag (yellow with black letters) in area 6. An NCD condition removes the pointer but the tape remains.

d) Invalid data removes LOC and/or G/S scales and pointers and draws LOC and/or G/S failure flags in area 2 and area 8 respectively. Failure flags are yellow with black letters. An NCD condition removes the pointer but not the scale.

e) Provision for an ILS Deviation Monitor is available as an option.

f) Invalid and NCD data remove FMA characters of the affected display (areas 10, 11, 12, 13, 14, 15, 16). When an FMA FAULT bit is set in a specific AFDS discrete input word, a horizontal line is drawn through the center of the displayed annunciation.

g) Invalid data removes selected target speed pointer and annunciates SEL SPD on the top left side of the EADI. A NCD condition removes the pointer only.

h) Invalid and NCD data removes V1 pointer and annunciates V1 INOP. This shows as a 2-line message immediately below SEL SPD annunciation.

i) Invalid data removes stick shaker speed barber pole and annunciates SPD LIM. A NCD condition on the ground or in air for less than 10 seconds will remove barber pole. A NCD condition in the air (greater than 10 seconds) will remove barber pole and annunciates SPD LIM on lower left side of EADI.

j) Invalid and NCD data removes airspeed trend arrow from EADI speed tape.

k) Invalid data blanks MACH readout and draws MACH flag (yellow with black letters). A NCD condition draws four dashes in place of numerics.

B. BITE and Monitor

(1) The FCC provides in-flight detection of AFDS and related component failures. Detection includes:

(a) System fault detection and fault location.

(b) Monitoring of servos, sensors, input interfaces and output interfaces.

(c) Self test.

(d) Storage of identified faults.

(2) Fault Detection and Location

(a) Faults are detected through self tests, system monitors within the FCC, and external fault data from interfacing LRUs. The FCC also receives hydraulic valid signals to aid in fault location.

(b) The FCC, in conjunction with the MCDP, can identify the location of an in-flight detected failure to the FCC itself, to a specific interfacing LRU, or to a functional group of LRUs.

(3) Monitoring

(a) The FCC monitors variable and discrete inputs and internally generated valid signals, including:

1) Servo loop operation.
2) Flap and stabilizer position.
3) Hydraulic power.
4) Data bus operation.
5) Status words.
6) Input signal faults.
7) Input signal tolerance.
8) Valid discretes.
9) Internal AC and DC power.

(b) The FCC monitors all incoming and outgoing interfaces, checking data for reasonableness and checking sign status matrix bits on data buses. The FCC also makes cross channel data comparisons with other FCCs.

(4) In-flight Self Test

(a) The in-flight self test is performed by hardware and software independently in each of the FCCs. Safety critical functions are tested at least once a second. Non-critical functions are tested at least once every ten seconds.

(b) Faults detected by FCC self-test are stored in memory for access by shop support equipment. A maximum of five faults can be stored at one time. This fault data can be removed from memory only in the maintenance shop.

(5) Fault Data Storage

(a) The FCC provides permanent storage for up to five internal faults which result in an FCC invalid, and temporary storage for five words of fault data per flight leg.

(b) Fault data in temporary storage is formatted for transmission over a ARINC 429 data bus to the MCDP. The FCC clears temporary data at the start of each flight leg. Data provided to the MCDP includes:

1) Fault identity and location.
2) Source of fault detection.
3) Fault effect on flight deck.
4) Fault intermittent or constant.

(6) MCDP Initiated Testing.

(a) The MCDP decodes and stores, in non-volatile memory, fault data from three flight control computers, one thrust management computer, and two flight management computers. The MCDP can store up to 350 independent faults which were detected either in-flight or during ground testing.

(b) The MCDP outputs test discrete signals to the FCCs to initiate ground testing. The test discrete signals are inhibited when the airplane is not on the ground. The MCDP provides test control and parameters for the FCC ground test.

(c) The FCC tests the AFDS, including:

1) Central Processor Unit
2) Memories
3) Interfacing electronics
4) Power supplies

(d) FCC tests include:

1) Automatic self-test - tests FCC hardware not tested during in-flight automatic self-test.
2) Automatic interface tests.
   a) ILS Tune Inhibit
   b) RA Test Inhibit
   c) Bus Isolate
   d) Autopilot Servos
   e) Automatic Stabilizer Trim
3) Manual Interface Tests
   a) Servo Detents
   b) MCP
   c) ASA
   d) FCC EFIS Display
   e) Analog Data Comparison

(e) The MCDP displays flight faults and ground faults and initiates ground testing (AMM 22-41-00).

C. Control
(1) The autopilot is ON when power to airplane is applied and the following circuit breakers are closed.
(a) MODE CONT PNL L
(b) MODE CONT PNL R
(c) FLT CONT CMPTR PWR L
(d) FLT CONT CMPTR SERVO L
(e) FLT CONT CMPTR SERVO R
(f) FLT CONT CMPTR PWR R
(g) FLT CONT CMPTR PWR C
(h) FLT CONT CMPTR SERVO C
(2) Proper operation requires the following systems be powered and operational.
(a) Thrust Management System (AMM 22-32-00/501).
(b) Electrical Power-Control (AMM 24-22-00/201)
(c) Flight Control System Electronics Units (AMM 27-09-00/201)
(d) Aileron and Aileron Trim Control System (AMM 27-11-00/501)
(e) Rudder and Rudder Trim Control System (AMM 27-21-00/501)
(f) Elevator Control System (AMM 27-31-00/501)
(g) Horizontal Stabilizer Trim Control System (AMM 27-41-00/501)
(h) Leading Edge Slat System (AMM 27-81-00/501)
(i) Hydraulic Power (AMM 29-11-00/501)
(j) Engine Indication and Crew Alerting System (AMM 31-41-00/501)
(k) Air/Ground Relays (AMM 32-09-02/201)
(l) Master Dim and Test (AMM 33-16-00/501)
(m) Air Data Computing System (AMM 34-12-00/501)
(n) Inertial Reference System (AMM 34-21-00/501)
(o) Electronic Flight Instrument System (AMM 34-22-00/501)
(p) Instrument Landing System (AMM 34-31-00/501)
(q) Radio Altimeter System (AMM 34-33-00/501)
(r) Flight Management Computer System (AMM 34-61-00/501)
(3) The autopilot is OFF when airplane power is removed.

AUTOPILOT (FLIGHT CONTROL) - ADJUSTMENT/TEST

1. General
   A. This procedure contains two tasks. The first task is an operational test of the Autopilot/Flight Director System (AFDS). The second task is a system test of the AFDS.
   B. The operational test is to be used without tools or test equipment for a fast check of the operation of the system. The system test uses the MCDP to test the AFDS interfaces and autoland function.

TASK 22-10-00-715-001

2. AFDS Operational Test
   A. General
      (1) This task makes sure the Flight Control Computers (FCCs), the Mode Control Panel (MCP), and the Autoland Status Annunciators (ASAs) operate correctly. This test uses only equipment installed on the airplane.
   B. Equipment
      (1) Hand-held light source
   C. Consumable Materials
      (1) Black tape
   D. References
      (1) AMM 24-22-00/201, Electrical Power - Control
      (2) AMM 29-11-00/201, Main (Left, Center, and Right) Hydraulic Systems
      (3) AMM 31-41-00/501, Engine Indicating and Crew Alerting System (EICAS)
      (4) AMM 31-51-00/601, Warning System
      (5) AMM 33-16-00/501, Master Dim and Test
      (6) AMM 34-12-00/501, Air Data Computing System
      (7) AMM 34-21-00/501, Inertial Reference System
      (8) 34-22-00/501, Electronic Flight Instrument System (EFIS)
      (9) AMM 34-31-00/501, Instrument Landing System
      (10) AMM 34-33-00/501, Radio Altimeter System
      (11) AMM 34-61-00/501, Flight Management Computer System
   E. Access
      (1) Location Zone
          211/212 Flight Compartment

   F. Prepare to Test

      S 865-002
      (1) Check that these circuit breakers on the overhead circuit breaker panel, P11, are closed:
          (a) 11A1, VOR/MKR L

(b) 11A2, ILS CENTER
(c) 11A7, EFIS DSPL SW L
(d) 11A10, AIR DATA CMPTR L
(e) 11A11, AIR DATA AOA SENSOR L
(f) 11A12, AIR DATA BARO CORRECT L
(g) 11A17, AUTOFLIGHT WARN
(h) 11A32, IND LIGHTS TEST
(i) 11B17, CLOCK IND L
(j) 11B18, WARN ELEC B
(k) 11C6, FLT CONT ELEC 1L AC
(l) 11C7, FLT CONT ELEC 1L DC
(m) 11C8, FLT CONT ELEC 2L AC
(n) 11C9, FLT CONT ELEC 2L DC
(o) 11C14, FLAP/STAB POS SENSING C
(p) 11C30, LANDING GEAR POSITION AIR/GND SYS 1
11C29, LANDING GEAR POSITION AIR/GND SYS 2 ALTN
(q) 11C29, LANDING GEAR POSITION AIR/GND SYS 2 ALTN
(r) 11D1, STANDBY BUS AC
(s) 11D2, STANDBY BUS DC
(t) 11D6, CAT III BUS ISOL BAT
(u) 11D17, LEFT ENGINE EEC DISCRETES
(v) 11E3, ADI LEFT
(w) 11E4, EFIS CONT PNL L
(x) 11E6, HSI LEFT
(y) 11E8, FMCS CDU LEFT
(z) 11E9, FMCS CMPTR LEFT
(aa) 11E10, ILS LEFT
(ab) 11E11, DME LEFT
(ac) 11E16, MODE CONT PNL L
(ad) 11E17, FLT CONT CMPTR PWR L
(ae) 11E18, FLT CONT CMPTR SERVO L
(af) 11E20, FLT CONT CMPTR PWR C
(ag) 11E21, FLT CONT CMPTR SERVO C
(ah) 11E24, ADI RIGHT
(ai) 11E25, EFIS CONT PNL RIGHT
(aj) 11E27, HSI RIGHT
(ak) 11E29, FMCS CDU RIGHT
(al) 11E30, FMCS CMPTR RIGHT
(am) 11E31, ILS RIGHT
(an) 11E32, DME RIGHT
(ao) 11E33, VOR RIGHT
(ap) 11E34, MODE CONT PNL R
(aq) 11E35, FLT CONT CMPTR PWR R

(ar) 11E36, FLT CONT CMPTR SERVO R
(as) 11F1, IRS LEFT
(at) 11F5, RAD ALTM LEFT
(au) 11F8, EFIS SYM GEN L
(av) 11F9, EFIS SYM GEN C
(aw) 11F14, TMC AC
(ax) 11F15, TMC DC
(ay) 11F16, TMC SERVO
(az) 11F20, RAD ALTM CENTER
(ba) 11F21, IRS CENTER
(bb) 11F22, IRS RIGHT
(bc) 11F24, EFIS DSPL SW RIGHT
(bd) 11F26, RAD ALTM RIGHT
(be) 11F29, EFIS SYM GEN RIGHT
(bf) 11F30, AIR DATA CMPTR RIGHT
(bg) 11F31, AIR DATA AOA SENSOR RIGHT
(bh) 11F32, AIR DATA BARO CORRECT RIGHT
(bi) 11G17, FLT CONT ELEC 1R AC
(bj) 11G18, FLT CONT ELEC 1R DC
(bk) 11G26, FLT CONT ELEC 2R AC
(bl) 11G27, FLT CONT ELEC 2R DC
(bm) 11J17, FLAP/STAB POS SENSING L
(bn) 11J26, FLAP/STAB POS SENSING R
(bo) 11J34, WARN ELEX A
(bp) 11J36, CLOCK IND R
(bq) 11M32, RIGHT ENGINE EEC DISCRETES
(br) 11P30, LIGHTING INSTRUMENT & PANEL GLARE SHIELD
(bs) 11R3, LEFT IND LIGHTS 3
(bt) 11R5, IND LIGHTS DIM CONT
(bu) 11R29, RIGHT IND LTS 2
(bv) 11R30, RIGHT IND LTS 3
(bw) 11T3, CAT III BUS ISOL L
(bx) 11T30, CAT III BUS ISOL R
(by) 11U9, MAINT CONT DSPL
(bz) 11U15, AIR/GND SYS 1
(ca) 767-300;
11U23, POSITION AIR/GND SYS 2
(cb) 767-200;
11U24, POSITION AIR/GND SYS 2

S 865-003
(2) Supply electrical power (Ref 24-22-00).

S 715-004
(3) To complete this test it is necessary that these systems or components operate correctly. Use references only if and as necessary.
(a) Engine Indicating and Crew Alerting System (EICAS) (AMM 31-41-00/501).
(b) Warning System (Ref 31-51-00).

(c) Master Dim and Test (Ref 33-16-00).
(d) Air Data Computer (AMM 34-12-00/501).
(e) Inertial Reference System (AMM 34-21-00/501).
(f) Flight Instrument System (Ref 34-22-00).
(g) Instrument Landing System (Ref 34-31-00)
(h) Radio Altimeter (Ref 34-33-00).
(i) Flight Management Computer System (Ref 34-61-00).

S 865-005
(4) Align the left, center, and right Inertial Reference Units in NAV Mode (AMM 34-21-00).

<u>NOTE</u>: Approximately 10 minutes are necessary to align the IRU's. The "Prepare To Test" steps that follow can be done during this time.

S 865-006
(5) Make sure the L, C, and R FLT CONTROL SHUTOFF switches on the P61 panel are ON.

S 865-007

<u>WARNING</u>: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(6) Move all persons and equipment away from the control surfaces.

S 865-039
(7) Supply hydraulic power (Ref 29-11-00).

S 865-009
(8) Make sure the MCDP is off.

S 865-010
(9) On the P1 panel, set the captain's instrument source select switches to these positions:
(a) FLT DIR switch to L
(b) NAV switch to FMC-L
(c) ADC, EFI, and IRS switches to normal (ALTN indication on switch does not show)

S 865-011
(10) On the P3 panel, set the first officer's instrument source select switches to these positions:
(a) FLT DIR switch to R

(b) NAV switch to FMC-R
(c) ADC, EFI, and IRS switches to normal (ALTN indication on switch does not show)

S 865-012
(11) On the MCP, set the A/T and the two F/D switches to OFF.

G. Test AFDS Operation

S 715-013
(1) Do these steps to do a check of the A/P DISC light, the AUTOPILOT light, and the MCP switch-lights:
(a) Make sure the A/P DISC and AUTOPILOT lights, on the P1-3 panel, are not on.

NOTE: If the lights are on, push the A/P disengage switch two times to cancel the warning.

(b) On the P5 panel, set the IND LTS DIM/BRT switch on the right lighting control panel to BRT.
(c) On the P5 panel, push and hold the TEST switch on the right lighting control panel to set Master Dim and Test to on.
(d) Make sure the A/P DISC light and the AUTOPILOT light are on.
(e) Make sure the bottom half of the MCP switch-lights (dot-bar matrices) are on.
(f) Release the TEST switch on the right lighting control panel to set Master Dim and Test off.

S 715-014
(2) Do these steps to do a check of the MCP light-plate, switch-lights, and automatic brightness sensor:
(a) On the P5 panel, turn the PANEL/FLOOD GLARESHIELD switch on the left lighting control panel full clockwise (CW).
(b) Make sure the MCP light-plate and the top half of the switch/lights are on.
(c) Turn the PANEL/FLOOD GLARESHIELD switch counterclockwise (CCW).
(d) Make sure the MCP light-plate and switch-lights dim.

(e) AIRPLANES WITH -231 OR -232 MCP;
Do these steps to check the bezel light sensor:
1) Turn the PANEL/FLOOD GLARESHIELD switch an equal distance between fully clockwise and fully counterclockwise.
2) Put tape over the MCP light sensor.
3) Make sure the MCP display backlighting dims.
4) Remove the tape from the MCP light sensor.
5) Make sure the MCP backlighting become brighter.
6) Point a light source at the MCP light sensor.
7) Make sure the MCP backlighting becomes brighter.

S 715-015
(3) Do these steps to do a check of the MCP controls and displays:
(a) Make sure the MCP light-plate is on.
(b) Make sure the MCP digital displays show as follows:

1)	IAS/MACH	200
2)	HDG	000
3)	VERT SPD	Blank
4)	ALT	10,000

NOTE: The MCP digital display values given are only for when the airplane is initially supplied with power. The display values can be different if airplane power has been on since last flight.

(c) Do a check of the IAS/MACH control and display as follows:
1) Turn the IAS/MACH control knob CW to increase the value that shows to 399 (maximum value).
2) Turn the control knob CCW to decrease the value that shows.
3) Turn the control to set the display to 250 (depress the knob if the window is blank).
4) Make sure the speed bugs on the left and right IAS/MACH Airspeed instruments are indicating 245 to 255.
5) Turn the control knob to set the display back to 200.
(d) Do a check of the HDG SEL control and display as follows:
1) Turn the HDG SEL control knob CW to increase the value that shows.
2) Make sure the MCP HDG display value and the selected heading indicator locations on the two EHSIs agree (±1 degree).
3) Turn the HDG SEL control knob to some other values and monitor the results.
4) Turn the control knob to set the display back to the initial heading.
(e) Do a check of the VERT SPD control and display as follows:
1) Turn VERT SPD control UP and DN.
2) Make sure the display does not change.
(f) Do a check of the ALT control and display as follows:
1) Turn the ALT control knob CW to increase the value that shows.
2) Turn the control knob CCW to decrease the value that shows.

3) Turn the control knob to set the display back to 10,000.

S 715-016

(4) Do these steps to do a check of the Autoland Status Annunciator (ASA):
(a) Push and hold the TEST 1 switch on the Captain's ASA.
(b) Make sure the ASA shows LAND 3 (green) on the top display and NO LAND 3 (yellow) on bottom display.
(c) Release the test switch.
(d) Make sure the displays become black.
(e) Push and hold the TEST 2 switch.
(f) Make sure the ASA shows LAND 2 (green) on the top display and NO AUTOLAND (yellow) on the bottom display.
(g) Release the test switch.
(h) Make sure the displays become black.
(i) Do the above steps again for the First Officer's ASA.

S 715-018

(5) Do these steps to do a check of the Flight Director Annunciations:
(a) Set the Captain's and the First Officer's F/D switch on the MCP to ON.
(b) Make sure these indications show:
1) The flight director indicator shows on each display.
2) FD (green letters) shows on each display.
(c) Set the Captain's INSTR SOURCE SEL FLT DIR switch to the C position, then to the R position, and then back to L position. Set the switch in each position for a minimum of 5 seconds.
1) Make sure these indications show at each switch position:
a) The flight director indicator stays on the Captain's display.
b) FD stays on each display.
(d) Set the First Officer's INSTR SOURCE SEL FLT DIR switch to the C position, then to the L position, and then back to the R position. Set the switch in each position for a minimum of 5 seconds.
1) Make sure these indications show at each switch position:
a) The flight director indicator stays on the first officer's display.
b) FD stays on each display.

H. Test Autopilot Engage/Disengage Operation.

S 865-167

(1) Make sure all persons and equipment are away from the control surfaces.

S 865-019

(2) Engage the left autopilot in CMD.

S 755-020

(3) Make sure that HDG HOLD and V/S appear on the Captain's and First Officer's displays.

S 865-021
(4) Push the HDG SEL control knob on the MCP.

S 755-022
(5) Make sure the HDG HOLD indication changed to HDG SEL on the two displays.

S 975-023
(6) Write on paper the heading that shows in the HDG display of the MCP at this time. This is the initial heading.

S 865-024
(7) Turn the HDG SEL control knob to set a heading that is 100 degrees to the left of the initial heading.

S 755-025
(8) Make sure the control wheel moves CCW and the ailerons move in a left-wing-down direction.

S 865-026
(9) Set a heading that is 100 degrees to the right of initial heading.

S 755-027
(10) Make sure the control wheel moves CW and the ailerons move in a right-wing-down direction.

S 865-028
(11) Set HDG display to the initial heading.

S 865-029
(12) Turn the VERT SPD control to set -1000 FPM in the VERT SPD display of the MCP.

S 755-030
(13) Make sure the control column moves forward and the elevator moves in a nose down (trailing edge down) direction in less than 15 seconds.

S 865-031
(14) Set +1000 FPM in the VERT SPD display.

S 755-032
(15) Make sure the control column moves aft and the elevator moves in a nose-up (trailing edge up) direction in less than 15 seconds.

S 865-034
(16) Push the Captain's A/P Disengage switch to disengage the autopilot.

S 865-033
(17) Push the Captain's A/P Disengage switch again to cancel the disengage warning.

S 865-160
(18) Turn the Flight Director OFF then ON.

S 715-035
(19) Do the above steps again to test autopilot engage/disengage operation for the center and right autopilots.

I. Put the Airplane Back to Its Usual Condition.

NOTE: Do not put the airplane back to its usual condition if AFDS System Test is to be done.

S 865-038
(1) Set IRS to OFF if it is not necessary (AMM 34-21-00/501).

S 865-037
(2) Remove the pressure from the left, center, and right hydraulic systems if it is not necessary (Ref 24-22-00).

S 865-036
(3) Remove electrical power if it is not necessary (Ref 24-22-00).

TASK 22-10-00-735-055

3. AFDS System Test

A. General
(1) This task uses the Maintenance Control Display Panel (MCDP) Ground Test procedures to test the AFDS interfaces and autoland function.
(2) The MCDP is found in the main equipment center on the E1 rack. The MCDP control switches and display are found on the front of the MCDP. If a MCDP remote control unit is not to be used, two persons are necessary to do this system test. One person will be in the main equipment center and one person will be in the flight compartment.
(3) A connection is in the flight compartment for a hand-held MCDP remote control unit. This lets MCDP Ground Tests be done from the Flight Compartment. The MCDP data shows on the bottom EICAS display.

(4) Instructions for use of the MCDP are found in the Autoflight BITE - Maintenance Practices (Ref 22-00-02).

B. Equipment
(1) Nose Gear Steering Valve Lockpin - A09003-1
(2) Remote Control Unit, Maintenance Control Display Panel - A22001-22 (preferred), A22001-15 (optional).

C. References
(1) 9-11-00/201, Towing
(2) 22-00-02/201, Autoflight BITE
(3) 24-22-00/201, Electrical Power - Control
(4) 29-11-00/201, Main (Left, Center, and Right) Hydraulic Systems
(5) AMM 31-41-00/501, Engine Indicating and Crew Alerting System (EICAS)
(6) AMM 34-12-00/501, Air Data Computing System
(7) AMM 34-21-00/501, Inertial Reference System
(8) 34-31-00/501, Instrument Landing System

D. Access
(1) Location Zone
211/212 Flight Compartment

E. Prepare to Test

S 865-040
(1) Make sure the circuit breakers shown in the Operational Test are closed before the MCDP is put into GRD TEST mode.

S 485-045
(2) If the MCDP remote control unit is to be used, do the steps that follow:
(a) Install the MCDP remote control unit as follows:
1) On the main power distribution panel, P6, open the P6-5 door.
2) Remove the dust cover from connector D1447 found behind the P6-5 door.
3) Connect the MCDP remote control unit to connector D1447.
4) Close the P6-5 door.
(b) On the P61 panel, push the CONF/MCDP switch on the EICAS MAINT panel to show MCDP data on the EICAS bottom display unit.

NOTE: The same messages that show on the MCDP in the main equipment center will show on the EICAS bottom display unit. The indication MCDP FLT FAULTS or MCDP GRD TEST shows to identify which mode the MCDP is in.

(c) Make sure the EICAS bottom display unit shows the message MCDP OFF.

S 865-053

(3) On the P61 panel, set the L and R EEC MAINT ENG POWER switches to TEST.

NOTE: If the EEC is not supplied with power before the MCDP is put into GRD TEST mode, the messages NO INFC TMC EEC L PRIM (SEC) and NO INFC TMC EEC R PRIM (SEC) will show during the test.

F. Test AFDS

S 715-091

(1) Do AFDS Operational Test.

S 865-092

(2) Remove the pressure from the left, center, and, right hydraulic systems (Ref 29-11-00).

S 715-094

(3) Do MCDP Power Up and Self-Test (Ref 22-00-02).

S 865-093

(4) Set the A/T switch on the MCP to ARM.

S 745-095

(5) Do MCDP Ground Test 30-CURRENT FAULT REPORT.

(a) Make sure that only these messages show during INFC FAULT?

NO INFC TMC EEC L PRI
NO INFC TMC EEC L SEC
NO INFC TMC EEC R PRI
NO INFC TMC EEC R SEC

NOTE: The above messages will not show if you supply power to the EEC before you set the MCDP to the ground test mode.

(b) Make sure that only these messages show during CURRENT OP FAULT?

CURRENT FAULT EEC L PRI
CURRENT FAULT EEC L SEC
CURRENT FAULT EEC R PRI
CURRENT FAULT EEC R SEC

NOTE: The above messages will not show if the engines are in operation.

S 865-101

WARNING: NOSE GEAR STEERING MUST BE LOCKED WHEN RUDDER MOVEMENT CAN OCCUR TO PREVENT INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(6) Move the towing lever on the nose-gear metering-valve module to the TOW POSITION.

S 495-102
(7) Install the nose-gear steering-valve lockpin (Ref 09-11-00).

S 865-109
(8) On the P10 panel, set the L and C STAB TRIM CUTOUT switches to the NORM position.

S 865-103
(9) On the MCP, set switches as follows:
(a) A/T ARM SW switch - ARM
(b) DISENGAGE bar - up position

S 755-105
(10) Make sure the ASA has no indication.

S 715-104
(11) Do the Instrument Landing System (ILS) Operational Test (Ref 34-31-00).
(a) Make sure no failures occur.

S 865-099

(12) It will be necessary during MCDP Ground Test 40-AUTOLAND for the hydraulics to be powered. Pressurization of hydraulics must obey the steps that follow:

(a) Hydraulic power can be supplied by the airplane electrically-driven pumps or by ground hydraulic carts.

(b) To make sure there is sufficient power when the airplane electrically driven pumps are used, operate only the hydraulic devices necessary for this test.

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(c) Move all persons and equipment away from the control surfaces.

S 865-100

(13) Do not power left, right, and center hydraulic systems (Ref 29-11-00) until MCDP message VFY HYD ON shows during MCDP Ground Test 40-AUTOLAND.

S 745-106

(14) Do MCDP Ground Test 40-AUTOLAND.

NOTE: The bottom EICAS display is supplied with power by the right AC BUS. It will be temporarily without power during operation of the R BUS TIE BREAKER (BTB) if APU or external power is used as airplane power. The remote MCDP switches will stay in operation but the MCDP display will operate only in the main equipment center. After the bottom EICAS display starts operation again, it is necessary to push the CONF/MCDP switch again.

(a) Make sure no failure messages show.

S 865-139

(15) Do the steps that follow to clear EICAS maintenance message(s) "STAB TRIM" and/or "YAW DAMPER" if they show:

WARNING: ACCIDENTAL SPOILER MOVEMENT CAN OCCUR WHEN THESE CIRCUIT BREAKERS ARE OPENED. INJURY TO PERSONS CAN OCCUR.

(a) Open these circuit breakers on the P11 panel and then close them in less than 25 seconds:
1) 11C6, FLT CONT ELEC 1L AC
2) 11C8, FLT CONT ELEC 2L AC
3) 11G17, FLT CONT ELEC 1R AC
4) 11G26, FLT CONT ELEC 2R AC

(b) If a faultball stays set to yellow on a Stabilizer Trim/Aileron Lockout Module (SAM) and/or a Yaw Damper Module (YDM) after the above circuit breakers have been opened and closed, push the reset switch on the front of the applicable SAM or YDM.

G. Put the Airplane Back to Its Usual Condition.

S 095-108

WARNING: STAY AWAY FROM THE NOSE GEAR WHEELS WHEN YOU REMOVE THE LOCKPIN. THE NOSE GEAR WHEELS CAN MOVE QUICKLY TO THE CENTER POSITION AND CAUSE INJURY TO PERSONS AND DAMAGE TO EQUIPMENT.

(1) Do these steps to remove the nose-gear steering valve lockpin:
   (a) Make sure the nose gear wheels are in the center position.
   (b) Remove the nose-gear steering valve lockpin.

S 865-137

(2) Switch the MCDP to off.

S 085-110

(3) Do these steps to remove the MCDP remote control panel if it is installed:
   (a) Disconnect the MCDP remote control panel from connector D1447.
   (b) Install dust cover on connector D1447.

S 865-146

(4) On the P61 panel, set the L and R EEC MAINT ENG POWER switches to NORM.

S 865-134

(5) Set the IRS to off if it is not necessary (AMM 34-21-00/501).

S 865-135

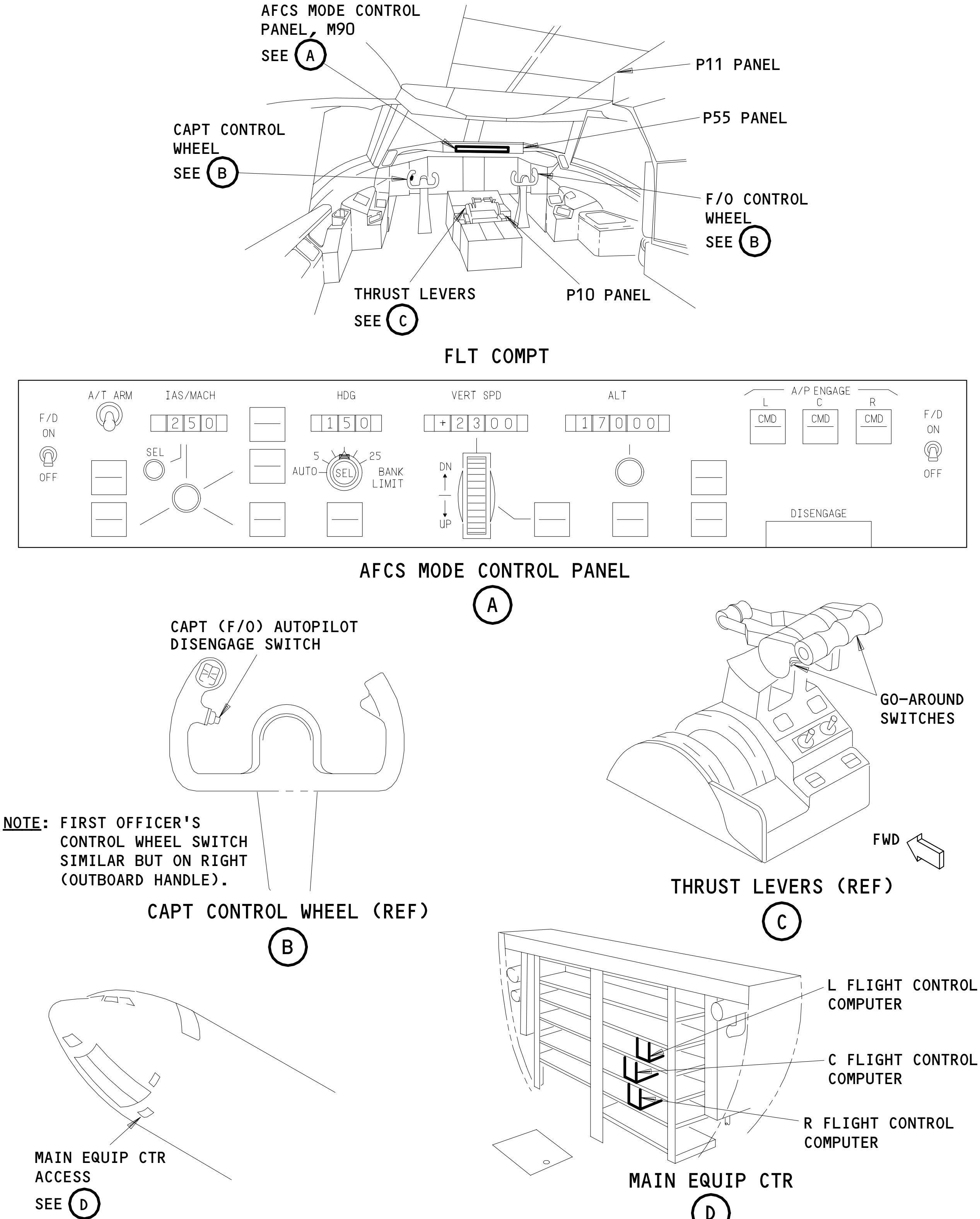
(6) Remove the pressure from the left, center, and right hydraulic systems if it is not necessary (Ref 29-11-00).

S 865-136

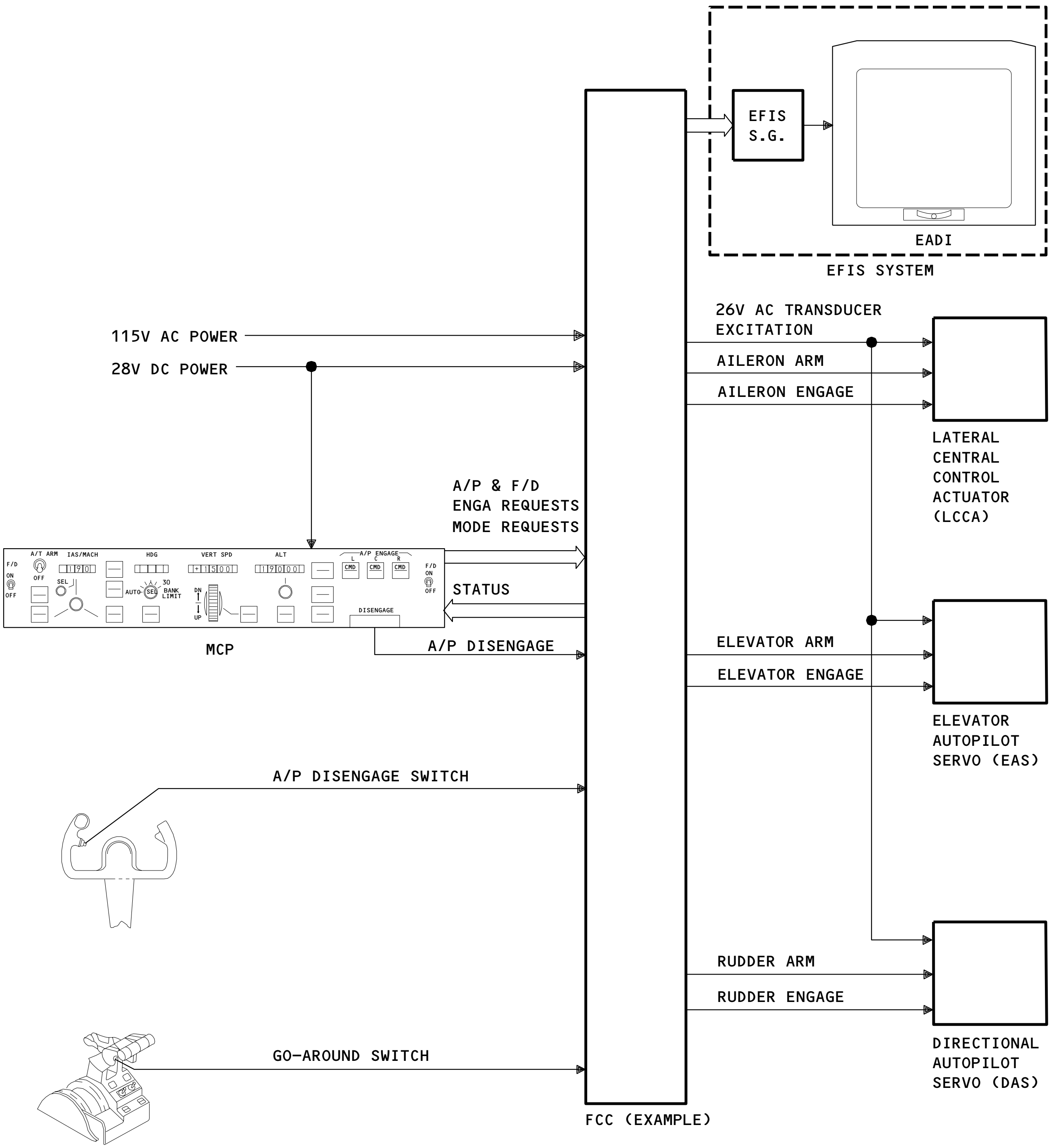
(7) Remove electrical power if it is not necessary (Ref 24-22-00).

AUTOPILOT/FLIGHT DIRECTOR POWER - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. Power to the Autopilot/Flight Director System (AFDS) is provided by the right and left primary power sources (AC) and the right and left Transformer-Rectifier Units (DC). During Autoland operation, the center FCC receives power from the standby inverter and the airplane battery.
   B. The controls and interlocks of the AFDS consist of the following units:
      (1) One Autoflight Control System (AFCS) Mode Control Panel (MCP), located below the glareshield (P55).
      (2) Three Flight Control Computers (FCCs) located in the forward Electrical/Electronics (E/E) compartment in rack-shelf E1-3 (left), E1-4 (center), and E1-5 (right).
      (3) One autopilot disengage switch located on the inboard side of the outboard horn of each control wheel.
      (4) One go-around switch located on the underside of each thrust lever.
      (5) AFDS mode annunciation on the Electronic Attitude Director Indicator (EADI).
   C. Mode Control Panel/Flight Control Computer Interfaces (Fig. 2)
      (1) Autopilot (A/P) and flight director (F/D) engage requests are generated at the Mode Control Panel (MCP) and are supplied to the Flight Control Computers (FCCs). Once accepted, the A/P engage requests can result in the arming and engaging of the A/P servos.
      (2) Mode requests for both A/P and F/D are generated at the MCP and transmitted by digital data bus to the FCCs.
      (3) Flight director engage requests will be automatically generated by the FCC's in the go-around mode. This option supports windshear capabilities and is enabled by grounding the FCC program pin F/D AUTOMATIC ON.
      (4) The A/P disengage bar logic tells the FCC if the bar is up or down. This allows the FCC to enable or disable the A/P servos.
      (5) The FCCs provide engage and mode status to the MCP. Mode status is indicated by the dot-bar matrix on the A/P engage and mode select switch/lights. Mode status is also indicated on the EADIs (Ref 22-14-00).
   D. FCC Input Discretes From Remote Switches
      (1) Two A/P disengage switches (one on each control wheel) provide discrete signals to the FCCs which disengage the autopilot. The go-around switches (one on each thrust lever) provide discretes to the FCCs to start the Go-Around (GA) mode.



Autopilot/Flight Director System Component Location
Figure 1



AFDS Power and Engage Interlocks
Figure 2

E. AFDS Input Power

(1) The FCCs receive 28 vdc and 115 vac, 400 Hz power for internal use. The 115 vac is also reduced to 26 vac in the FCC and supplied to the A/P servos (LCCA, DAS, and EAS). The MCP power supply receives 28 vdc. Both the MCP and the FCC use 28 vdc for the actuator and servo arm, engage, and disengage functions.

2. Component Details

A. AFCS Mode Control Panel (Fig. 3)

(1) The AFCS Mode Control Panel provides the primary interface between the flight crew and the autopilot/flight director and thrust management (TMS) systems. It contains switches and logic circuits for mode selectors and control of all A/P, F/D, and TMS autothrottle (A/T) functions.

(2) Autopilot Engage/Disengage Switches

(a) The MCP has 3 pushbutton switch/lights for manual selection of the L, C, and R FCCs' in the CMD mode. Switch/light legends are lighted whenever power is applied to the MCP. A dot-bar matrix on each switch/light is lit by FCC logic when the request has been accepted by the FCC.

(b) The A/P disengage bar is a rocker switch with contacts that interface with all three FCCs. It provides positive disengagement of all A/P servos. A dayglow-orange stripe on top of the bar gives a clear indication of the disengaged state.

(3) Autothrottle Arm Switch

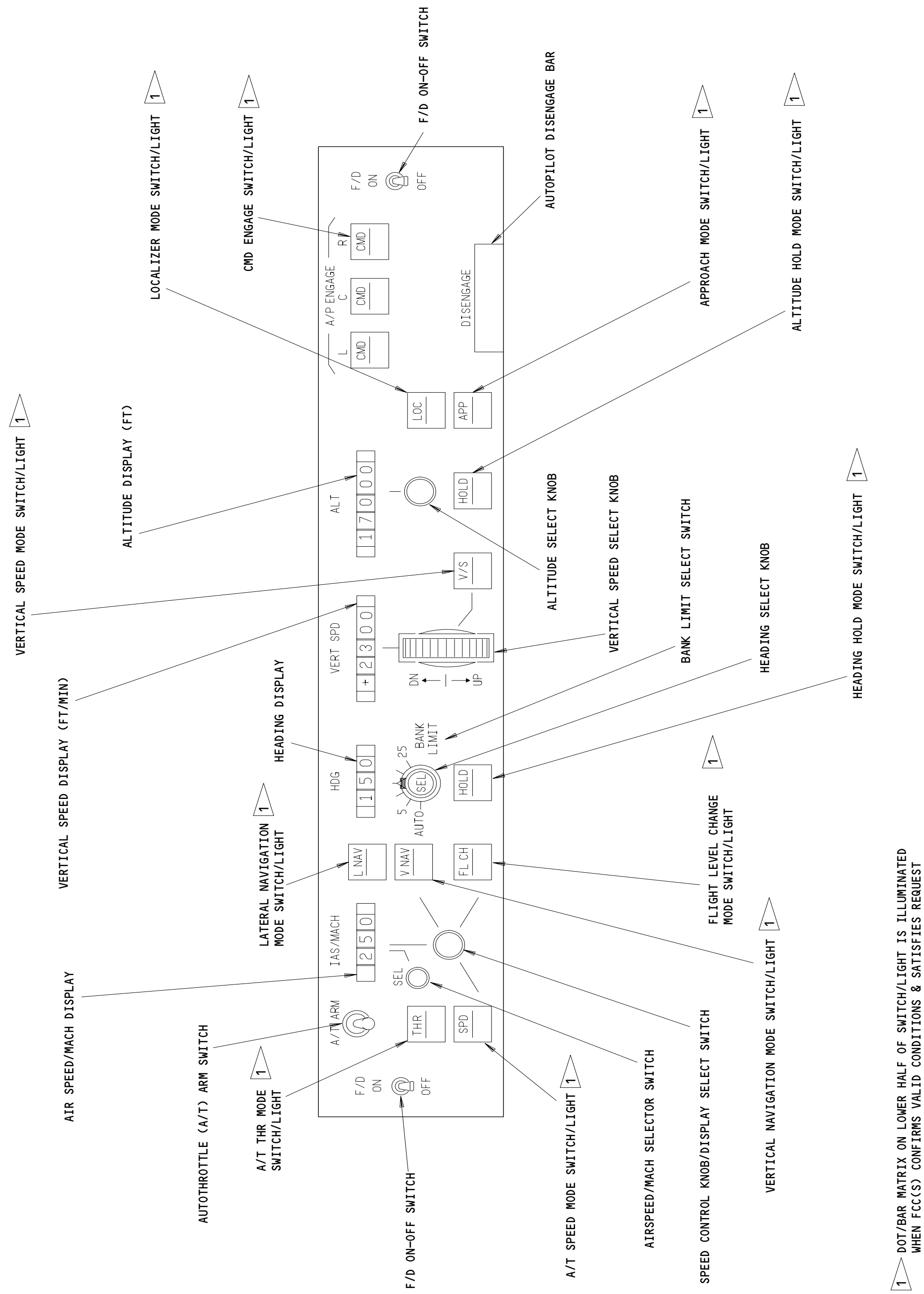
(a) The Autothrottle (A/T) toggle switch allows arming and disarming of the TMS for engagement in the thrust (THR) or speed (SPD) mode.

(4) Autothrottle Mode Engage Switches

(a) Two pushbutton switch/lights provide manual selection of the SPD and THR modes. Contacts of these switches interface with the Thrust Management Computer (AMM 22-31-00/001). The dot-bar matrix, controlled by TMC logic, lights when the mode is engaged.

(5) Flight Director Switches

(a) The flight director system is turned on/off by either of two F/D toggle switches on the MCP. The left F/D switch controls the captain's EADI command bars. The right F/D switch controls the first officer's EADI command bars.



AFCS Mode Control Panel
Figure 3

(6) Autopilot/Flight Director Mode Select Controls

(a) The MCP has eight pushbutton switch/lights for manual selection of the following AFDS modes:

1) Vertical Speed - V/S
2) Altitude Hold - ALT HLD
3) Vertical Navigation - VNAV
4) Flight Level Change - FLCH
5) Heading Hold - HDG HLD
6) Lateral Navigation - LNAV
7) Localizer - LOC
8) Approach - APP

(b) Each switch/light operates switch contacts that interface with all three FCCs. Dot-bar illumination is controlled by FCC logic.

(c) A pushbutton switch labeled SEL, allows manual selection of either airspeed or mach for FLCH (flight level change) or A/T SPD (speed) modes. Switch contacts interface with the FCCs and the TMC.

(d) The HDG SEL (heading select) inner knob assembly contains a push action switch for manual selection of the HDG SEL mode. The switch contacts interface with the FCCs.

(7) Heading Reference Display and Control

(a) The heading reference portion of the MCP consists of a magnetic wheel heading display window and a rotary control with two concentric knobs. The display window is capable of showing 000 to 359 degrees in increments of one degree. The display is initialized to 000 when power is first applied to the MCP. At LOC capture, the display is initialized to the selected runway heading from the FCC.

(b) The outer knob of the rotary control has six positions for selection of APFD bank angle limits (5, 10, 15, 20, 25, AUTO). The knob position is digitally transmitted to the FCCs. The inner knob increments the heading display when rotated (CW increases valve). This heading reference is optically sensed to drive the decoders and then digitally transmitted to the FCCs. The HDG SEL function was discussed under mode select.

(8) Vertical Speed Reference Display and Control

(a) The vertical speed (V/S) portion of the MCP consists of a magnetic wheel display window and a thumbwheel control. The display window displays V/S from +6000 to -8000 ft/min in 100 ft/min increments. When V/S mode is not selected, the display shows zero. With V/S active, the display shows the vertical speed at the time the mode was selected. (If V/S is zero ft/min, A/P goes into an altitude hold control while in V/S mode.) The V/S reference value is optically sensed to drive the decoders and then digitally transmitted to the FCCs.

(b) The thumbwheel is used to select a given vertical speed. Upward rotation of the thumbwheel decreases the selected value, downward rotation increases it.

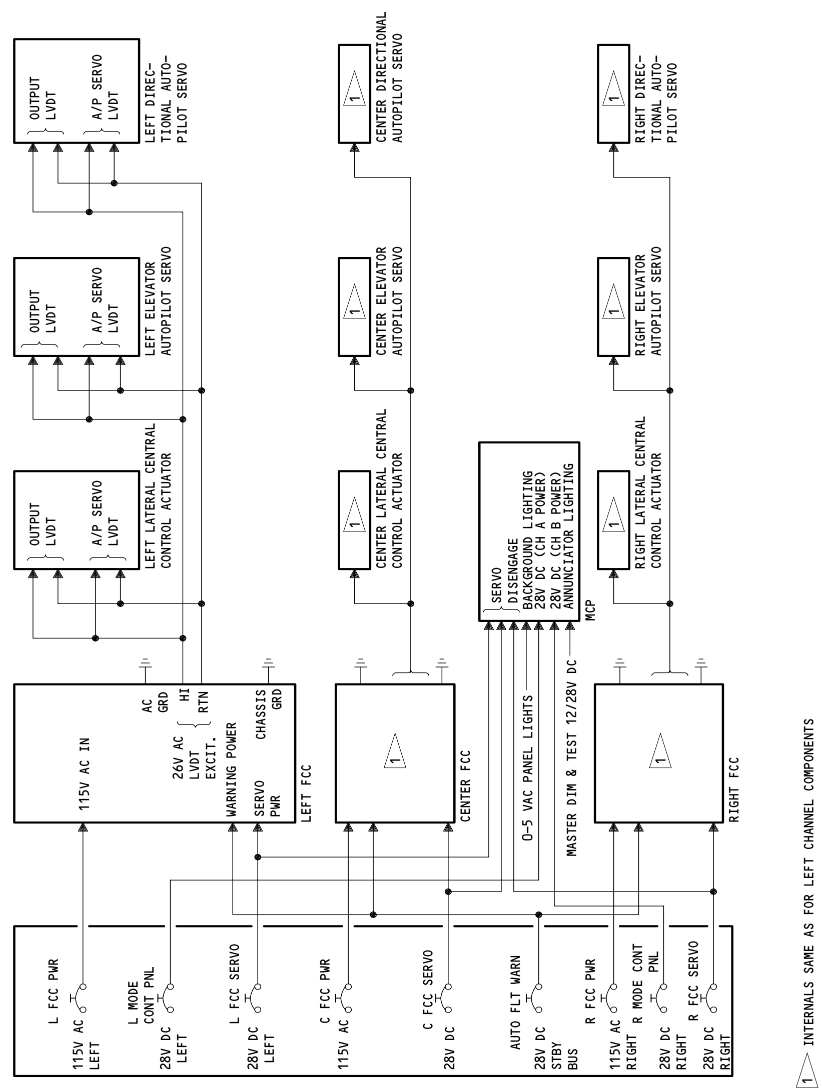
(9) Altitude Reference Display and Control
(a) The altitude select portion of the MCP consists of a display window and a rotary select knob. The display can show 0 to 50,000 feet in increments of 100 feet. When power is applied to the MCP, the altitude window is set to 10,000 feet. The rotary knob increments the display when rotated (CW increases value). The altitude reference is digitally transmitted to the FCCs, the FMCs, and the warning electronics module.
(10) IAS/MACH Reference Display and Control
(a) The indicated airspeed (IAS) reference section of the MCP consists of a rotary knob with an integral pushbutton switch and an IAS/MACH display window. Display limits are 100 to 399 knots in increments of one knot, and 0.40 to 0.95 Mach in increments of 0.01 Mach.
(b) If the FCC airspeed is invalid, the IAS/MACH window displays 200 knots. The display is blanked when the VNAV mode becomes active or the speed control switch in the knob is pressed while speed (or Mach) is displayed. If the switch is pushed when either A/T SPD, FLCH or F/D takeoff mode is active, the display does not change.
(c) The airspeed and Mach number reference values are stored in the MCP. The reference valves are transmitted in digital form to the TMC, the FCCs, and the mach/airspeed indicators (to drive the airspeed reference cursors).

B. Flight Control Computer
(1) The FCC is an 8 MCU component with a max weight of 28.1 lb. It is cooled by blow-through air and has a single shell electrical connector per ARINC 600. Two removable side covers permit access to all circuit cards and modules.
(2) Autopilot outputs of the FCC include arm, engage, and disengage commands to the A/P servos; system engagement and autoland status annunciations; and aural and visual warnings of system disconnect.
(3) The FCC generates steering signals for the F/D command bars. The FCC command source selection is accomplished externally by the instrument source select switch.
(4) The FCC provides automatic stabilizer trim commands to the stabilizer trim and aileron lockout module (SAM).
(5) The FCC use a set of controls laws which are combined, as needed, to perform the following functions:
(a) Vertical position control (used for altitude select, altitude hold, and for glideslope and flare control during approach).
(b) Airspeed control (used for flight level change, go-around, and pitch takeoff flight director; speed protection for minimum speed, flap placards and Vmo/Mmo is provided for these modes).
(c) Vertical speed control.

(d) Heading/track control (heading select and hold in cruise, and track hold hand for go-around and takeoff flight director).
(e) Automatic landing.

C. Autopilot Disengage and Go-Around Switches
(1) Autopilot Disengage Switches
(a) The A/P disengage switches are single action, multiple pole, pushbutton switches.
(b) Pressing either A/P disengage switch disconnects any engaged FCC. The A/P disengage switch must be pressed a second time to reset the A/P disengage warning.
(2) Go-Around Switches
(a) Each palm actuated GA switch contains a set of contacts for each FCC. The switch provides discrete logic used in the GA mode.

3. Operation
A. Functional Description
(1) Autopilot/Flight Director System Power
(a) General
1) Power for the Autopilot/Flight Director System is provided by the engine or APU driven generators via the left and right AC buses. The left and right Transformer Rectifier Units (TRU's) provide DC power to the left and right DC buses respectively. During normal operation, airplane cruise or takeoff, the center AC and DC buses are powered by the left buses. During approach for landing with triple channel autoland operation, center AC and DC buses are powered by the static inverter and hot battery bus. Center bus switching is provided by the FCC's internal isolation request and system relay logic. The switching provides each FCC with an independant power source during autoland operation.
(b) Autopilot/Flight director System Power Distribution (Fig. 4)
1) AC Power
a) The 115v ac power supplied to the FCCs is used by the internal power supply and is also reduced to 26v ac for use by the LVDTs in the flight control actuators (LCCAs, DASs & EASs). The MCP also receives 5v ac power for background lighting from the glareshield panel lights dimmer control unit.



Autopilot/Flight Director System Power Distribution
Figure 4

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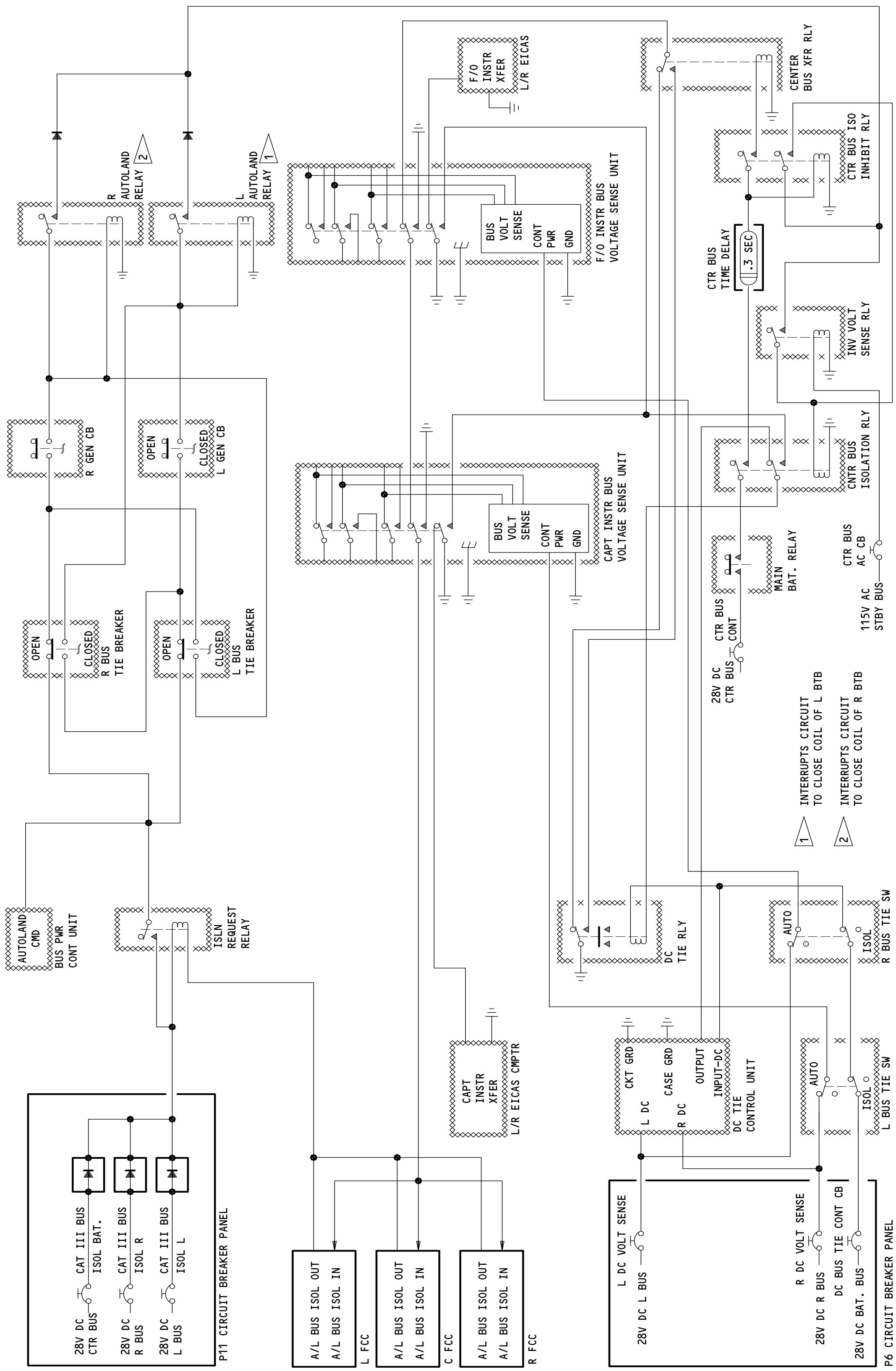
2) DC Power
   a) Each FCC receives dual 28 vdc power (1 from standby power) for monitor circuits for caution and warning alerts. The FCCs receive 28 vdc servo power for servo arming and disarming. The MCP receives the 28 vdc for the A/P disengage bar. A variable dc (12v or 28v) is provided to the MCP from master dim & test for annunciator illumination.

(c) AC Power
1) The FCC power supply receives 115 vac power and generates dc voltages required for the electronics circuits. The 115 vac is also reduced to 26 vac to provide servo excitation.
2) The 26 vac excitation power is supplied to the following transducers:
   a) Aileron, elevator, and rudder surface position transducers
   b) Aileron, elevator, and rudder servo position transducers.

(d) DC Power
1) Standby 28 vdc power is supplied to the FCC as an independent power source for caution alert and warning monitor circuits.
2) Left MCP 28 vdc power is supplied to the MCP and Captain's Autoland Status Annunciator (Ref 22-14-00).
3) Left FCC 28 vdc servo power is used to power the aileron, elevator, and rudder arm and engage solenoids if the following conditions are satisfied:
   a) The MCP DISENGAGE bar is up.
   b) The FCC logic is satisfied to complete aileron, elevator, and/or rudder arming or engaging.

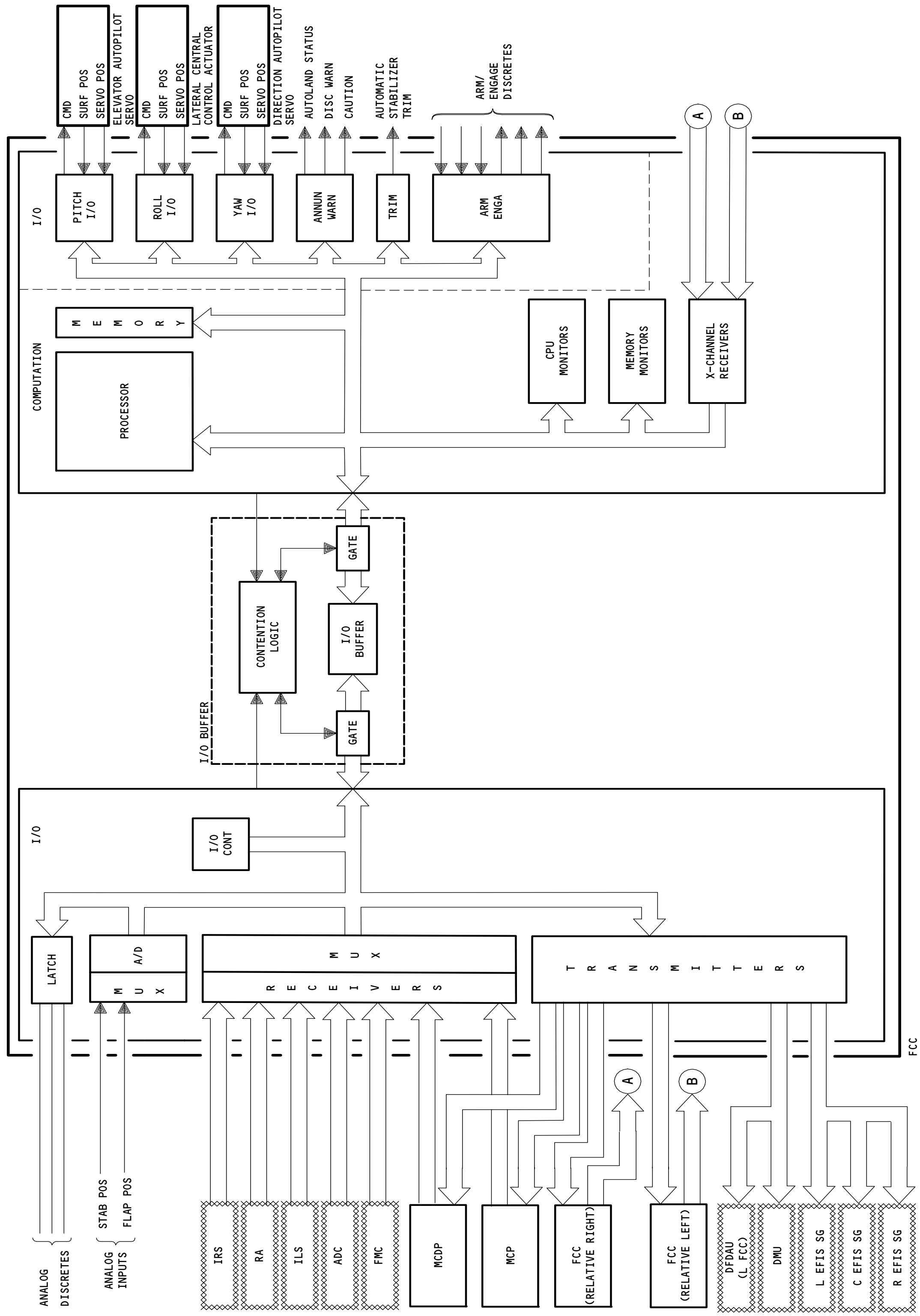
(e) Autopilot Flight Director System Autoland Bus Isolation (Fig. 5)
1) Three independent AC and DC power sources are required to initiate a fail-operational Category III landing. The main engine-driven generators are the normal sources for two of the independent sources and the standby power system (battery and static inverter) provides the third. If one engine-driven generator is inoperative, the APU-driven generator may be substituted for that source.
2) The L and C FCCs are supplied from the left power system and the R FCC from the right power system. The bus tie breakers (BTBs) are normally open.



AFDS Autoland Bus Isolation Schematic
Figure 5

3) If the APU is running prior to a generator failure, the auxiliary power breaker (APB) and the associated BTB will close allowing the APU generator to power the failed generator's channel. If the APU is not running, both BTBs will close so the opposite engine-driven generator can assume both bus loads (Ref 24-22-00).
4) Bus isolation discrete is issued when a ground is generated at the FCC's A/L BUS ISOL inputs. The ground is generated when:
   a) The approach (APP) mode is selected.
   b) The A/P is single channel engaged, multi-channel armed (offline channels are automatically armed by the FCCs when APP is selected) in the CMD mode.
   c) Autoland Status Annunciator (ASA) is <u>not</u> displaying NO LAND 3.
5) The ground allows the ISLN REQUEST RELAY (K122) to energize thereby providing 28 vdc to the AUTOLAND RELAYS (K526 and K527). The 28 vdc energizes both relays which in turn provides 28 vdc to energize the CENTER BUS ISOL RELAY (K123). With K123 energized, the MAIN BAT RELAY (K104) provides power to energize the CENTER BUS XFER RELAY (K107). This removes the ground to the FCC's A/L BUS ISOL IN pin. The open signals the FCC's that the center buses are now powered from the hot battery bus (DC) and static inverter (AC). If either the left or right INSTR BUS VOLTAGE SENSE UNIT (M1079 or M1217) de-energizes, then the ground to the FCC's A/L BUS ISOL in pin is completed.
6) The autoland bus isolation logic maintains the center AFDS on standby bus power for the duration of the approach unless reset by one of the following conditions:
   a) ASA shows NO LAND 3
   b) Total autopilot disconnect.
   c) Go-around has been initiated.
7) Center bus isolation is latched when the following conditions are true.
   a) Autopilot is engaged
   b) The airplane is below alert height and not in the G/A mode.
   c) Center bus isolation was previously set.

(2) AFDS FCC Operation
   (a) FCC Functional Partitioning (Fig. 6)



Flight Control Computer Functional Partitioning Schematic
Figure 6

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1) The FCC is divided into two separate sections; one section is controlled by the CAPS processor and the other section is controlled by the I/O controller. The section under the I/O controller provides input buffering, signal conversion, and storage of analog, digital and discrete signals received from external sensors and control switches. The section under the CAPS processor uses the CAPS transfer bus to provide processor related functions such as memory access, computation of analog servo commands, processing of discrete input information, monitoring functions, and processing of cross channel receiver information. The CAPS processor section also contains and implements the primary application program for the FCC. The link between the I/O controller section and the CAPS processor section is provided by the I/O buffer. Data words may be stored or retrieved by either the I/O controller or the CAPS processor, with the storage location based on the source and contents of the data word. The I/O buffer also contains three internally generated status bits for each parameter. The status bits are read by the processor in order to determine the status of the data.

(b) ARINC 429 I/O

1) The ARINC 429 I/O assembly provides an interface between the internal I/O bus and the ARINC 429 input and output buses. The assembly handles seven ARINC 429 input buses from external sources, and transmits data to external units via three output buses. Three ARINC 429 input ports are also provided for FCC internal signal wraparound to facilitate built-in test requirements.

(c) Digital Discrete Input

1) The digital discrete assembly provides an interface between the internal FCC I/O bus and discrete inputs. The assembly also receives one ARINC 429 input (from IRS) and provides three ARINC 429 outputs (to the MCP, FDAU, and EFIS). Data transfer is directed by the I/O controller.

(d) Analog Input

1) The analog input assembly provides for the conversion of analog input signals into digital words for use in FCC mode control computations. Internal test voltages are monitored for software scaling and fault detection, and wraparound signals from pitch servo assembly and roll/yaw servo assembly are converted to digital words for use in software monitoring routines. The converted inputs are multiplexed onto the I/O data bus as two 8-bit words.

(e) Roll/Yaw Servo

1) The roll/yaw servo assembly provides an interface between the CAPS processor and the aileron and rudder hydraulic servo valves. Servo commands from the CAPS processor are converted to analog signals and then summed with the present servo position signal to provide the drive for the airplane hydraulic servo valves.

(f) Discrete/Warn I/O
  1) The discrete/warn I/O assembly provides discrete-to-TTL level translation of +28 volt discrete signals for use by the FCC processors. The assembly also monitors the status of the autopilot servo engage signals, and provides warning outputs if a problem is detected.

(g) Monitor
  1) The monitor assembly provides many of the monitor and miscellaneous system functions. These functions include computation and iteration monitoring, system clocks, real-time interrupts, transfer bus termination logic, and two non-volatile fault indicators (used to provide an indication of failures in the CAPS processor or power supply).

(h) Pitch Servo
  1) The pitch servo assembly provides an interface between the CAPS processor and the elevator servo loop. Servo commands from the CAPS processor are converted to analog signals and then summed with the present servo position signal to provide the drive for the airplane hydraulic servo valves.

(i) Engage/Warn
  1) The engage/warn assembly provides an interface between the CAPS processor and the autopilot servo engage solenoids. The assembly also provides discrete-to-TTL level translation of elevator, aileron, and rudder engage signals, and the left and right autopilot engage signals for use by the CAPS processor. Window comparators are provided to monitor the power supply voltages.

(j) Cross Channel Receiver
  1) The cross channel receiver assembly provides the FCC with an input buffer for data from the other two FCC's in the system. The input data is received on ARINC 429 buses and converted to parallel words for use by the CAPS processor. Separate input buffers are maintained for each of the other FCC's in the system (Ref 22-15-00).

(k) Read/Write Memory
  1) The read/write memory assembly provides 4k words of high speed CMOS RAM for use by the CAPS processor as scratchpad memory. The assembly also provides 32 words of non-volatile memory for storage of fault information. In addition to the memory circuits, the assembly provides transfer bus logic decoding for up to 13 outputs. These outputs are used to enable devices which require no more than four transfer bus addresses.

(l) Arm/Warn
  1) The arm/warn assembly provides an interface between the CAPS processor and the autopilot servo arm solenoids. Circuitry is also provided for half of the warn output and for half of the dual FCC power supply monitor. Additional circuits allow the CAPS processor to examine the status of interface data for verification or safety reasons.

(m) I/O Control

1) The I/O control assembly provides the processing control functions for the various I/O interface assemblies. The assembly services the I/O assemblies by any of three methods: unconditional periodic service, conditional periodic service, or interrupt service. The controller operates independent of the CAPS processor and communicates with the CAPS through I/O data path assembly or by means of an interrupt.

(n) I/O Data Path

1) The I/O data path assembly provides an interface between the I/O processor and the CAPS processor. The I/O data parameters are stored in the assembly buffer by the I/O processor. The buffer is also read/write accessed by the CAPS processor.

(o) CAPS Processor

1) The CAPS processor consists of a CPU control assembly and a CPU data path assembly. The CPU data path assembly provides the data manipulation and processing circuitry required to execute the CAPS instruction set. The CPU control assembly contains the circuitry required for operation and control of the processor system and program memory.

(p) Program Memory

1) The program memory assemblies contain the operating program (software) for the FCC. The software provides the commands which control the operation of the CAPS processor.

(q) Power Supply

1) The power supply assembly provides the required operating voltages for the FCC. The input to the power supply is provided by the airplane 115 V ac, 400 Hz system. The outputs of the power supply are 26v ac, +28, -28, +15, -15, +12, -12, and +5vdc.

(3) FCC Signal Selection

(a) General

1) The selection of local or voted FCC input data depends on the mode of operation and type of data used. Local data is used when two of the three sensor sources are invalid and system operation will be limited to F/D guidance only. Voted data is used for all autopilot and flight director modes if two of the three signals are valid.

(b) Inner Loop Data

1) Inner loop data is from the Inertial Reference System (IRS) and is used for autopilot stability. These signals are:

a) Pitch, pitch rate, roll, and roll rate during cruise.

b) Acceleration during approach.

2) Inner loop data is considered critical and always monitored by Signal Selection Fault Detection (SSFD). The SSFD signal voting provides high integrity input monitoring.

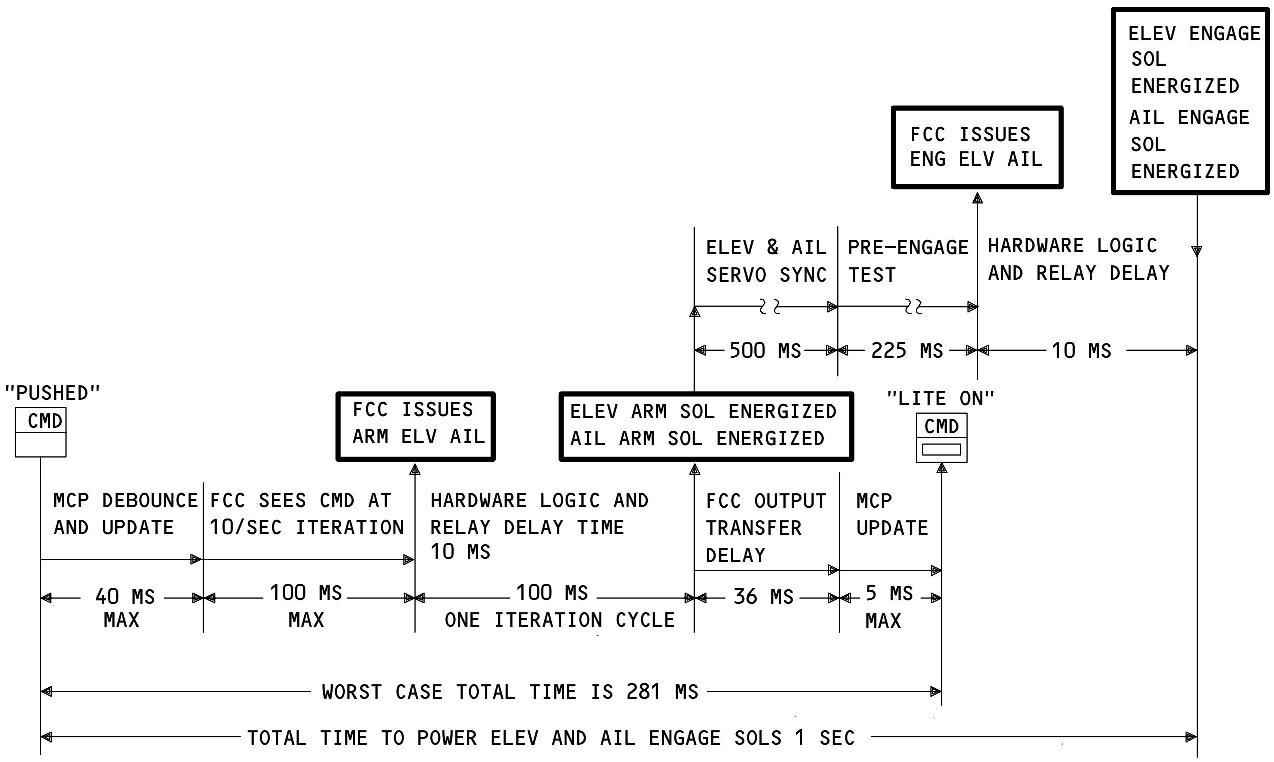
(c) Outer Loop Data
1) Outer loop data is from the following units:
a) Air Data Computer (ADC)
b) Flight Management Computer (FMC)
c) Instrument Landing System (ILS)
d) Radio Altimeter (RA)
2) The above units provide the following signals:
a) Angle of attack, altitude rate, altitude, airspeed, mach, and impact pressure (ADC)
b) VNAV and LNAV control laws (FMC).
c) Localizer and glide slope deviation and runway course (ILS).
d) Altitude above terrain (RA).

(d) Signal Selection
1) Triple Sensors
a) For single-channel engage or F/D modes, local outer loop data is used by the FCC. An invalid outer loop signal required for an autopilot engage mode causes the A/P to hold attitude and display mode fail and caution annunciations (Ref 22-14-00). An invalid outer loop signal does not disengage the A/P. If the invalid condition corrects before a timeout, the mode is automatically re-entered. An invalid local signal required for any F/D mode removes the F/D command bars from the EADI.
b) During multi-channel CMD, SSFD voted ILS and RA data is used by the FCC. The SSFD voting sets invalid signals to zero or the last voted value and selects the mid value signal of the three inputs. The first invalid signal required for multi-channel operation changes the ASA display (Ref 22-14-00). A second invalid input of the same signal disengages all A/P channels, inhibits engagement in channels with local failures, changes the ASA display, and display warning annunciations (Ref 22-14-00).
c) Signal selection while on the ground is inhibited. Local sensor data will be used for computation within each FCC.

2) Dual Sensors
a) Signal selection for ADC data allows selection of one of the two outer loop ADC data sources. For a F/D mode, left ADC data is used by all FCC's if the captain's F/D switch is ON and right ADC data is used if only the first officer's F/D switch is ON. If the autopilot is engaged, left ADC data is used if the left or center channel is engaged and right ADC data is used if the right channel is engaged. If any of the data being used becomes invalid, the FCC will automatically switch over and use the opposite ADC data.
b) For single-channel engage or F/D modes, failure of both ADC signals causes the autopilot to show mode fail and caution annunciations and to disengage. During multi-channel engage, failure causes no change until after touchdown and disengage.

(4) Flight Director Engage Logic
(a) When either MCP F/D switch is placed in the ON position, a F/D ON signal is transmitted to the FCCs. This signal is also transmitted to the EFIS signal generators for mode annunciation on the EADIs. The F/D ON signal engages the FCCs in the F/D mode. Once engaged, the selected FCC drives the appropriate EADI command bars (Ref 34-22-00).
(b) When the airplane is in the air, the F/D SELECT signal is also used to generate a MODE ON output. This is used to prepare the mode logic to activate CMD or FD modes. The MODE ON output will also result from the following conditions:
1) Autopilot engage with F/D OFF.
2) Autopilot engage with F/D ON.

(5) Autopilot Engagement Timing (Fig. 7)
(a) CMD Engage Sequence
1) When a CMD switch/light is pressed, MCP software debounces the switch contacts and encodes the engage request for transmission to the appropriate FCC. The FCC varifies the request and then applies power to the elevator and aileron ARM solenoids. Once armed these servos synchronize with the actual position of the flight controls.
2) A pre-engage test monitors the synchronization and verifies the servo's response to command. Upon completion of the test, the FCC powers the elevator and aileron engage solenoids.



Autopilot Engagement Timing Diagram
Figure 7

3) The dot-bar matrix in the CMD switch/light is lighted by an engage request accepted signal within 41 milliseconds after the ARM solenoids are energized.

(6) Autopilot/Flight Director System ARM and ENGAGE Logic (Fig. 8)

(a) A CMD engage request is initiated when the appropriate MCP switch/light is pressed. The MCP software verifies that the switch activity is valid and encodes the activity into an engage request digital transmission. The engage request is sent to the appropriate FCC. FCC software and hardware monitors determine if the engage request can be implemented.

(b) Once the engage request is received, the FCC energizes the appropriate arm solenoids if the FCC is not inhibited. The FCC is inhibited if:

1) Aileron or elevator arm solenoids are already energized
2) Electrical power monitor detects a failure
3) MCP parameters or interface is invalid
4) Both cross channel FCC test words have failed
5) Invalid configuration ground test discrete set

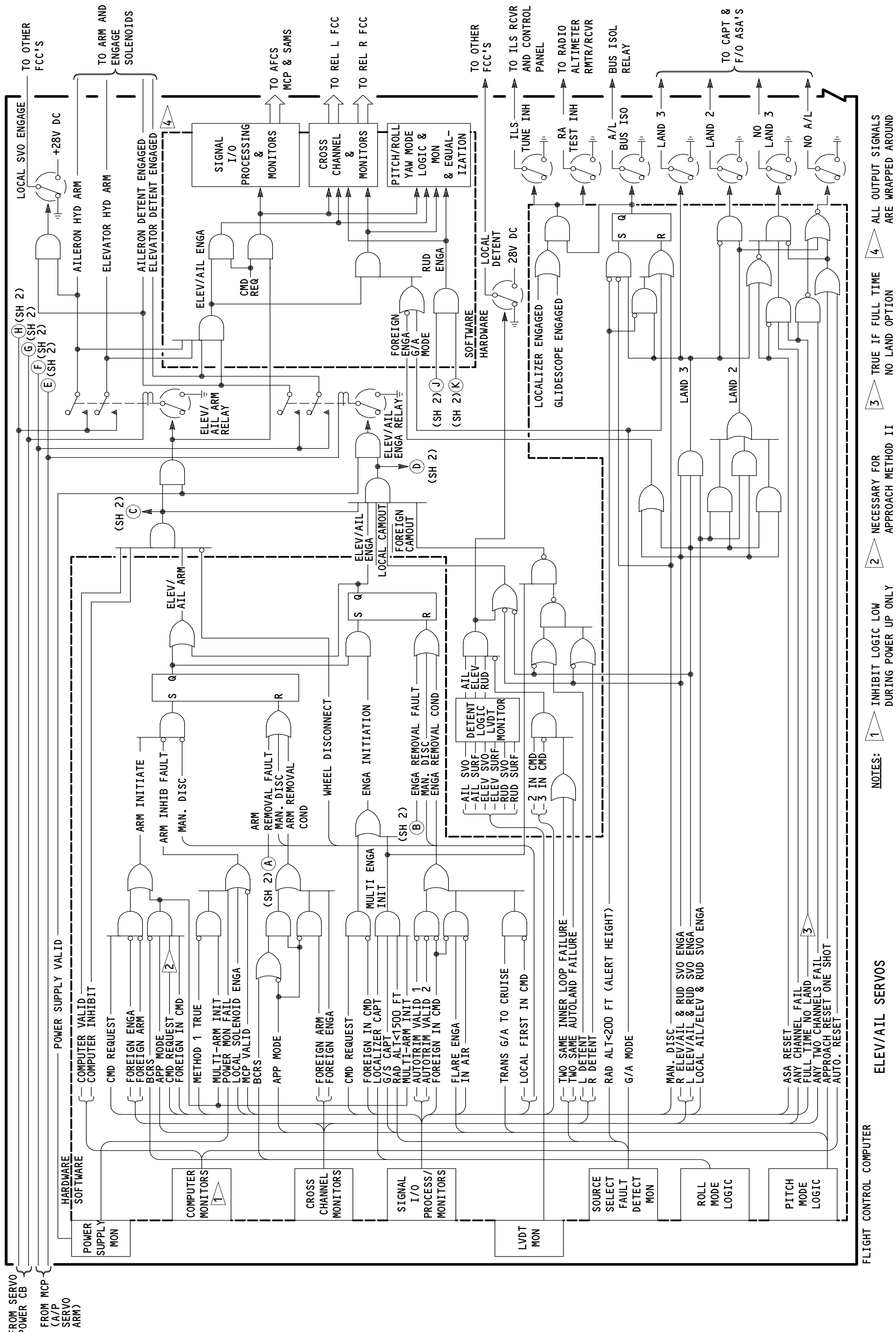
(c) Once armed, the A/P channel transitions from ARM to OFF if:

1) Aileron or elevator engage solenoids are already energized
2) FCC memory failed or self-test monitors are not verified
3) Autopilot disengage switch is pressed
4) Aileron exceedance test failed
5) Rudder solenoid already energized
6) Inner loop data invalid
7) Program pin status invalid

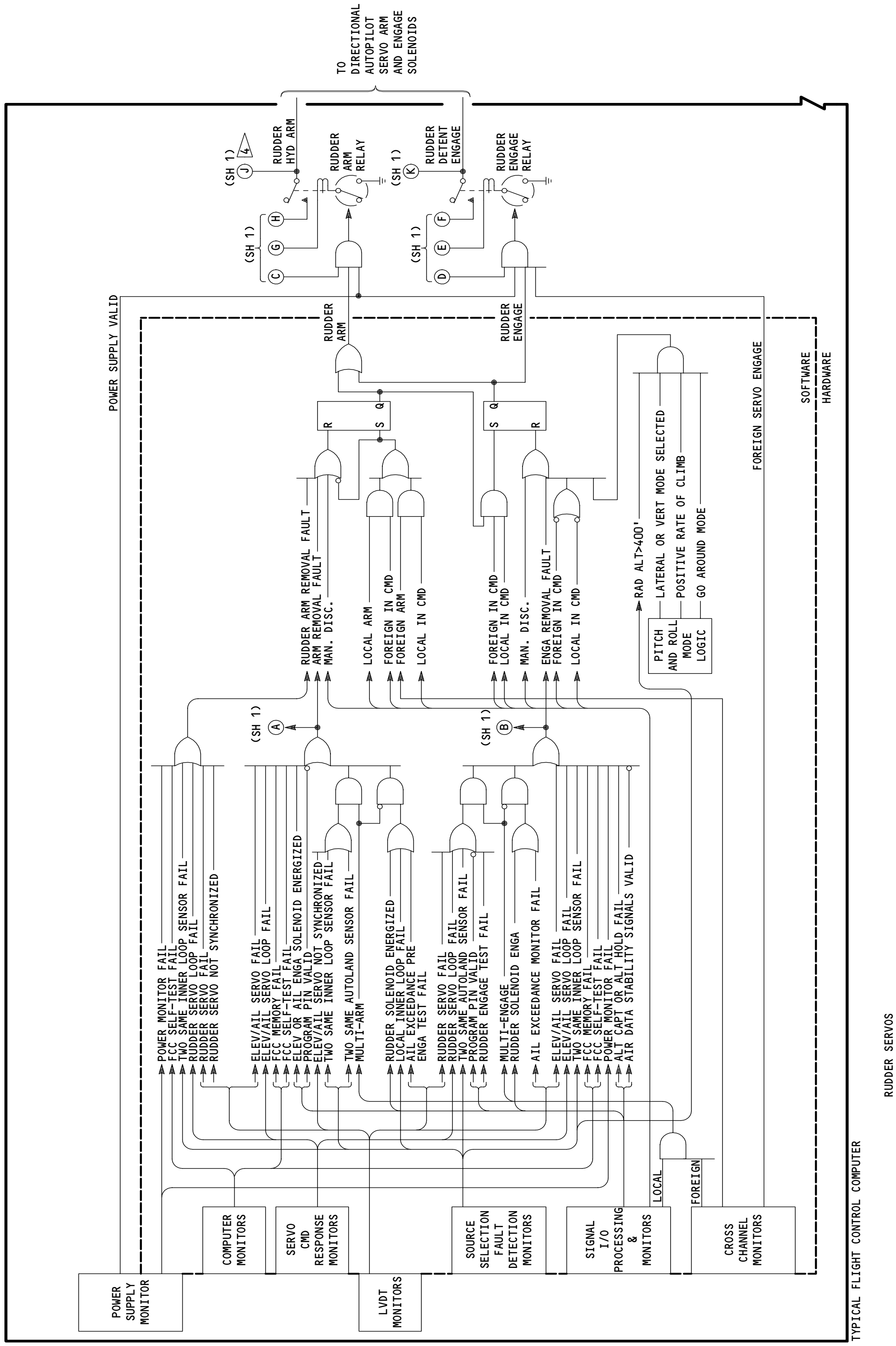
(d) After the arm solenoids are energized, an engage request acceptance signal is sent to the MCP. This signal causes the MCP to illuminate the appropriate CMD switch/light. The appropriate engage solenoids are also energized.

(e) Once engaged, the FCC transitions from CMD to OFF if:

1) FCC memory failed or self-test monitors are not verified
2) Rudder solenoid is energized
3) Aileron or elevator servo or servo loop failed
4) Any two identical inner loop sensors failed or invalid
5) FCC electrical power monitor detects a failure
6) Aileron exceedance detector monitor detects a failure
7) ALT HLD or override function cannot be performed
8) Autopilot disengage switch pressed



Autopilot/Flight Director System Arm and Engage Logic Schematic
Figure 8 (Sheet 1)



Autopilot/Flight Director System Arm and Engage Logic Schematic
Figure 8 (Sheet 2)

9) Air data computer stability signals invalid for seven seconds

(f) An engage request accepted signal is sent to the engage/disengage logic and the flight annunciation logic for engage annunciation on the EADI. The cross-channel data buses allow each FCC to provide its engage status to the other two FCCs (Ref 22-15-00).

(7) Multichannel Autopilot Engage

(a) Multichannel CMD engage requests are initiated when the APP mode is selected and one FCC is engaged in the CMD mode. Each unengaged FCC automatically energizes its appropriate arm solenoids if the FCC is not inhibited. The FCC is inhibited if:

(b) The FCC is inhibited if:

1) FCC program pin status invalid
2) Conditions under single channel arm inhibit occur

(c) Once armed, each A/P channel transitions from ARM to OFF if:

1) Aileron, elevator, or rudder servos fail or servo loops not verified
2) Aileron or elevator servos not synchronized to surface position
3) Rudder servo not synchronized to surface position within 3 seconds
4) Any two identical aileron or elevator inner loop or autoland sensor parameters failed or invalid
5) Any two identical rudder inner loop sensor parameters or interfaces invalid
6) Conditions under single channel ARM to OFF transition occur

(d) After the arm solenoids are energized, each CMD switch/light is lighted. The appropriate engage solenoids are then energized upon capture of glideslope, localizer on course (AGTE<°2), radio altitude less than 1500 feet and 5 seconds.

(e) Once engaged, each FCC transitions from CMD to OFF if:

1) FCC memory failed or self-test monitors are not verified
2) Aileron, elevator, or rudder servo or servo loop fails
3) Any two identical inner loop sensors failed or invalid
4) Electrical power monitor detects a failure
5) Any two identical autoland sensors failed or invalid
6) FCC program pin status invalid
7) ALT HLD or override function cannot be performed
8) Autopilot disengage switch pressed

(f) The cross-channel data buses allow each FCC to provide its engage status to the other two. If the first-in-command FCC (first FCC engaged in CMD during multichannel operation) is disengaged, the first-in-command status is automatically transferred to the next FCC as follows: LEFT, RIGHT, CENTER, LEFT.

B. BITE

(1) The FCC provides in-flight detection of AFDS and related component failures. Detection includes:

(a) System fault detection and fault location

(b) Monitoring of servos, sensors, input interfaces, and output interfaces
(c) Self Test
(d) Storage of internal fault identification

(2) Fault Detection and Location

(a) The FCC detects flight faults in the following AFDS system components:
1) Autopilot Servos
2) MCP - FCC Interfaces
3) FCC

(b) The FCC, in conjunction with the MCDP, can identify the location of the flight faults listed above. The MCDP ground tests verify the proper operation of the MCP front panel switch/lights, switches, and displays. The ground tests also verify proper operation of the A/P disengage and go-around switches.

(3) FCC Interface Monitors

(a) The A/P servo control functions are checked by command/response monitors which verify the function of the servo loop, and by current drive wraparounds which check the circuitry to the servo drivers. The command/response monitors verify that the actuator moves when commanded. The current drive wraparounds verify that the servo is functioning correctly by comparing actual servo movement with the movement commanded by the FCC software. Detent comparators verify that the airplane control surfaces are following the commanded signals. Dual comparators, one in hardware and one in software, monitor the servo position and the airplane control surface position. An incorrect comparison generates a fault signal for use in software.

(b) Linear variable differential transducer (LVDT) signal outputs are checked by hardware monitors. The hardware monitors detect faults caused by shorted wires, broken wires, or loss of excitation, and generate a fault signal for use in software. These monitors are used for the servo position sensor inputs. Signal inputs from sensors other than LVDT's are monitored by software. Signal selection fault detection routines are used to determine sensor correctness and to provide voter outputs for multichannel operation. Incorrect sensor signals are voted out of the final results.

(c) Discrete input and output interfaces are monitored by software which detects shorted wires. These monitors are used for the caution/warning inputs and outputs, engage and arm solenoid power, trim outputs, engage and arm commands and inputs, and autoland status annunciator inputs and outputs. The interface monitor circuits are tested, as appropriate, at power up or prior to engagement of autopilot/autoland.

(d) The ARINC 429 inputs from system LRUs are checked in two ways. First, the FCC checks the sign status matrix to verify the validity of the incoming data. The buffer memory locations are also periodically tested to determine if the data words have been updated. If the words have not been updated, then a fault is assumed and the data words at the tested memory locations are not used.

(e) The ARINC 429 test words are generated by the FCC in order to verify the correct operation of the circuitry involved in communication with the mode control panel and with the other FCCs in the system. The test words are sent to the receiving system unit, operated on, and then transmitted back to the originating FCC. The results are compared with the expected answers to determine if faults exist.

(4) FCC Self-Test

(a) The FCC provides a number of self-tests. The tests are divided into two basic categories: power-up/pre-engage tests and operation verification tests. The power-up/pre-engage category includes tests which would cause adverse system reactions if performed during flight. The operation verification category provides critical function monitoring such as the CAPS processor memories, interfacing electronics, and power supplies. These types of functions are monitored on a continuous or recurring basis.

(b) CAPS Processor

1) The CAPS processor self-test is implemented using several techniques, and provides checks of processor components and data, address, and control paths. During execution of the self-test program, the program monitor counts the number of microcycles in each iteration of the self-test program and also monitors the execution time of the program. A second hardware-implemented check is provided by the processor parity monitor. The parity monitor checks the instruction mapping into the microprogram memory. Both self-test program monitors are checked at power up.

2) The computation rate of the processor is continuously checked during the normal operation of the FCC. During each 25 millisecond computation period, a software generated signal is sent to the program monitor. If the signal is more than ± 5 milliseconds out of tolerance, a fault signal is generated. The circuit does not reset until two consecutive within tolerance signals are received. The monitor circuit is checked at power up for correct fault response.

(c) Memories

1) The program memory is checked by a software routine. The checksum of the ROM contents is compared with the expected sum (stored in memory), and a fault signal is generated if the comparison fails.

2) The RAM is checked by three monitors in the FCC. At power up, a write-then-read test is performed on each memory location. During flight, the RAM is in use, so two RAM locations, with complementary addresses, are checked with a write-then-read test. The in-flight test verifies the condition of the address and data lines. During normal operation, parity bits are added to words written into memory and checked when words are read from memory.

(d) Interfacing Electronics

1) Internal I/O monitoring is performed primarily in software. If the I/O logic fails, the CAPS processor indicates a fault. The I/O processor address and data lines are checked by the CAPS processor with a write-then-read test. A failure is indicated if the CAPS processor does not receive the proper word. Wraparound monitoring is used to check, in part, the ARINC 429 I/O and analog I/O circuits. Test words are wrapped around to test the 429 UART and 429 multiplexer circuits. The MCP and MCDP outputs provide the low-speed 429 wraparounds and the FCC left cross-channel output provides the high-speed 429 wraparound. The analog wraparound signals are generated and fed back into the FCC analog inputs. The analog signals wrapped around are servo position, force link position, servo drive, and commanded output.

2) The inertial reference system (IRS) input is brought in through two separate paths. The IRS information is stored in separate memory locations thus allowing data comparisons by the CAPS processor. If a fault exists in either path, the comparison will indicate an inequality.

3) The ARINC 429 outputs which are not wrapped around are checked by hardware monitors. The monitors detect failures of one side of each ARINC 429 driver output and send a fail signal to the software if a fault is detected.

4) The A/D and D/A converter is used to check the reference voltage used in the conversion logic. The reference voltage is scaled and inputted. The digital result is compared, in software, with the expected result, and a fault is indicated if the voltage is out of tolerance. The analog wraparounds cannot be used to find reference voltage faults because both the D/A and A/D converters use the reference voltage generated by the A/D converter. A separate reference is, therefore, required.

(e) Power Supplies

1) The power supply voltages are continuously monitored for overvoltage, undervoltage, and overcurrent conditions. The +5, +15, and -28 vdc outputs are monitored by additional overvoltage monitors which latch the entire power supply in a shut-down state. The 115 Vac input power must be interrupted to reset these monitors. The voltage and frequency of the 28 vac supply output are checked by software.

(5) Internal Fault Storage

(a) The FCC provides permanent storage for up to five internal faults which result in an FCC invalid, and temporary storage for five words of fault data per flight leg.

1) Faults detected by FCC self-test are stored in memory for access by shop support equipment. A maximum of five faults can be stored at one time. This fault data can only be removed from memory in the maintenance shop.

2) Fault data in temporary storage is formated for transmission over a ARINC 429 data bus to the MCDP. The FCC clears temporary data at the start of each flight leg. Data provided to the MCDP includes:

3) Fault identity and location

4) Source of fault detection

5) Fault effect on flight deck

6) Fault intermittent or constant

7) Fault occurred in-flight or on-ground

AUTOPILOT/FLIGHT DIRECTOR POWER

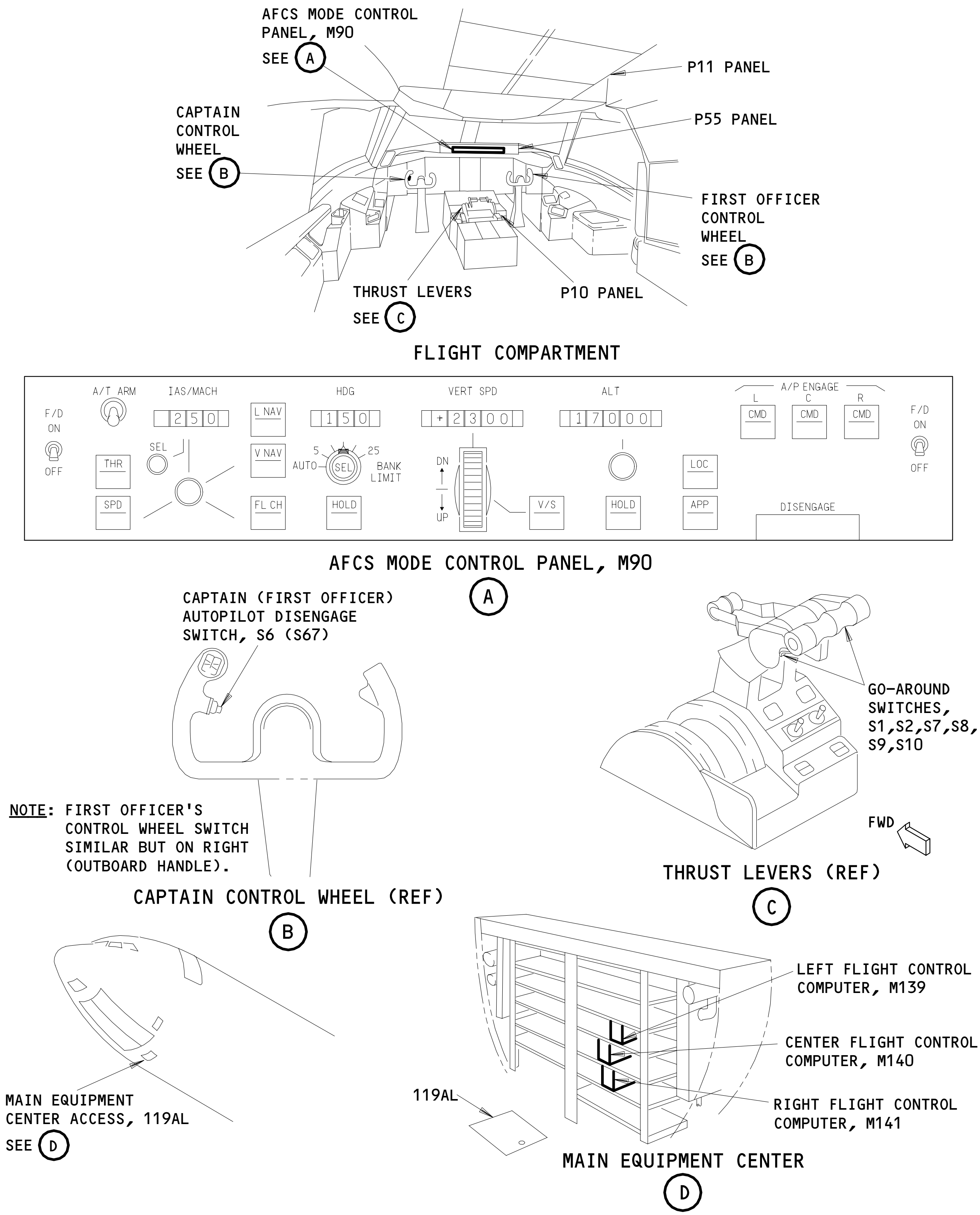
COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	REFERENCE
ACTUATOR - (REF 27-11-00, FIG. 101)				
C LATERAL CONTROL, M275				
L LATERAL CONTROL, M274				
R LATERAL CONTROL, M276				
ANNUNCIATOR - (REF 22-14-00, FIG. 101)				
AUTOLAND STATUS, CAPT, N70				
AUTOLAND STATUS, F/O, N71				
CIRCUIT BREAKERS	--		FLT COMPT, P11	
AUTOFLIGHT WARN, C521		1	11A17	*
FLT CONT CMPTR PWR C, C515		1	11E20	*
FLT CONT CMPTR PWR L, C513		1	11E17	*
FLT CONT CMPTR PWR RIGHT, C514		1	11E35	*
FLT CONT CMPTR SERVO C, C524		1	11E21	*
FLT CONT CMPTR SERVO L, C522		1	11E18	*
FLT CONT CMPTR SERVO RIGHT, C523		1	11E36	*
MODE CONT PNL L, C516		1	11E16	*
MODE CONT PNL R, C517		1	11E34	*
COMPUTER - (REF 22-31-00, FIG. 101)				
THRUST MANAGEMENT, M183				
COMPUTER - (REF 31-41-00, FIG. 101)				
L EICAS, M10181				
R EICAS, M10182				
COMPUTER - C FLIGHT CONTROL, M140	--	1	119AL, MAIN EQUIP CTR, E1-4	22-11-01
COMPUTER - L FLIGHT CONTROL, M139	--	1	119AL, MAIN EQUIP CTR, E1-3	22-11-01
COMPUTER - R FLIGHT CONTROL, M141	--	1	119AL, MAIN EQUIP CTR, E1-5	22-11-01
COMPUTER - (REF 34-61-00, FIG. 101)				
L FLIGHT MANAGEMENT, M134				
R FLIGHT MANAGEMENT, M135				
INDICATOR - (REF 29-31-00, FIG. 101)				
HYDRAULIC SYSTEM CONTROL PANEL, M10				
LIGHT - (REF 22-14-00, FIG. 101)				
AUTOPILOT CAUTION, L19				
A/P DISC, LD/M779				
MODULE - (REF 27-58-00, FIG. 101)				
C FLAP/STABILIZER POSITION, M839				
L FLAP/STABILIZER POSITION, M838				
R FLAP/STABILIZER POSITION, M840				
MODULE - (REF 27-09-00, FIG. 101)				
1L SPOILER CONTROL, M530				
2R SPOILER CONTROL, M534				
3L SPOILER CONTROL, M532				
MODULE - (REF 34-16-00, FIG. 101)				
ALTITUDE ALERT, M617				

* SEE THE WDM EQUIPMENT LIST

Autopilot/Flight Director Power - Component Index
Figure 101 (Sheet 1)

COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	REFERENCE
PANEL - AFCS MODE CONTROL, M90	--	1	FLT COMPT, P55	22-11-02
PANEL - (REF 22-41-00, FIG. 101)				
MAINTENANCE CONTROL DISPLAY, M168				
RECEIVER - (REF 34-31-00, FIG. 101)				
C ILS, M157				
L ILS, M156				
R ILS, M158				
RELAY - (REF 31-01-06, FIG. 101)				
ISOLATION REQUEST, K122				
RELAY - (REF 31-01-36, FIG. 101)				
AIR/GROUND, SYS NO. 1, K142				
RELAY - (REF 31-01-37, FIG. 101)				
AIR/GROUND, SYS NO. 2, K201				
SERVO - (REF 22-12-00, FIG. 101)				
C ELEVATOR AUTOPILOT, M272				
L ELEVATOR AUTOPILOT, M271				
R ELEVATOR AUTOPILOT, M273				
SERVO - (REF 22-13-00, FIG. 101)				
C DIRECTIONAL AUTOPILOT, M278				
L DIRECTIONAL AUTOPILOT, M277				
R DIRECTIONAL AUTOPILOT, M279				
SWITCH - AFCS GO-AROUND, S1,S2,S7,S8,S9,S10	--	6	FLT COMPT, P10	22-11-04
SWITCH - AUTOPILOT DISENGAGE, CAPT, S67	--	1	FLT COMPT, CONTROL WHEEL, CAPT	22-11-03
SWITCH - AUTOPILOT DISENGAGE, F/O, S6	--	1	FLT COMPT, CONTROL WHEEL, F/O	22-11-03
SWITCH - (REF 22-32-00, FIG. 101)				
C LEADING EDGE SLAT, S523				
L LEADING EDGE SLAT, S521				
R LEADING EDGE SLAT, S522				
SWITCH - (REF 34-22-00, FIG. 101)				
L INSTRUMENT SOURCE SELECT, CAPT, S1				
R INSTRUMENT SOURCE SELECT, F/O, S9				
SYMBOL GENERATOR - (REF 34-22-00, FIG. 101)				
C EFIS, M149				
L EFIS, M148				
R EFIS, M150				
TRANSMITTER/RECEIVER - (REF 34-33-01, FIG. 101)				
C RADIO ALTIMETER, M204				
L RADIO ALTIMETER, M202				
R RADIO ALTIMETER, M203				
UNIT - (REF 31-01-06, FIG. 101)				
INSTRUMENT BUS VOLTAGE SENSE, M1079				
UNIT - (REF 31-51-00, FIG. 101)				
WARNING ELECTRONICS, P51				
UNIT - (REF 34-21-00, FIG. 101)				
C INERTIAL REFERENCE, M160				
L INERTIAL REFERENCE, M159				
R INERTIAL REFERENCE, M161				

Autopilot/Flight Director Power - Component Index
Figure 101 (Sheet 2)



Autopilot/Flight Director Power - Component Location
Figure 102

FLIGHT CONTROL COMPUTER - REMOVAL/INSTALLATION

1. General
   A. Three Flight Control Computers (FCC) are installed on the main equipment center. The left FCC is on shelf E1-3, the center FCC is on shelf E1-4, and the right FCC is on shelf E1-5. The computers have hold-down hooks on the front to attach them to the rack. One multi-pin connector is at the rear of each unit.

TASK 22-11-01-004-001

2. Remove the Flight Control Computer
   A. References
      (1) 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
      (2) 20-10-01/401, E/E Rack-Mounted Components
      (3) 20-41-01/201, Electrostatic Discharge Sensitive Devices
   B. Access
      (1) Location Zones
         119/120 Main Equipment Center

      (2) Access Panel
         119AL Main Equipment Center

   C. Prepare For Removal

      S 864-018
      (1) Set the two F/D switches on the MCP to OFF.

      S 864-002
      (2) For left FCC removal, open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
         (a) 11A17, AUTOFLIGHT WARN
         (b) 11E16, MODE CONT PNL L
         (c) 11E17, FLT CONT CMPTR PWR L
         (d) 11E18, FLT CONT CMPTR SERVO L
         (e) 11E34, MODE CONT PNL R

      S 864-003
      (3) For center FCC removal, open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
         (a) 11A17, AUTOFLIGHT WARN
         (b) 11E16, MODE CONT PNL L
         (c) 11E20, FLT CONT CMPTR PWR C
         (d) 11E21, FLT CONT CMPTR SERVO C
         (e) 11E34, MODE CONT PNL R

      S 864-019
      (4) For right FCC removal, open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
         (a) 11A17, AUTOFLIGHT WARN
         (b) 11E16, MODE CONT PNL L

(c) 11E34, MODE CONT PNL R
(d) 11E35, FLT CONT CMPTR PWR R
(e) 11E36, FLT CONT CMPTR SERVO R

S 014-021
(5) Open the main equipment center access door, 119AL (Ref 06-41-00).

D. Remove the Flight Control Computer

S 914-004

CAUTION: DO NOT TOUCH THE FLIGHT CONTROL COMPUTER BEFORE YOU DO THE PROCEDURE FOR DEVICES THAT ARE SENSITIVE TO ELECTROSTATIC DISCHARGE. ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE FLIGHT CONTROL COMPUTER.

(1) Do the procedure for devices that are sensitive to electrostatic discharge (Ref 20-41-01).

S 024-005
(2) Remove the FCC (Ref 20-10-01).

S 434-006
(3) Install a protective cover on the connector at the rear of the FCC.

TASK 22-11-01-404-007

3. Install the Flight Control Computer

A. References
(1) 20-10-01/401, E/E Rack-Mounted Components
(2) 20-41-01/201, Electrostatic Discharge Sensitive Devices
(3) 22-00-02/201, Autoflight BITE
(4) 24-22-00/201, Electrical Power - Control

B. Access
(1) Location Zones
119/120 Main Equipment Center

(2) Access Panel
119AL Main Equipment Center

C. Install the Flight Control Computer

S 914-008

CAUTION: DO NOT TOUCH THE FLIGHT CONTROL COMPUTER BEFORE YOU DO THE PROCEDURE FOR DEVICES THAT ARE SENSITIVE TO ELECTROSTATIC DISCHARGE. ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE FLIGHT CONTROL COMPUTER.

(1) Do the procedure for devices that are sensitive to electrostatic discharge (Ref 20-41-01).

S 034-009
(2) Remove the protective cover from the connector at the rear of the FCC.

S 424-010
(3) Install the FCC (Ref 20-10-01).

D. Test the Flight Control Computer.

S 864-011
(1) Supply electrical power (Ref 24-22-00).

S 864-020
(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11A17, AUTOFLIGHT WARN
(b) 11E16, MODE CONT PNL L
(c) 11E17, FLT CONT CMPTR PWR L
(d) 11E18, FLT CONT CMPTR SERVO L
(e) 11E20, FLT CONT CMPTR PWR C
(f) 11E21, FLT CONT CMPTR SERVO C
(g) 11E34, MODE CONT PNL R
(h) 11E35, FLT CONT CMPTR PWR R
(i) 11E36, FLT CONT CMPTR SERVO R

S 864-012
(3) Make sure the MCDP is out of Ground Test Mode. Set the MCDP to OFF or go into Flight Faults Mode.

NOTE: After a FCC is replaced, the MCDP must be put out of Ground Test Mode before MCDP Ground Test 01-FCC is done. This removes fault messages from the MCDP memory that are corrected.

S 864-013
(4) Put the MCDP into Ground Test Mode (Ref 22-00-02).

S 724-014
(5) Do MCDP Ground Test 01-FCC (Ref 22-00-02).
(a) Make sure no fault messages show.

E. Put the Airplane Back to Its Usual Condition

S 864-015
(1) Set the MCDP to off.

S 864-016
(2) Close the 119AL access door.

S 864-017
(3) Remove electrical power if it is not necessary (Ref 24-22-00).

AFCS MODE CONTROL PANEL - MAINTENANCE PRACTICES

1. General
   A. This procedure has these tasks:
      (1) The removal of the AFCS Mode Control Panel (MCP).
      (2) The installation of the MCP.
      (3) The removal of a mode select switch and/or a lamp for the mode select switch.
      (4) The installation of a mode select switch and/or a lamp for the mode select switch.
   B. The AFCS Mode Control Panel (MCP) is in the center of the glareshield, P55, on the flight deck. The MCP is installed on supports in the glareshield structure. Four captive screws in the bottom of the glareshield attach the MCP to the glareshield. Three electrical connectors are at the rear of the MCP. An opening at the rear of the MCP automatically connects with the equipment cooling system when the panel is installed.

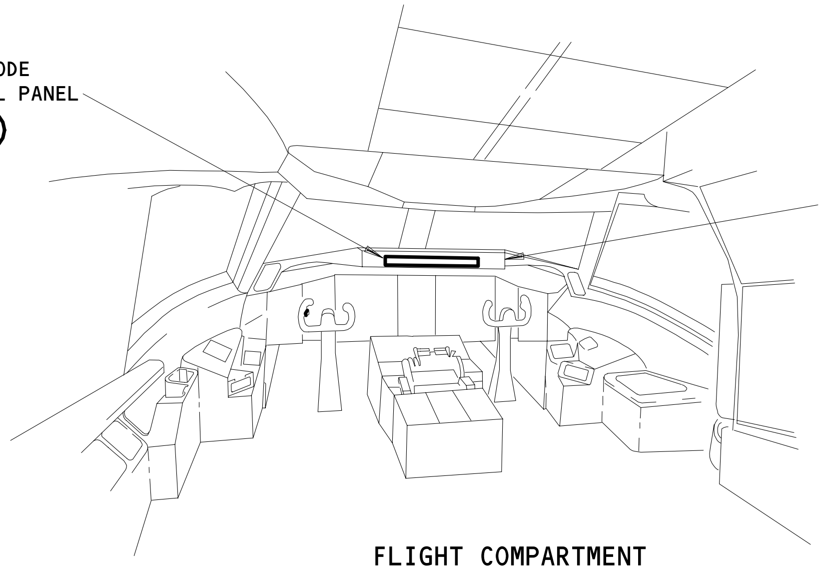
TASK 22-11-02-002-001

2. Mode Control Panel Removal
   A. Reference
      (1) AMM 20-41-01/201, Electrostatic Discharge Sensitive Devices
   B. Access
      (1) Location Zone
          211/212 Flight Compartment

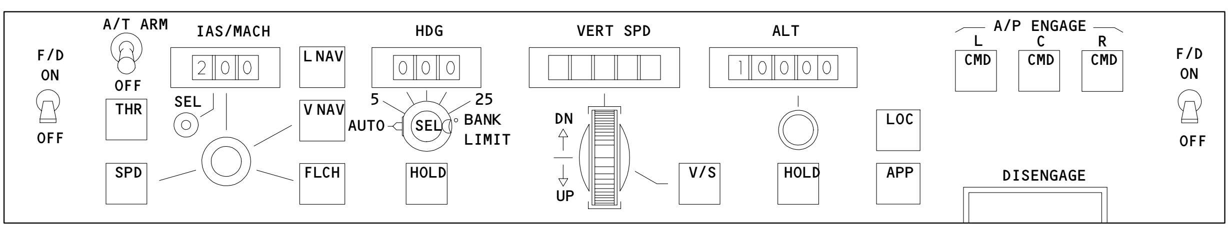
   C. Prepare For Removal

      S 862-002

      (1) Open these circuit breakers and attach DO-NOT-CLOSE tags:
          (a) P11 Overhead Circuit Breaker Panel:
              1) 11B7, STBY INSTR LTS
              2) 11E16, MODE CONT PNL L
              3) 11E17, FLT CONT CMPTR PWR L
              4) 11E18, FLT CONT CMPTR SERVO L
              5) 11E20, FLT CONT CMPTR PWR C
              6) 11E21, FLT CONT CMPTR SERVO C
              7) 11E34, MODE CONT PNL R
              8) 11E35, FLT CONT CMPTR PWR R
              9) 11E36, FLT CONT CMPTR SERVO R



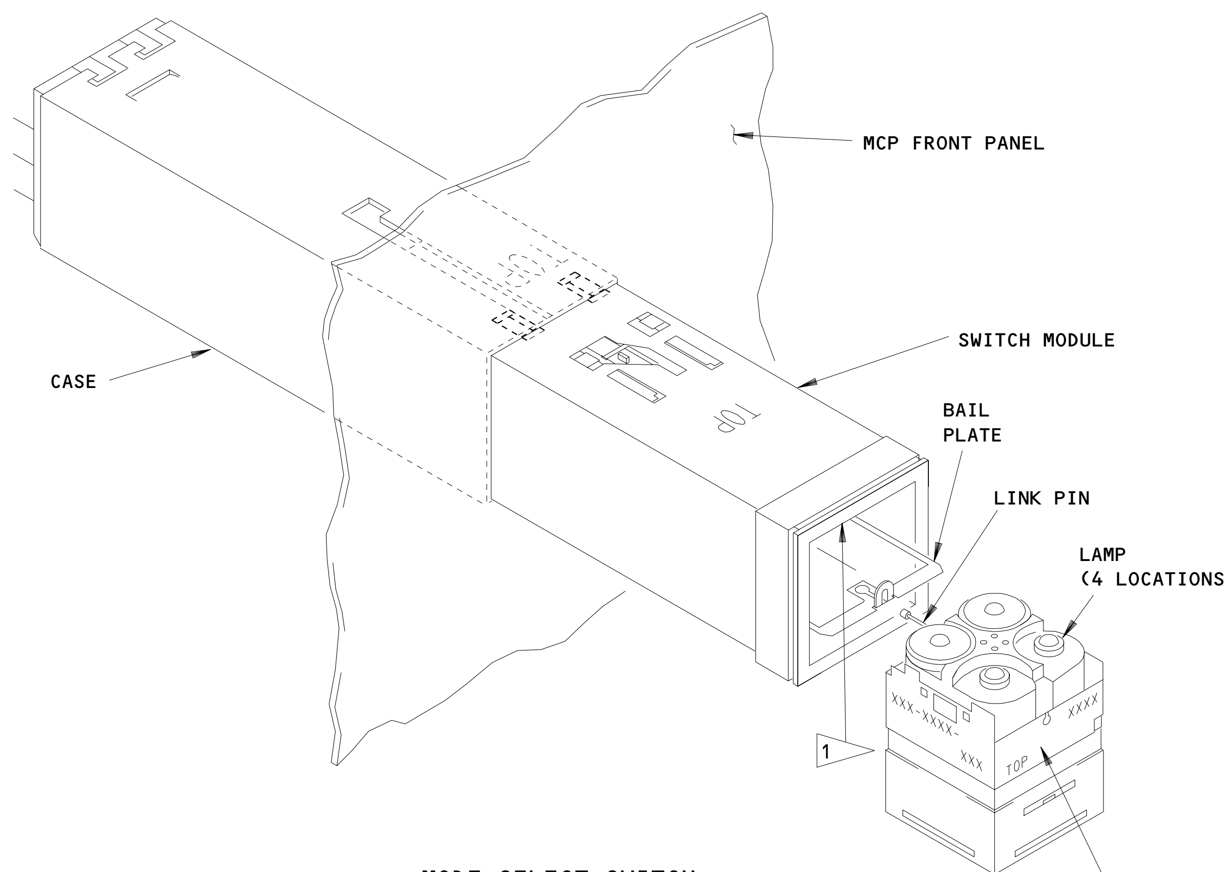
FLIGHT COMPARTMENT



AFCS MODE CONTROL PANEL

A

AFCS Mode Control Panel Installation
Figure 201



MODE SELECT SWITCH
(EXAMPLE)

1 THE JACKSCREW IS AT THE TOP IN THE SWITCH MODULE, AND BEHIND THE MATING PLATE FOR THE LENS MODULE.

Mode Select Switch Removal/Installation
Figure 202

10) 11P30, GLARESHIELD INSTR AND PANEL
11) 11R3, IND LTS L
12) 11R30, IND LTS R

(b) P6 Main Power Distribution Panel:

1) 6K32, R3 PILOT SECONDARY IND LT
2) 6L32, L3 PILOT SECONDARY IND LT

D. Remove the Mode Control Panel

S 912-003

CAUTION: DO NOT TOUCH THE MCP BEFORE YOU DO THE PROCEDURE FOR DEVICES THAT ARE SENSITIVE TO ELECTROSTATIC DISCHARGE. ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE MCP.

(1) Do the procedure for devices that are sensitive to electrostatic discharge (AMM 20-41-01/201).

S 032-004

(2) Loosen the four screws on the bottom of the MCP supports in the glareshield that attach to the MCP.

S 012-005

CAUTION: CAREFULLY MOVE THE MCP OUT FROM ITS SUPPORTS IN THE GLARESHIELD TO PREVENT FORCE ON THE ELECTRICAL CABLES. DAMAGE TO THE ELECTRICAL CABLES COULD OCCUR IF FORCE IS PUT ON THEM.

(3) Move the MCP out from its supports in the glareshield until you can get access to the electrical connectors at the rear of the MCP unit.

S 032-045

CAUTION: CROSS CONNECTION POSSIBILITY WHEN WORKING WITH THIS COMPONENT. CLEARLY IDENTIFY CONNECTIONS UPON DISCONNECTION AND FUNCTION CHECK UPON RECONNECTION.

(4) Disconnect the electrical connectors.
(a) Apply protective covers to the electrical connectors.

S 022-007
(5) Remove the MCP from its supports in the glareshield.

TASK 22-11-02-402-008
3. Mode Control Panel Installation
A. References
(1) AMM 20-41-01/201, Electrostatic Discharge Sensitive Devices
(2) AMM 22-00-02/201, Autoflight BITE
(3) AMM 24-22-00/201, Electrical Power - Control
B. Access
(1) Location Zone
211/212 Flight Compartment

C. Install the Mode Control Panel

S 912-009

CAUTION: DO NOT TOUCH THE MCP BEFORE YOU DO THE PROCEDURE FOR DEVICES THAT ARE SENSITIVE TO ELECTROSTATIC DISCHARGE. ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE MCP.

(1) Do the procedure for devices that are sensitive to electrostatic discharge (AMM 20-41-01/201).

S 032-010
(2) Remove the protective covers from the connectors at the rear of the MCP.

S 032-011

CAUTION: MAKE SURE THAT THE AIR INLET COVER ON THE MCP IS REMOVED. DAMAGE TO EQUIPMENT COULD OCCUR IF THIS OPENING IS BLOCKED.

(3) Make sure that the air inlet cover on the MCP is removed.

S 212-012

CAUTION: MAKE SURE THAT THE AIR OUTLET AT THE REAR OF THE MCP SUPPORTS IN THE GLARESHIELD IS CLEAR OF BLOCKAGE. EQUIPMENT DAMAGE COULD OCCUR IF THIS OPENING IS BLOCKED.

(4) Make sure the air outlet at the rear of the supports in the glareshield is clear.

S 412-013

CAUTION: CAREFULLY PUT THE MCP INTO ITS SUPPORTS IN THE GLARESHIELD TO PREVENT FORCE ON THE ELECTRICAL CABLES. DAMAGE TO ELECTRICAL CABLES COULD OCCUR IF FORCE IS PUT ON THEM.

(5) Move the MCP into its supports in the glareshield until the electrical cables can be connected to the rear of the MCP.

S 432-046

CAUTION: CROSS CONNECTION POSSIBILITY WHEN WORKING WITH THIS COMPONENT. POSITIVELY IDENTIFY CONNECTIONS PRIOR TO RECONNECTION.

(6) Connect the electrical connectors.

S 422-015
(7) Move the MCP into its installed position.

S 432-016
(8) Tighten the screws (4) on the bottom of the MCP supports in the glareshield.

S 752-017
(9) Make sure the new MCP has the same switch configuration as the MCP that was removed (Fig. 201).
(a) Make sure the new MCP has a THR lens installed.

D. Test Mode Control Panel.

S 862-018
(1) Supply electrical power (AMM 24-22-00/201).

S 862-019
(2) Set the two F/D switches on the MCP to OFF.

S 862-020
(3) Remove the DO-NOT-CLOSE tags and close these circuit breakers:
(a) P11 Overhead Circuit Breaker Panel:
1) 11B7, STBY INSTR LTS
2) 11E16, MODE CONT PNL L
3) 11E17, FLT CONT CMPTR PWR L
4) 11E18, FLT CONT CMPTR SERVO L
5) 11E20, FLT CONT CMPTR PWR C

6) 11E21, FLT CONT CMPTR SERVO C
7) 11E34, MODE CONT PNL R
8) 11E35, FLT CONT CMPTR PWR R
9) 11E36, FLT CONT CMPTR SERVO R
10) 11P30, GLARESHIELD INSTR AND PANEL
11) 11R3, IND LTS L
12) 11R30, IND LTS R

(b) P6 Main Power Distribution Panel:
1) 6K32, R3 PILOT SECONDARY IND LT
2) 6L32, L3 PILOT SECONDARY IND LT

S 712-021

(4) Do MCDP ground test 04-MCP (AMM 22-00-02/201).
(a) Make sure no failures occur during test.

S 862-022

(5) Set the MCDP to off.

S 862-023

(6) Remove electrical power if it is not necessary (AMM 24-22-00/201).

TASK 22-11-02-022-024

4. Mode Select Switch and Lamp Removal (Fig. 202)

A. General
(1) This task has steps to remove a mode select switch and to remove a lamp for the mode select switch.

B. Equipment
(1) 50445 - Extractor Tool, Jay-E1 Product Inc., 23301 S. Wilmington, Carson, CA 90745

C. Access
(1) Location Zones
211/212   Flight Compartment

D. Procedure

S 862-025

(1) Open these circuit breakers and attach the DO-NOT-CLOSE tags:
(a) P11 Overhead Circuit Breaker Panel
1) 11B7, STBY INSTR LTS

2) 11E16, MODE CONT PNL
3) 11E18, FLT CONT CMPTR SERVO L
4) 11E21, FLT CONT CMPTR SERVO C
5) 11E34, MODE CONT PNL
6) 11E36, FLT CONT CMPTR SERVO R
7) 11P30, GLARESHIELD INSTR AND PANEL
8) 11R3, IND LTS L
9) 11R30, IND LTS R

(b) P6 Main Power Distribution Panel:
1) 6K32, R3 PILOT SECONDARY IND LT
2) 6L32, L3 PILOT SECONDARY IND LT

S 032-026

(2) Use your fingernails (or the extractor tool) in the grooves at the sides of the lens module to pull the lens module out.

NOTE: The lens module is now held by a link pin and a bail plate.

(a) Push the link pin to the rear of the groove on the bail plate to remove the lens module if it is necessary.

(b) NOTE: If removing lens module with P/N 837-0034-XXX, rotate the lens module 90 degrees. At 90 degrees the bail feature will unlock from the lens module.

S 022-027

(3) To remove a lamp, pull the lamp out of the rear of the lens module.

S 022-028

(4) To remove the switch, do these steps:
(a) Turn the jackscrew counterclockwise until it stops.

NOTE: The jackscrew is at the top in the switch module, and behind the mating plate for the lens module.

(b) Pull the switch module out from the case.

TASK 22-11-02-422-029

5. Mode Select Switch and Lamp Installation (Fig. 202)

A. General

(1) This task has steps to install a mode select switch and to install a mode select switch lamp.

B. References
   (1) AMM 22-10-00/501, Autopilot/Flight Director System
   (2) AMM 24-22-00/201, Manual Control
C. Access
   (1) Location Zones
       211/212   Flight Compartment

D. Procedure

   S 862-038
   (1) Make sure that these circuit breakers are open and the DO-NOT-CLOSE tags are attached:
      (a) P11 Overhead Circuit Breaker Panel
         1) 11B7, STBY INSTR LTS
         2) 11E18, FLT CONT CMPTR SERVO L
         3) 11E21, FLT CONT CMPTR SERVO C
         4) 11E36, FLT CONT CMPTR SERVO R
         5) 11E16, MODE CONTROL PANEL
         6) 11E34, MODE CONTROL PANEL
         7) 11P30, GLARESHIELD INSTR AND PANEL
         8) 11R3, IND LTS L
         9) 11R30, IND LTS R
      (b) P6 Main Power Distribution Panel:
         1) 6K32, R3 PILOT SECONDARY IND LT
         2) 6L32, L3 PILOT SECONDARY IND LT

   S 422-030
   (2) To install the switch, do these steps:
      (a) Turn the jackscrew counterclockwise until the jackscrew stops before you install the switch module.
      (b) Push the switch module into the case.

         NOTE: Put the side that has the word "TOP" in the up position.

      CAUTION: DO NOT USE THE JACKSCREW TO PULL THE SWITCH MODULE INTO ITS POSITION. THIS CAN CAUSE DAMAGE TO THE SWITCH MODULE AND CASE.

      (c) Push the switch module with your finger while you lightly turn the jackscrew clockwise.
         1) Continue to push the switch module and turn the jackscrew clockwise until the jackscrew is tight.

   S 862-032

   CAUTION: DO NOT USE A DIFFERENT LAMP PART NUMBER. THE INCORRECT LAMP CAN CAUSE DAMAGE TO THE MCP.

   (3) To install a new lamp, put the lamp into the rear of the lens module.

S 432-031
(4) To install the lens module, do these steps:
(a) Put the link pin into the rear of the groove on the bail plate and pull the link pin to the other end of the groove.
(b) NOTE: If installing lens module with P/N 837-0034-XXX, rotate the lens module 90 degrees after placing the bail feature into the grove of the lens module.

(c) Push the lens module into the switch module.

NOTE: Put the side that has the word "TOP" in the up position.

E. Test the mode select switch and lamp.

S 862-033
(1) Supply the electrical power (AMM 24-22-00/201).

S 862-034
(2) Set the two F/D switches on the MCP to OFF.

S 862-035
(3) Remove the DO-NOT-CLOSE tags and close these circuit breakers:
(a) P11 Overhead Circuit Breaker Panel
1) 11B7, STBY INSTR LTS
2) 11E16, MODE CONTROL PANEL
3) 11E18, FLT CONT CMPTR SERVO L
4) 11E21, FLT CONT CMPTR SERVO C
5) 11E34, MODE CONTROL PANEL
6) 11E36, FLT CONT CMPTR SERVO R
7) 11P30, GLARESHIELD INSTR AND PANEL
8) 11R3, IND LTS L
9) 11R30, IND LTS R
(b) P6 Main Power Distribution Panel:
1) 6K32, R3 PILOT SECONDARY IND LT
2) 6L32, L3 PILOT SECONDARY IND LT

S 712-036
(4) Do MCDP ground test 04-MCP (AMM 22-00-02/201).
(a) Make sure no failures occur during test.

S 862-037
(5) Remove the electrical power if it is not necessary (AMM 24-22-00/201).

AUTOPILOT DISENGAGE SWITCHES - REMOVAL/INSTALLATION

1. General
   A. One autopilot disengage switch is found on each control wheel. The removal and installation tasks for each switch are the same.

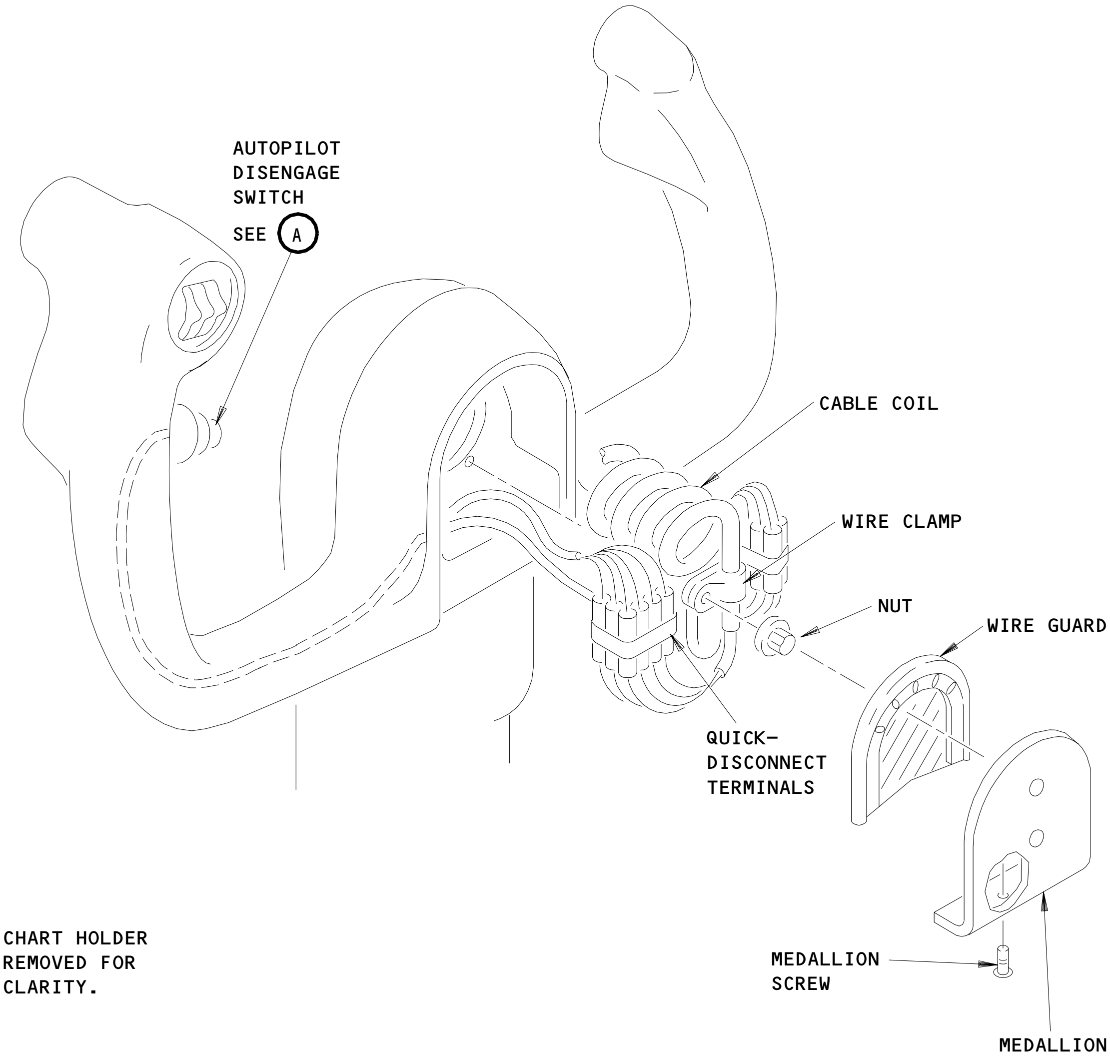
TASK 22-11-03-034-001

2. Remove the Autopilot Disengage Switch (Fig. 401)
   A. Equipment
      (1) Insertion and Extraction tool (Deutsch No. M15570-20).
   B. Consumable Materials
      (1) G00165 Threading wire, a 30-inch length of copper wire (No. 18 to 22 gage), without insulation if possible (used to pull the wire bundle through the control wheel)
   C. Access
      (1) Location Zones
          211/212 Flight Compartment

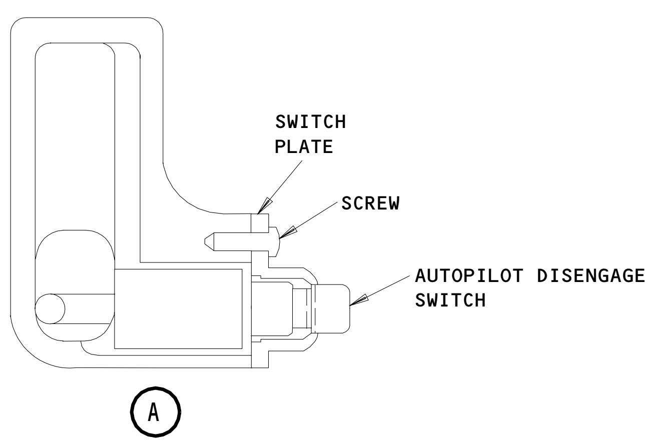
   D. Prepare for Removal

      S 864-002
      (1) Set the two F/D switches on the MCP to OFF.

      S 864-003
      (2) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
         (a) 11A17, AUTOFLIGHT WARN
         (b) 11E16, MODE CONT PNL L
         (c) 11E17, FLT CONT CMPTR PWR L
         (d) 11E18, FLT CONT CMPTR SERVO L
         (e) 11E20, FLT CONT CMPTR PWR C
         (f) 11E21, FLT CONT CMPTR SERVO C
         (g) 11E34, MODE CONT PNL R
         (h) 11E35, FLT CONT CMPTR PWR R
         (i) 11E36, FLT CONT CMPTR SERVO R



NOTE: CHART HOLDER REMOVED FOR CLARITY.



Autopilot Disengage Switch Installation
Figure 401

E. Remove the Autopilot Disengage Switch

S 034-004

(1) Remove the screw that attaches the medallion to the control wheel.

S 034-005

(2) Remove the medallion with chart holder.

NOTE: It is not necessary to remove the chart holder from the medallion.

S 034-006

(3) Remove the nuts to disconnect the wire clamps.

S 034-007

(4) Pull the quick-disconnect terminals away from the control wheel.

S 034-008

(5) Use the extraction tool to disconnect these eight wires from the switch side of the quick-disconnect terminals:

WIRE LABEL AND COLOR CODE TABLE FOR SWITCH MFG P/N 721101-A1	
SWITCH SIDE WIRE LABEL	AIRFRAME SIDE WIRE COLOR CODE
BRN	WHITE/BROWN
YEL	WHITE/YELLOW
GRN	WHITE/GREEN
BKB	WHITE/BLACK-BLUE
BKY	WHITE/BLACK-YELLOW
BKA	WHITE
BLU	WHITE/BLUE
BKG	WHITE/BLACK-GREEN

WIRE LABEL AND COLOR CODE TABLE FOR SWITCH MFG P/N SW43896	
SWITCH SIDE WIRE COLOR	AIRFRAME SIDE WIRE COLOR CODE
BROWN	WHITE/BROWN
YELLOW	WHITE/YELLOW
GREEN	WHITE/GREEN
BLACK	WHITE/BLACK-BLUE
ORANGE	WHITE/BLACK-YELLOW
WHITE	WHITE
BLUE	WHITE/BLUE
GRAY	WHITE/BLACK-GREEN

S 434-009
(6) Attach one end of the threading wire around the wire bundle (eight wires) that is connected to the switch.

S 034-010
(7) Remove the screw that holds the switchplate to the control wheel.

S 024-011
(8) Remove the switchplate assembly and pull the wires through the control wheel.

NOTE: Permit approximately 3 or 4 inches of the threading wire to come out of the control wheel.

S 034-012
(9) Remove the threading wire from the switch wire bundle.

S 034-013
(10) Remove the switchplate from the switch.

TASK 22-11-03-404-014
3. Install the Autopilot Disengage Switch (Fig. 401)
A. Equipment
(1) Insertion and Extraction tool (Deutsch No. M15570-20).
B. Consumable Materials
(1) G00165 Threading wire, a 30-inch length of copper wire (No. 18 to 22 gage), without insulation if possible (used to pull the wire bundle through the control wheel)
(2) C00259 Primer, BMS 10-11, Type 1

C. References
   (1) 20-30-03/201, Finishing Materials
   (2) 22-00-02/201, Autoflight BITE
   (3) 24-22-00/201, Electrical Power - Control

D. Access
   (1) Location Zone
       211/212 Flight Compartment

E. Install the Autopilot Disengage Switch

S 644-015

(1) Put primer on the threads of the switchplate and switch (Ref 20-30-03).

S 424-028

(2) Install the switchplate on the switch.

S 434-016

(3) Attach the end of the treading wire that comes out of the switch hole on the control wheel around the switch wire bundle.

S 424-017

(4) Pull the switch wire bundle down through the control wheel with the other end of the threading wire.

S 034-018

(5) Remove the threading wire from the switch wire bundle.

S 434-019

(6) Use the insertion tool to connect the switch wires to the quick-disconnect terminals. Refer to the WIRE LABEL AND COLOR CODE TABLE.

S 434-020

CAUTION: MAKE SURE THE WIRE CLAMP IS INSTALLED IN THE POSITION SHOWN ON FIG. 401. PUT THE CABLE COIL IN THE CENTER OF THE CONTROL WHEEL. FRICTION WILL RESULT IF THE WIRE CLAMPS ARE NOT INSTALLED CORRECTLY AND THE CABLE COIL IS NOT PUT IN THE CENTER. DAMAGE TO EQUIPMENT CAN OCCUR.

(7) Install the wire clamp into the control wheel with the nut.

NOTE: Install the wire clamp so the cable coil is in the center of the control wheel.

(a) Tighten the nut to 20 pound-inches.
(b) Put the quick-connect terminals into the control wheel.

S 434-030

CAUTION: ENSURE THAT WIRE GUARD ASSEMBLY SLIDES SMOOTHLY IN GROOVE OF WHEEL ASSEMBLY DURING INSTALLATION. KEEP ALL WIRES AND CONNECTORS CLEAR OF PLANE OF GROOVE TO PREVENT POSSIBLE WIRE INSULATION DAMAGE DURING ASSEMBLY.

(8) Install wire guard assembly. Insert lower end of wire guard assembly inside base of wheel. Firmly press upper edge of seal until guard is below medallion groove.

S 434-021

(9) Install the medallion and chartholder with the screw.

S 434-022

(10) Install the switchplate assembly with the screw.

F. Test the Autopilot Disengage Switch.

S 864-023

(1) Supply electrical power (Ref 24-22-00).

S 864-024

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
   (a) 11A17, AUTOFLIGHT WARN
   (b) 11E16, MODE CONT PNL L
   (c) 11E17, FLT CONT CMPTR PWR L
   (d) 11E18, FLT CONT CMPTR SERVO L
   (e) 11E20, FLT CONT CMPTR PWR C
   (f) 11E21, FLT CONT CMPTR SERVO C
   (g) 11E34, MODE CONT PNL R
   (h) 11E35, FLT CONT CMPTR PWR R
   (i) 11E36, FLT CONT CMPTR SERVO R

S 724-025

(3) Do MCDP ground test 11-SW A/P DISC (Ref 22-00-02).
   (a) Make sure no failure messages show.

S 864-026

(4) Set the MCDP to off.

S 864-027

(5) Remove electrical power if not necessary (Ref 24-22-00).

AFCS GO-AROUND SWITCHES - REMOVAL/INSTALLATION

1. General
   A. Six AFCS go-around switches are found in the engine thrust levers. Three switches are installed in the side of each thrust lever. The Removal/Installation steps are the same for each set of switches.

TASK 22-11-04-004-017

2. Remove the AFCS Go-Around Switches (Fig. 401)
   A. Reference
      (1) 20-10-26/201, Heat Guns, Soldering Guns and/or Soldering Irons.
   B. Access
      (1) Location Zones
          211/212 Flight Compartment

   C. Remove the AFCS Go-Around Switch

      S 864-001
      (1) Set the two F/D switches on the MCP to OFF.

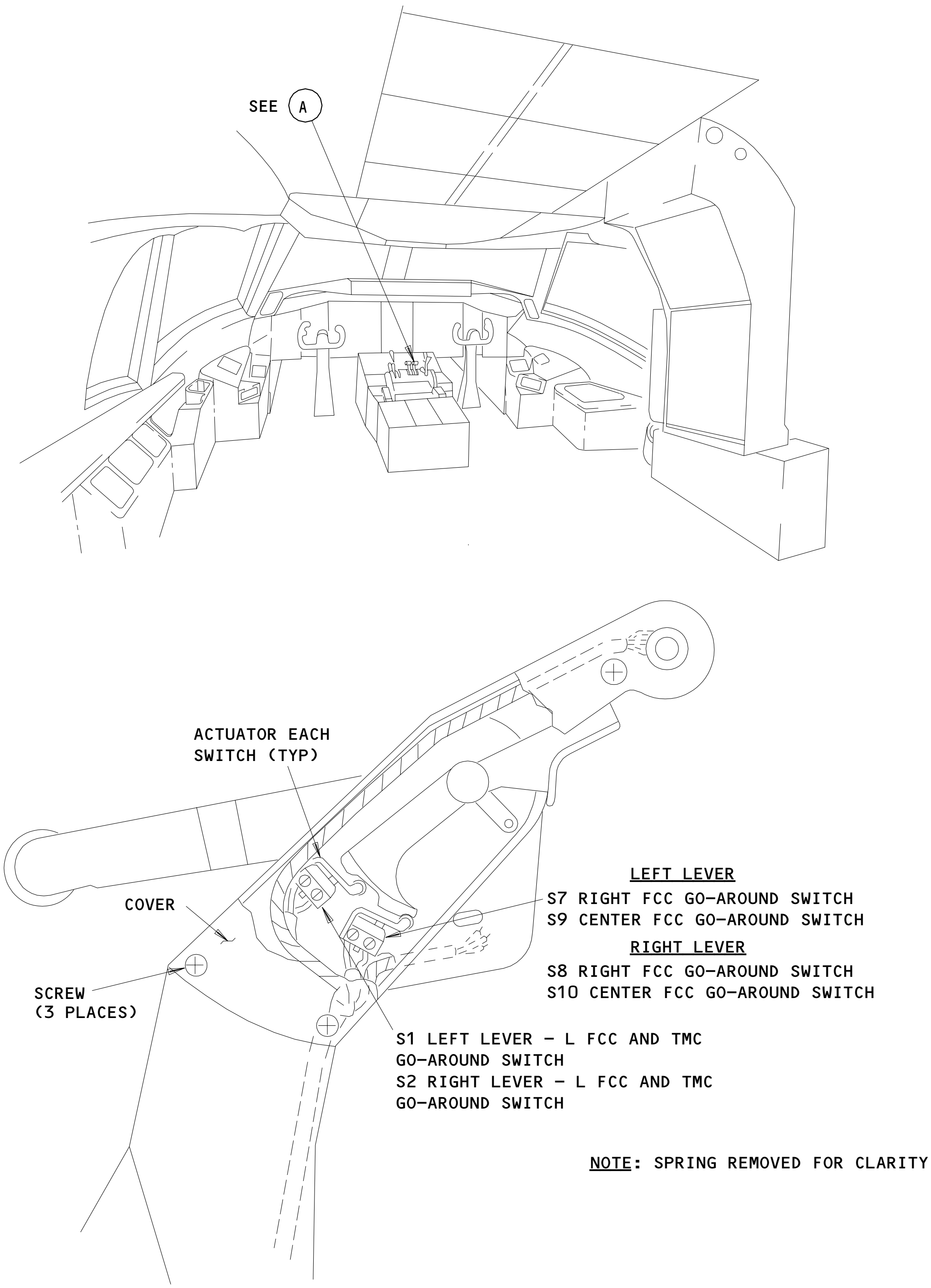
      S 864-002
      (2) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11A17, AUTOFLIGHT WARN
          (b) 11E16, MODE CONT PNL L
          (c) 11E17, FLT CONT CMPTR PWR L
          (d) 11E18, FLT CONT CMPTR SERVO L
          (e) 11E20, FLT CONT CMPTR PWR C
          (f) 11E21, FLT CONT CMPTR SERVO C
          (g) 11E34, MODE CONT PNL R
          (h) 11E35, FLT CONT CMPTR PWR R
          (i) 11E36, FLT CONT COMPTR SERVO R
          (j) 11F14, TMC AC
          (k) 11F15, TMC DC
          (l) 11F16, TMC SERVO

      S 034-003
      (3) Remove the screws and cover from the thrust lever.

      S 024-004
      (4) Remove the screws from the applicable switch and actuator assembly.

      S 034-005
      (5) Remove the heat-shrink tubing from the wire terminals.

      S 034-018
      (6) Unsolder the wires from the switch (Ref 20-10-26).



AFCS GO-AROUND SWITCH
A

AFCS Go-Around Switches Installation
Figure 401

TASK 22-11-04-404-006

3. Install the AFCS Go-Around Switches (Fig. 401)

A. Equipment

(1) Heat gun - Commercially available.

B. Consumable Materials

(1) G02104 Heat Shrink Tubing, R876, color yellow,
Raychem Corp., 300 Constitution Dr.,
Menlo Park, CA 94015

C. References

(1) 20-10-26/201, Heat Guns, Soldering Guns and/or Soldering Irons

(2) 22-00-02/201, Autoflight Bite

(3) 24-22-00/201, Electrical Power - Control

D. Access

(1) Location Zones

211/212 Flight Compartment

E. Install the AFCS Go-Around Switch

S 434-007

(1) Put the heat-shrink tubing on the switch wires.

S 424-008

(2) Solder the wires to the new switch (Ref 20-10-26).

S 434-009

(3) Slide the tubing over the switch terminals.

S 434-010

(4) Shrink the tubing with the heat gun.

S 424-011

(5) Install the switch and actuator assembly with the screws.

S 434-012

(6) Install the cover on the thrust lever with the screws.

F. Test the Go-Around Switches

S 864-013

(1) Supply electrical power (Ref 24-22-00).

S 864-019

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:

(a) 11A17, AUTOFLIGHT WARN

(b) 11E16, MODE CONT PNL L
(c) 11E17, FLT CONT CMPTR PWR L
(d) 11E18, FLT CONT CMPTR SERVO L
(e) 11E20, FLT CONT CMPTR PWR C
(f) 11E21, FLT CONT CMPTR SERVO C
(g) 11E34, MODE CONT PNL R
(h) 11E35, FLT CONT CMPTR PWR R
(i) 11E36, FLT CONT COMPTR SERVO R
(j) 11F14, TMC AC
(k) 11F15, TMC DC
(l) 11F16, TMC SERVO

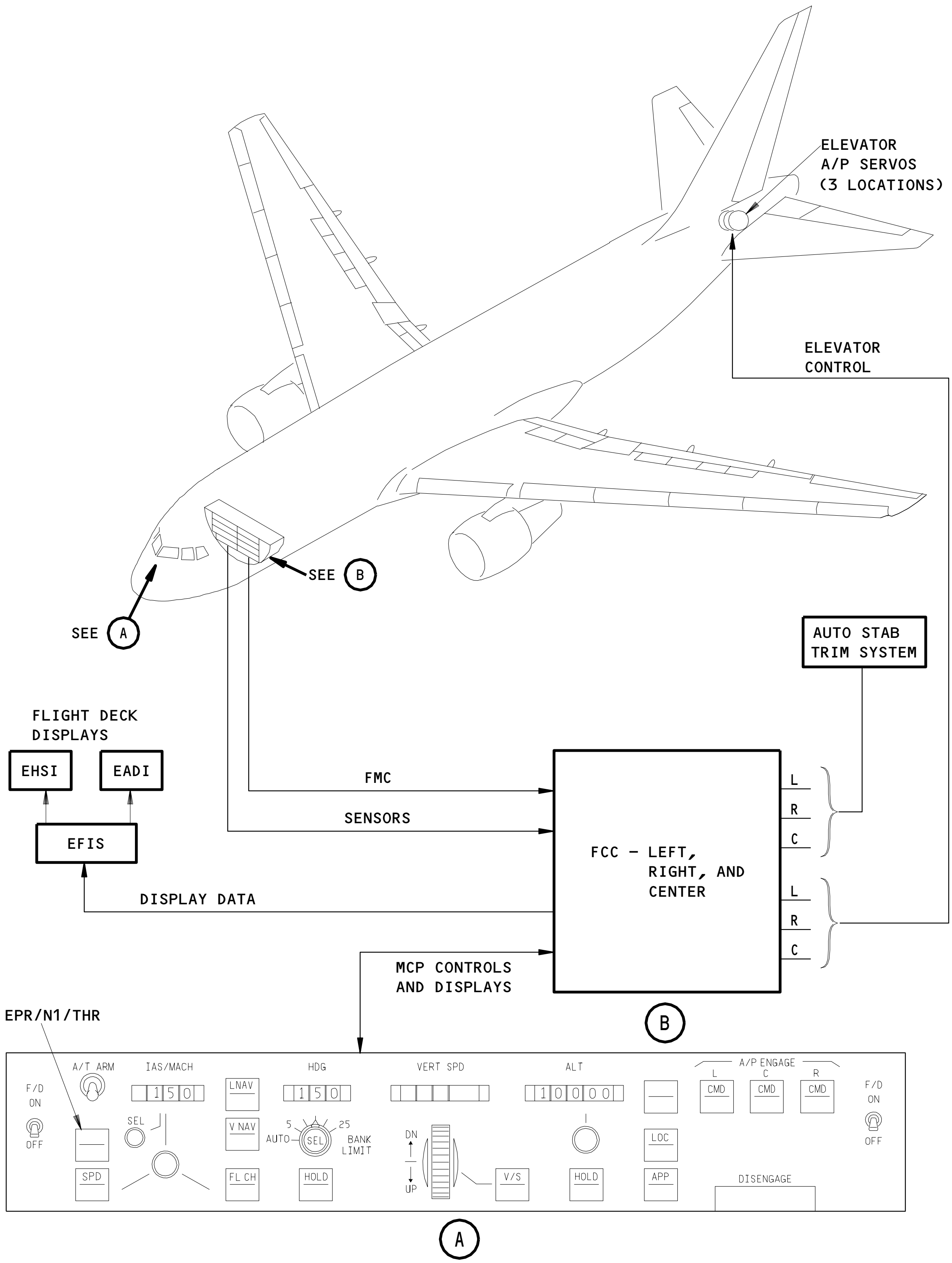
S 724-014
(3) Do MCDP ground test 13-SW G/A (Ref 22-00-02).
(a) Make sure no failure messages show during the test.

S 864-015
(4) Set the MCDP to off.

S 864-016
(5) Remove electrical power if it is not necessary (Ref 24-22-00).

AUTOPILOT/FLIGHT DIRECTOR PITCH CHANNEL - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The pitch channel consists of pitch mode switches on the AFCS mode control panel, pitch modes of the three Flight Control Computers (FCCs) and three Elevator Autopilot Servos (EAS). Each servo is dedicated to a single flight control computer. Primary AFDS control modes are selected on the AFCS Mode Control Panel (MCP). Flight director and mode displays are on the Electronic Attitude Director Indicators (EADIs).
   B. Each FCC generates commands for the Flight Director (F/D) and Autopilot (A/P). The FCC selected provides commands for either the A/P or F/D or both at the same time. The A/P and F/D modes are selected on the MCP. F/D displays are turned on by the F/D switches on the MCP. The FCC providing F/D displays is selected by the pilots FLT DIR source select switch.
      (1) Pitch axis control functions of the FCC provide pitch axis stability, manual pitch attitude control and programmed pitch attitude control of the airplane. The functions control the autopilot or provide flight director steering commands. The autopilot pitch control servos and elevator control system provide for pitch axis maneuvering. The automatic stabilizer trim system provides for the transfer of sustained elevator loads to the horizontal stabilizer. A Stabilizer trim and Aileron Lockout Module (SAM) provides the interface between the FCC and the stabilizer trim system.
      (2) Pitch Axis Stability
         (a) The attitude hold function of the FCC maintains airplane pitch attitude during turbulence with the autopilot engaged or provides F/D commands.
      (3) Manual Pitch Attitude Control
         (a) The basic mode of the pitch channel is vertical speed (V/S). Altitude hold is V/S equal to zero. Both are active whenever a flight director is on or an autopilot is engaged in CMD. Altitude hold is manually selected by pressing the HOLD button on the MCP. Vertical speed is manually selected by pressing the V/S button on the MCP or is automatically selected when CMD is engaged without selecting a mode. Altitude hold is automatic whenever the vertical speed is less than 300 feet per minute.



Pitch Axis Control
Figure 1

(b) Altitude select control is available whenever a flight director is on or an autopilot is engaged in CMD. A vertical speed may be selected and V/S pushbutton pressed. The airplane then climbs or descends to the altitude selected on the MCP. If in a climb or descent at the time the F/D is on or the autopilot is engaged, the airplane continues at the existing vertical speed until the selected altitude is reached. The mode then automatically transfers to altitude hold. The V/S or HOLD pushbuttons do not have to be pressed if the vertical speed and altitude differences are greater than altitude hold capture limits.

(4) Programmed Pitch Attitude Control

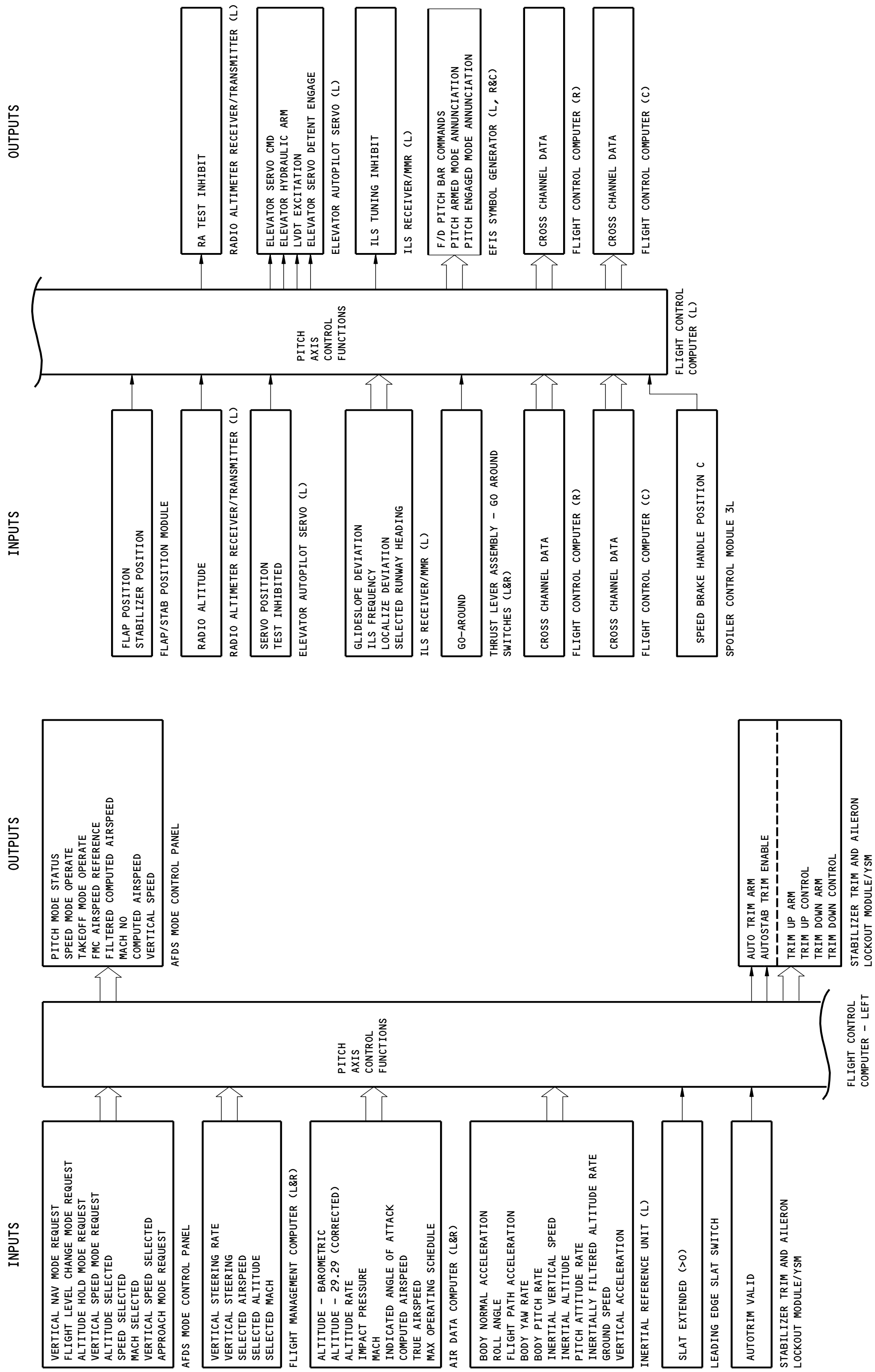
(a) Any of the programmed pitch attitude control functions can be used with a flight director engaged or an autopilot engaged in CMD. During sustained elevator deflections, the stabilizer may be automatically retrimmed. The FCC generates auto stabilizer trim commands for the SAM. The SAM drives the stabilizer trim system until FCC pitch attitude circuits return elevators to neutral and remove auto stab trim commands (Ref 22-22-00).

(b) The flight management computer system provides pitch steering commands when VNAV is selected. The airplane is then guided to the vertical flight profile programmed in the FMCS.

(c) Programmed pitch attitude control functions provide autopilot or flight director guidance to capture and track the glideslope centerline during an ILS approach. Additional programmed functions provide automatic compensation for windshear and perform the flare maneuver prior to touchdown. Vertical position correction provides a consistant touchdown point on the runway. A controlled pitchdown attitude maneuver lowers and holds the nose wheel down after the main gear touchdown. If a go-around is initiated, flight director or autopilot guidance establishes a pitch-up attitude and positive rate of climb. Go-around mode disengages at 400 feet or when another mode is selected. A selected vertical mode is then used for airplane guidance.

C. Flight Control Computer Pitch Axis Control Inputs and Outputs (Fig. 2)

(1) Inputs include mode requests from the MCP, servo and surface position from the APCS, flight reference data from IRUs and ADCs, and navigation data from ILS and RA systems. Vertical steering, rate target altitude, and airspeed/mach commands are supplied by the FMC. Auto stabilizer trim data is received from the FLAP/STAB POSITION module, and Stabilizer Trim and Aileron Lockout Module.



FCC/Systems Interfaces (Pitch)
Figure 2

(2) The FCC outputs include mode status discretes, airspeed/mach number, and vertical speed data to the MCP. Autopilot pitch commands are sent to the elevator autopilot servos and stabilizer trim and aileron lockout module. Flight director pitch bar command, pitch armed and engaged mode annunciation data is sent to the EFIS symbol generators (Ref 34-22-00).

2. Component Details

A. Elevator Autopilot Servo Locations and Linkages (Fig. 4)

(1) The three elevator autopilot servos are in section 48 of the fuselage below the elevator hinge line. The left EAS is located outboard of the aft left quadrant torque tube assembly. The center EAS is above the right quadrant assembly.

(2) The left and center EAS output levers are connected to the left quadrant assembly. The right EAS output lever is connected to the right quadrant assembly. The captain's control column is connected with cables to the left quadrant assembly. The first officer's control column is connected with cables to the right quadrant assembly. The two quadrants are ganged together through an interconnecting rod and asymmetry limiters.

(3) All EASs are mounted on the bulkhead immediately aft of the horizontal stabilizer. Each EAS is installed with 4 mounting bolts. An output rod connects the EAS crank to the quadrant assemblies. Two hydraulic lines and an electrical connector complete the installation.

(4) SAS 050-149;
autopilot operation of one or more EASs causes a mechanical feedback to drive the control columns and operates the elevator control system. The neutral shift linkage causes the elevator neutral reference point to vary in proportion to stabilizer position. The neutral shift linkage, driven by the stabilizer, causes an output through the feel and centering mechanism. The feel and centering mechanism is connected to each aft quadrant torque tube assembly. The repositioned output of the feel and centering mechanism causes the quadrant to rotate and drive the linkages to the elevator power control actuators (PCA)s. The PCAs then position the elevators either up or down to the new neutral reference point. The EAS then synchronize to the new elevator neutral position.

Not Used
Figure 3

B. Elevator Autopilot Servo (Fig. 5)

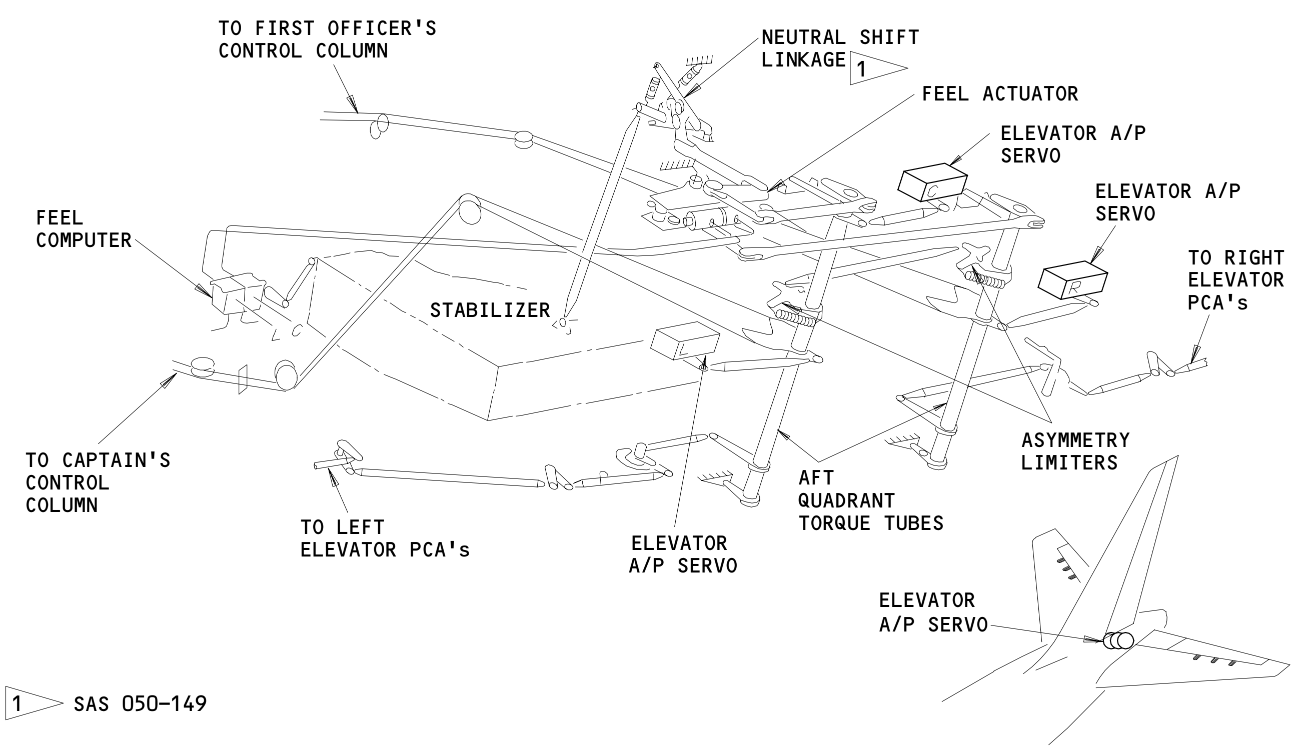
(1) Each elevator autopilot servo has two solenoid valves and an electrohydraulic servovalve. Two linear variable differential transducers (LVDTs) provide position feedback signals representing the position of the actuator piston and output crank (lever) on the EAS. The EAS also has a pressure regulator and a 25 micron filter.

(2) Electrohydraulic Control Valves

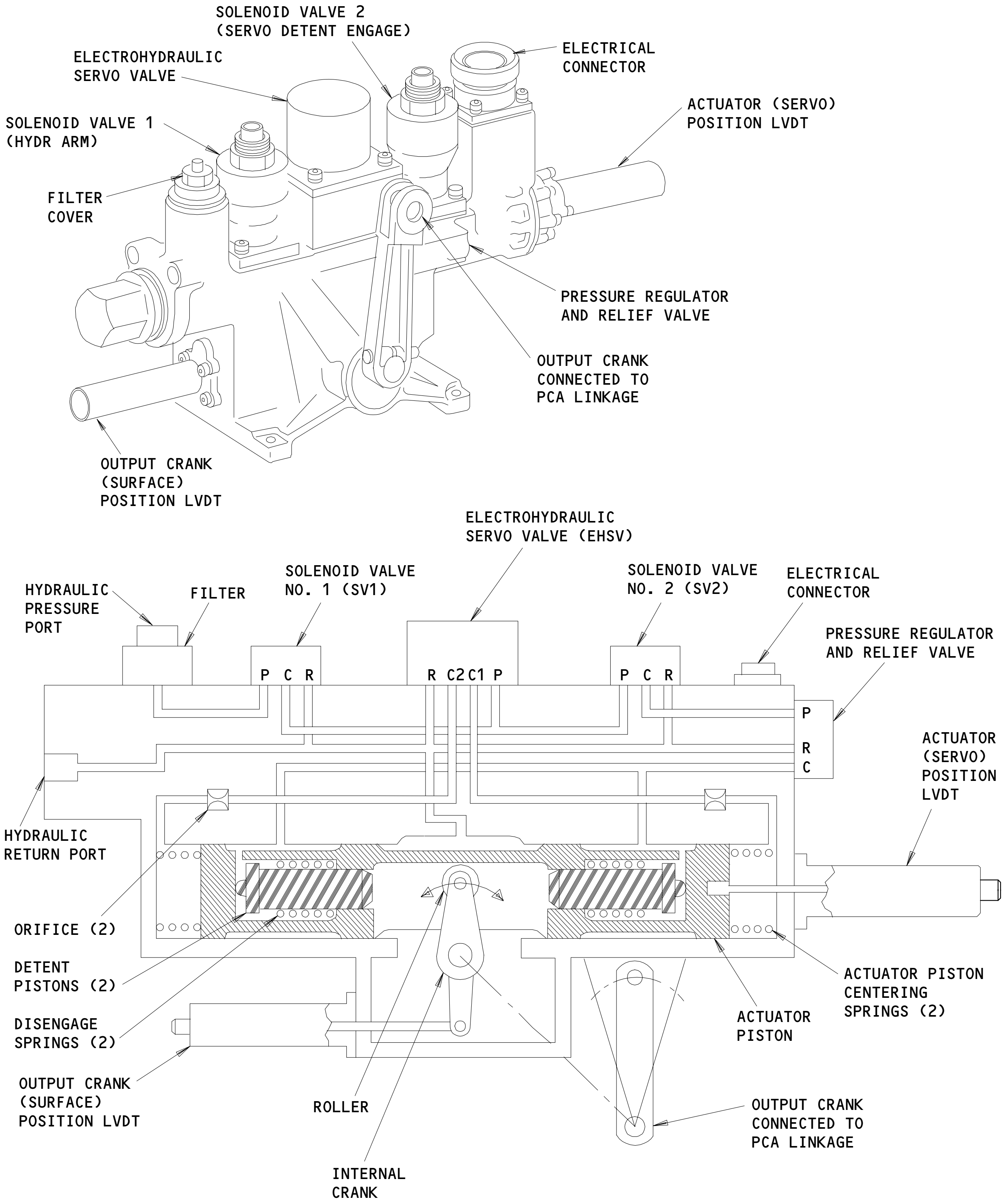
(a) Solenoid Valves

1) Two solenoid valves (SV1 and SV2) are installed on each EAS. Each solenoid valve is opened electrically to provide hydraulic pressure to the EAS when autopilot arm and engage logic circuits are completed. Each valve is installed with 4 bolts and sealed with a gasket plate. Electrical pins mate when the solenoid is bolted in position.

2) Solenoid valve number 1 (SV1) opens when ELEV HYD ARM DC is applied (autopilot is armed). It ports hydraulic pressure to solenoid valve number 2 (SV2) and the electrohydraulic servovalve (EHSV). Solenoid valve number 2 opens when ELEV DETENT ENG DC is applied (autopilot is engaged in CMD) and ports hydraulic pressure to the detent pistons. The detent pistons clamp the output linkage crank and transmit actuator piston position to the output crank.



Elevator Autopilot Servo Locations and Linkages
Figure 4



Elevator Autopilot Servo
Figure 5

(b) Electrohydraulic Servovalve (EHSV)

1) One electrohydraulic servovalve is installed on each EAS. The EHSV contains a sealed torque motor, a feedback spring, a projector jet and a piston. Hydraulic pressure through the valve can be ported to either of two outputs. When no error signal is applied to the torque motor, a small amount of hydraulic fluid flows through a flex pipe attached to the torque motor and out the projector jet. From the jet, equal pressure is applied to opposite ends of the piston holding it centered and closing both output ports.

2) When an error signal is applied, the torque motor armature rotates in proportion to the magnitude and direction of the input signal which moves the projector jet accordingly. The jet directs more hydraulic pressure to one end of the piston than the other causing it to move and opens the corresponding output port to complete hydraulic fluid flow through the EAS. When the error signal is nulled, the motor armature and jet return to center. This equalizes the pressures on the piston and, with the aid of the feedback spring, causes the piston to recenter and close both output ports.

3) The EHSV is installed with 4 bolts and sealed with a gasket plate. Electrical pins are mate when the EHSV is bolted in position.

4) The EHSV is controlled by the elevator error signal (ELEV SERVO CMD) from the FCC. In response to this command, the EHSV ports hydraulic pressure to either side of the actuator piston. Feedback from the actuator piston LVDT nulls the error signal at the FCC and piston movement stops.

(3) Linear Variable Differential Transducers (LVDT)

(a) The EAS has two LVDTs. The actuator piston (servo position) LVDT functions as a linear follow-up transmitter for closing the loop around the EHSV. It is operated by the actuator piston. The output position (surface position) is operated by the internal crank which is connected to the output crank and elevator control linkage. Both LVDTs are variable reluctance transformers with an output that varies directly with linear motion. The LVDTs use 26v ac excitation from the associated FCC. The LVDTs are not considered line replaceable units since they require nulling adjustments to be completed to match actuator piston position with output crank position.

(4) Pressure Regulator and Relief Valve

(a) The pressure regulator and relief valve regulates and limits hydraulic pressure applied to the actuator piston and detent piston. The pressure relief function allows manual inputs from the control linkage to override autopilot control.

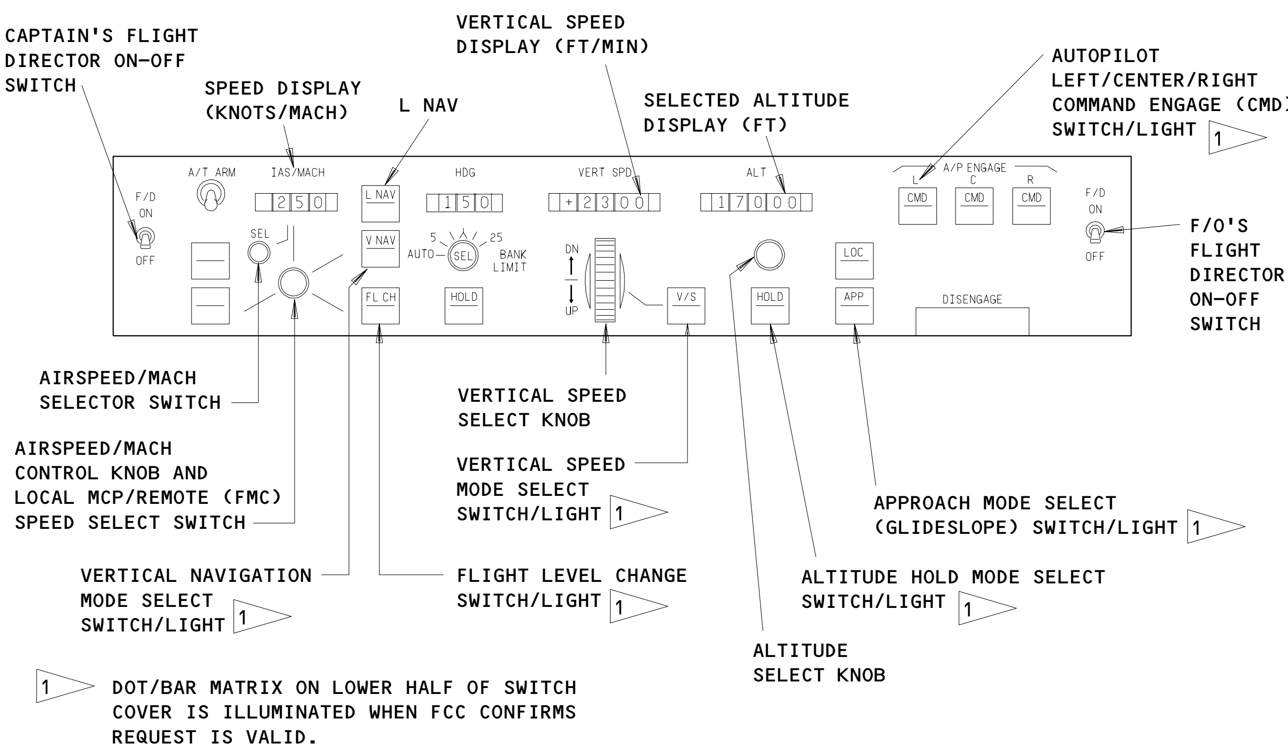
(5) Actuator Piston Assembly
   (a) The actuator piston translates autopilot input commands through the EHSV into mechanical positioning of the elevators. The actuator piston is normally centered by two springs and is moved by hydraulic pressure from the EHSV when the autopilot is armed or engaged. The springs center the actuator piston when pressure is released.
   (b) Two detent pistons inside the actuator piston are normally retracted by disengage springs. When the autopilot is engaged, SV2 opens. Hydraulic pressure through SV2 overrides spring tension and locks the detent pistons against the roller of the internal crank. Detent pistons may be forced back if sufficient force to overcome hydraulic pressure is applied manually through the control linkage (camout).
   (c) The actuator piston LVDT provides an electrical signal proportional to actuator piston position. This signal nulls the autopilot elevator position error signal from the FCC to stop movement of the actuator piston.
(6) Cranks
   (a) The internal crank roller is clamped by the detent pistons when the autopilot is engaged. The crank moves with the actuator piston. Motion of the internal crank moves the output position LVDT. One end of the output crank is directly connected to the internal crank at a common pivot point. The other end of the output crank is connected to the elevator linkage.
(7) Mechanical Control Sequence
   (a) Initially the actuator piston is fixed by the centering springs. With the autopilot not engaged (armed), SV1 is open, SV2 is closed, detent pistons are disengaged and the internal crank is free to move within the piston cavity. The output position LVDT provides internal crank position to the FCC for autopilot synchronization to elevator position. The FCC commands through the EHSV cause the actuator piston to follow the internal crank so that the internal crank remains centered within the piston cavity. The actuator piston position LVDT nulls the command signal to stop the actuator piston.
(8) Autopilot Control Sequence
   (a) With the autopilot engaged, SV1 and SV2 are open, the detent pistons are pressurized and the internal crank is clamped on the center of the actuator piston. When the EHSV receives a command from the FCC, hydraulic pressure is ported to one end of the actuator piston. The detent pistons carry the internal crank with the actuator piston to its commanded position. The output crank moves the linkage to the elevator power control actuators while the output position LVDT sends position information back to the FCC to null the command signal and stop the elevator.

(9) Camout
   (a) Camout occurs when the output crank position (surface position) does not correspond to the actuator piston position (servo position) as sensed by differing LVDT outputs. Causes of camout are mechanical jamming of output linkage, pilot override of an engage autopilot channel, or mechanical override of one autopilot channel by the remaining two during multi-channel operation.
      1) The force necessary on the control column for the pilot to manually disengage the autopilot is 24.8 ±0.39 pounds for each autopilot.

3. Operation

A. Functional Description

(1) Mode Control Panel - Pitch Controls (Fig. 6)
   (a) Flight Director ON/OFF switches
      1) Two F/D toggle switches control the left and right flight director displays.
   (b) Command Engage switches
      1) Three CMD ENGAGE switch/lights (L, C, R) provide inputs for command engage logic to dedicated FCCs.
   (c) The following MCP controls are associated with the pitch function:
      1) VNAV switch/light
      2) FLCH switch/light (Flight Level Change)
      3) Vert SPD select thumbwheel



Mode Control Panel - Pitch Control
Figure 6

4) V/S SPD switch/light
5) ALT Select Knob
6) ALT HOLD switch/light
7) APP switch/light
8) IAS/MACH Control Knob and local (MCP)/remote (FMC) speed select switches
9) IAS/MACH select switch

(2) Pitch Mode Functions (Fig. 7)

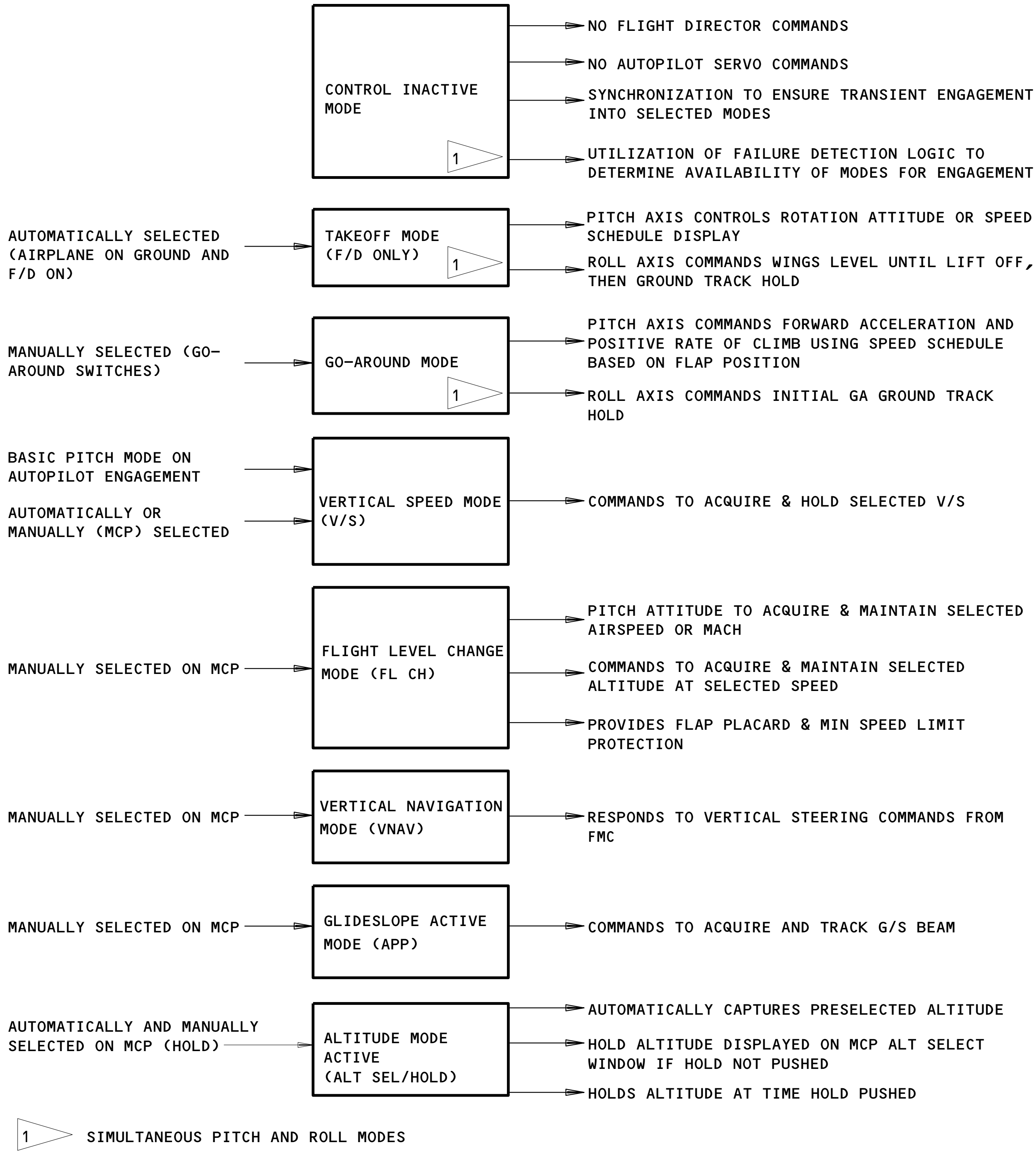
(a) The control inactive mode is the dormant condition of the AFDS when electrical power is on and no A/P or F/D commands are generated. Control loops are synchronized to control surface position for transient-free engagement. Failure detection logic determines the availability of each mode.

(b) Takeoff Mode (F/D Mode Only)

1) Takeoff mode is automatically selected when the airplane is on the ground and either or both flight directors are ON. The takeoff mode consists of two submodes: takeoff on-ground, and takeoff in-air. The takeoff on-ground submode provides the F/D with 8 degrees pitch up and wings level roll commands during the ground roll and until 2 seconds after radio altitude of 5 feet. The takeoff in-air submode begins 2 seconds after a radio altitude of 5 feet. The F/D pitch up command is washed out over a 3-second time period and replaced with pitch attitude commands referenced to scheduled airspeed. Airspeed commands are scheduled according to flap position and engine failure. This ensures safe F/D pitch attitude commands until the selected airspeed on the MCP is obtained. The F/D roll command is changed to provide ground track hold.

(c) Go-Around Mode

1) Go-around mode is selected by pressing the go-around switches on the thrust levers. The go-around mode provides A/P and F/D pitch commands for a safe positive rate of climb using speed scheduling based on flap position. The A/P and F/D roll commands provide ground track hold based on runway heading with crosswind compensation.



Pitch Mode Functions
Figure 7

(d) Vertical Speed (V/S) Mode
   1) Vertical speed is the basic pitch axis mode. When the V/S switch/light is pressed with the A/P in CMD or the F/D on, the vertical speed command acquires the selected vertical speed. The existing V/S mode is automatically acquired when the F/D is initially turned on or the F/D is overridden by engaging the A/P in CMD.

(e) Flight Level Change (FLCH) Mode
   1) The flight level change mode is manually selected by pressing the FLCH switch/light on the MCP. The mode uses software programmed elevator control laws to generate pitch attitude A/P or F/D commands. The pitch attitude commands control the airplane to obtain and maintain a MCP selected altitude using the MCP selected airspeed or mach. The FLCH mode provides flap placard and minimum speed limit protection. When limits are used, corresponding annunciation is provided on the EADI.

(f) Vertical Navigation (VNAV) Mode
   1) The mode is manually selected by pressing the VNAV switch/light on the MCP. The A/P or F/D responds to pitch steering commands from the FMC. Pitch steering commands are generated in the FMC as necessary to acquire and track a vertical path along a preplanned flight route.

(g) Glideslope Active Mode (APP)
   1) The mode is manually selected by pressing the APP switch/light on the MCP. Pitch commands are generated for A/P or F/D control to acquire and maintain capture of the glideslope. Each available off-line autopilot channel automatically transitions to arm mode for multi-channel operation when APP is selected.

(h) Autoland
   1) The autoland is only active during a multi-channel approach. Pitch commands provide flare and nose gear letdown during rollout.

(i) Altitude Mode Active
   1) The mode is manually selected for the altitude hold portion. It is automaticaly selected when the airplane is within the computed capture range.

(3) Output Signal Synchronization (Fig. 8)

(a) Elevator Autopilot Servo Command

1) Servo command signals are synchronized to the position of the servo to prevent transients upon engagement. Elevator surface position is summed with the elevator command output which is initially at zero. Assuming the elevator is displaced, the increase in elevator position LVDT output is integrated and limited to +10 and -28 degrees of elevator displacement. The resulting output is summed at the integrator until LVDT input is nulled.

2) The resulting elevator command signal is applied to the pitch servo amplifier to drive the electrohydraulic servovalve. With elevator servo hydraulic arm logic applied, the electrohydraulic servovalve moves to center the actuator piston (servo actuator) about the output crank linkage. The servo position LVDT provides feedback to null the elevator command signal and stop the servo actuator. At A/P engage, elevator servo detent engage logic is applied to the servo. Servo detent pistons then clamp the actuator piston to the output crank linkage without generating a transient. The A/P engage logic transfers pitch steering commands to the elevator trim integrator for A/P engaged operation.

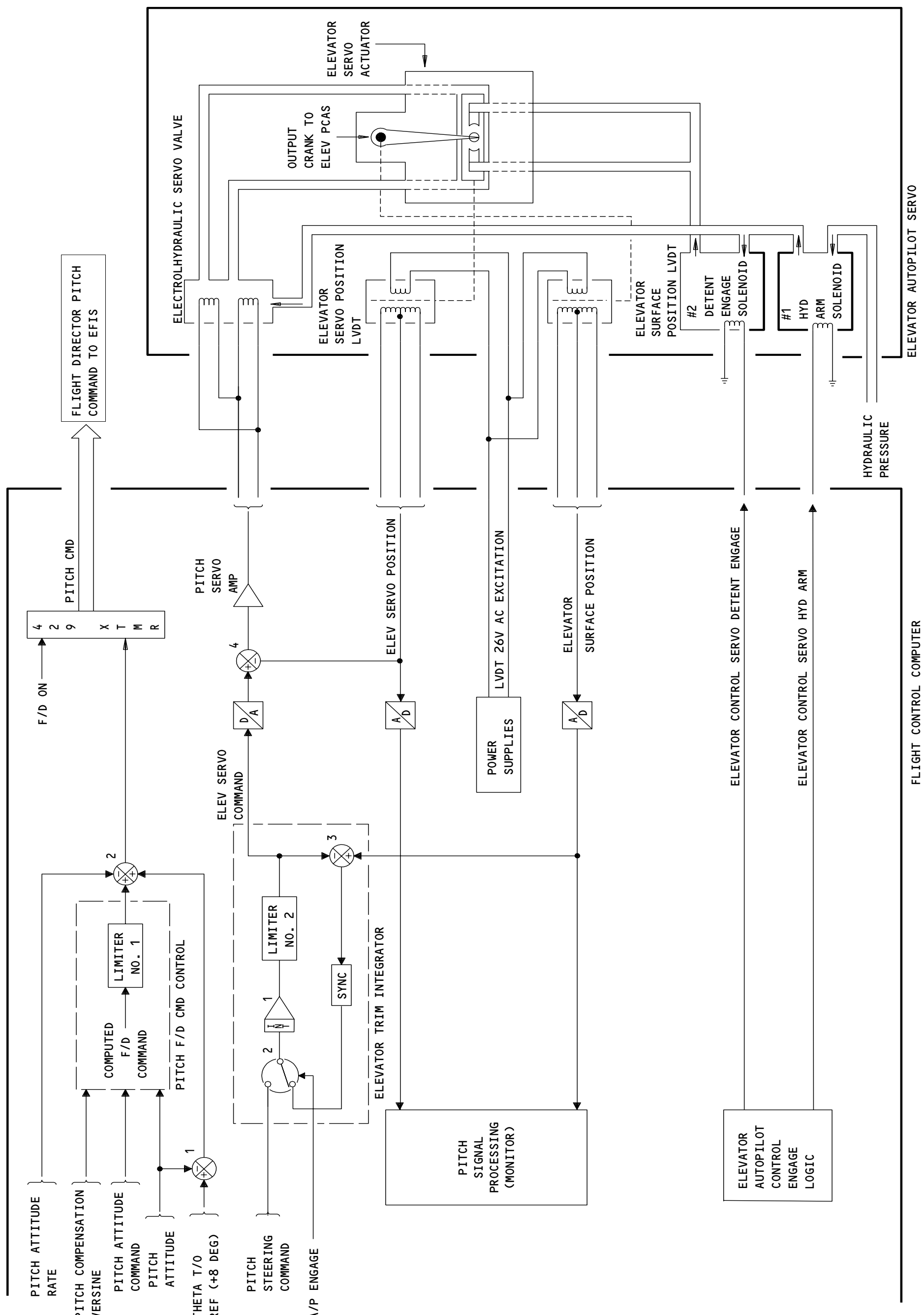
B. Control

(1) Flight Director Pitch Channel (Fig. 9)

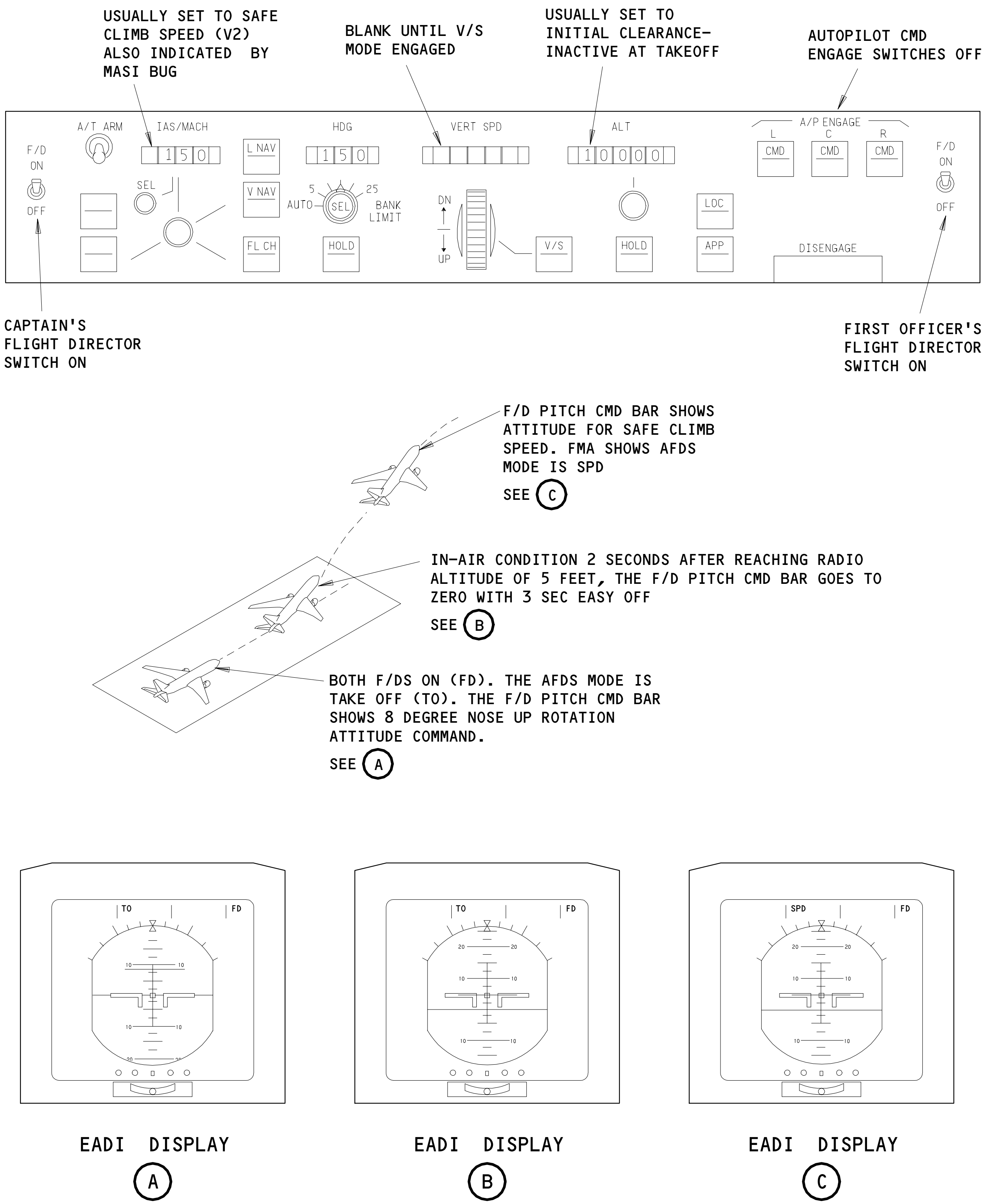
(a) Flight director pitch commands are computed continuously regardless of whether or not the A/P is engaged or the F/D is on. Positioning the F/D switches ON enables flight director displays for all available modes.

(b) The VNAV, V/S, ALT SEL, ALT HOLD, FLCH and APP modes are available as F/D modes. The IAS/MACH speed selection is available for the FLCH mode. The pilot controls the airplane in response to F/D commands on the EADI directly with the control column/wheel.

(c) The F/D pitch commands are computed from selected pitch attitude commands, present pitch attitude, pitch attitude rate, and versine. Computed F/D pitch commands are summed with pitch rate. If in takeoff (T/O) mode, a reference +8 degrees (theta) pitch signal is summed with pitch attitude and then summed with the F/D pitch commands. Computed F/D pitch commands are applied to the signal generators and EADIs of the EFIS.



Output Signal Synchronization
Figure 8



Flight Director - Pitch Channel
Figure 9

(d) A F/D ON discrete causes the F/D command bars to be in view and the F/D pitch command displayed. With the F/D switches off, the F/D ON discrete is not provided and the EFIS inhibits the F/D displays.

(e) The flight director switches on the MCP cause the flight director bars to appear on the EADI for the side selected. The flight director bars provide the pilots information for manually flying a preselected flight profile. If the F/D is switched ON while the aircraft is on the ground, takeoff mode will automatically engage. If the F/D is switched ON in flight and no other mode is selected, the vertical speed mode automatically engages.

(f) The flight director EADI indications are controlled by the FCC selected with the FLT DIR instrument select switch. The FCC controls the flight director or the autopilot in CMD, or both at the same time.

(2) Pitch Channel - Takeoff Mode

(a) Takeoff mode is automatically engaged when either or both flight director switches are switched ON while the airplane is on the ground. Takeoff mode displays appropriate flight director bar commands for proper rotation and climbout. The flight director remains in takeoff mode until another valid mode is engaged.

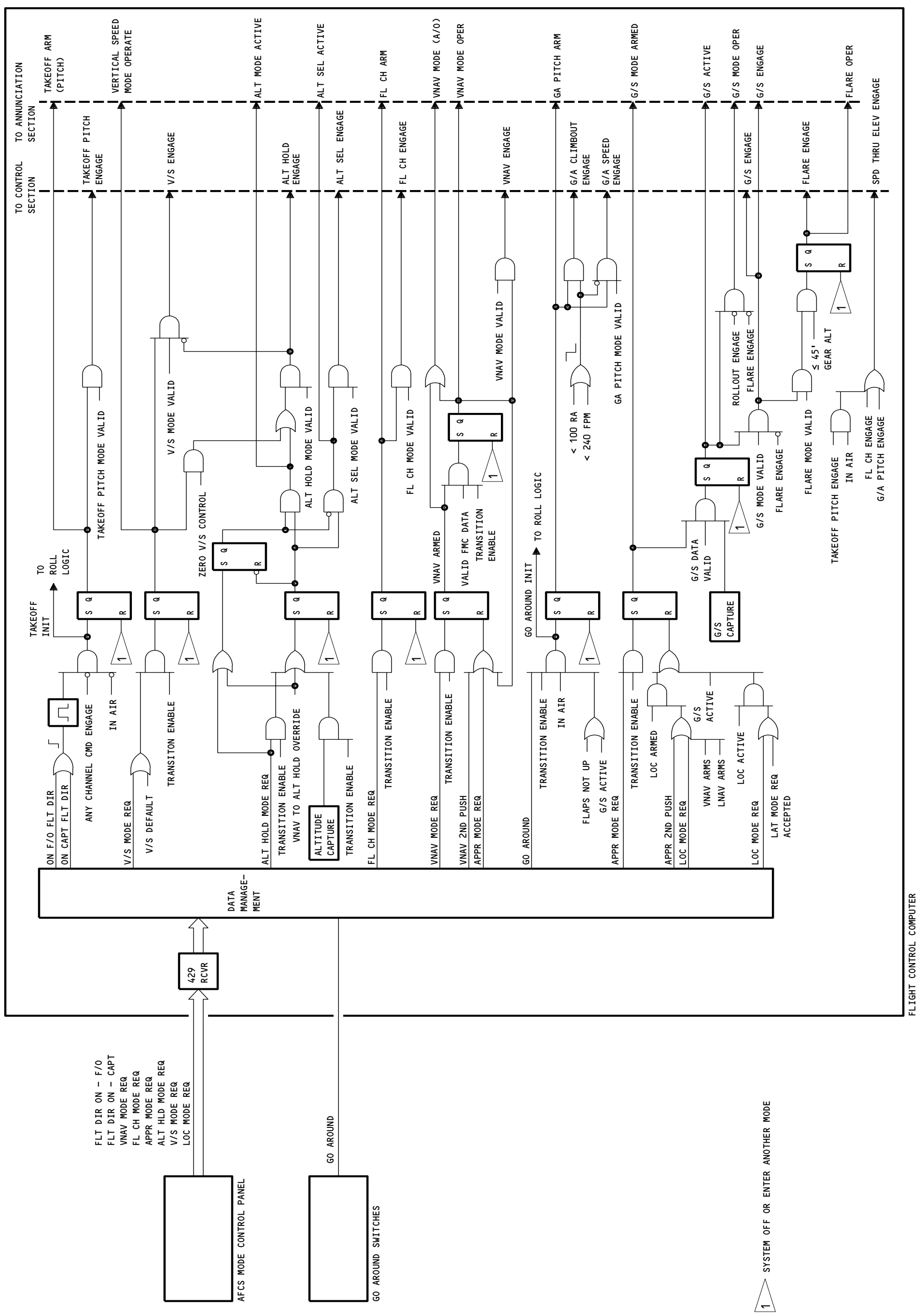
(b) Before takeoff, the crew must set a climbout speed on the MCP. The flight director will show a pitch-up command of 8 degrees. When the radio altimeter is more than 5 feet or 2 seconds, an INAIR signal is supplied which adjusts the 8 degree command to the set altitude command to keep the takeoff speed.

(c) The EADI flight mode annunciator displays TO when takeoff mode is engaged. It continues to display TO until takeoff mode is disengaged by selecting any other valid mode or by switching both flight directors OFF. The following are valid modes that disengage the takeoff mode:

1) Altitude mode active - select altitude hold or selected altitude captured.

2) Flight level change - select FLCH mode.
3) Vertical navigation - VNAV captured
   a) The VNAV ARM is a submode. When VNAV is pressed, the mode request is accepted, and the dot-bar matrix is illuminated, VNAV is annunciated on the MCP and displayed in the pitch mode armed section of the EADI. When the VNAV path is captured, VNAV mode is entered and VNAV is annunciated in the pitch mode engaged section of the EADIs. Pressing the VNAV switch/light a second time deactivates the VNAV modes.
4) If a valid mode is not selected and CMD is engaged, vertical speed mode automatically engages.

(3) Autopilot/Flight Director System Pitch Mode Logic and Control Signal Flow (Fig. 10)
   (a) Takeoff Mode Control Logic
      1) Takeoff mode logic causes Takeoff Attitude engage (TAKEOFF ATT ENG) to be a logic one when the airplane is not in the air, CMD switches are off, either or both F/D switches are ON, and TAKEOFF SPD ENG is logic zero. When TAKEOFF ATT ENG is logic one, the 8 degree pitch up bar command is enabled.
      2) At a radio altitude of 5 feet a logic zero is sent to AND and OR gates in the IN-AIR logic circuit. The logic zero is put through a 2 second delay. The NAND gate outputs a logic one (IN-AIR) 2 seconds after the airplane passed the 5 foot point. The IN-AIR logic one enables the SPD ENG and disables the 8 degree pitch-up bar command (TAKEOFF ATT ENG). This reduces the 8 degree pitch-up command toward the desired pitch altitude over a 2 second period, to maintain the takeoff airspeed.
   (b) Takeoff Mode Control Signal Flow and Monitoring
      1) The TAKE OFF SPD ENG provides flight director pitch commands using speed schedules for a safe climbout speed. The roll channel provides a versine compensation gain schedule to provide pitch corrections for an airplane bank angle greater than ±1 degree.
         a) The versine signal is modified by the versine compensation gain schedule as a function of flap position and inpact pressure (IAS).



Autopilot/Flight Director System Pitch Modes Logic Schematic
Figure 10 (Sheet 1)

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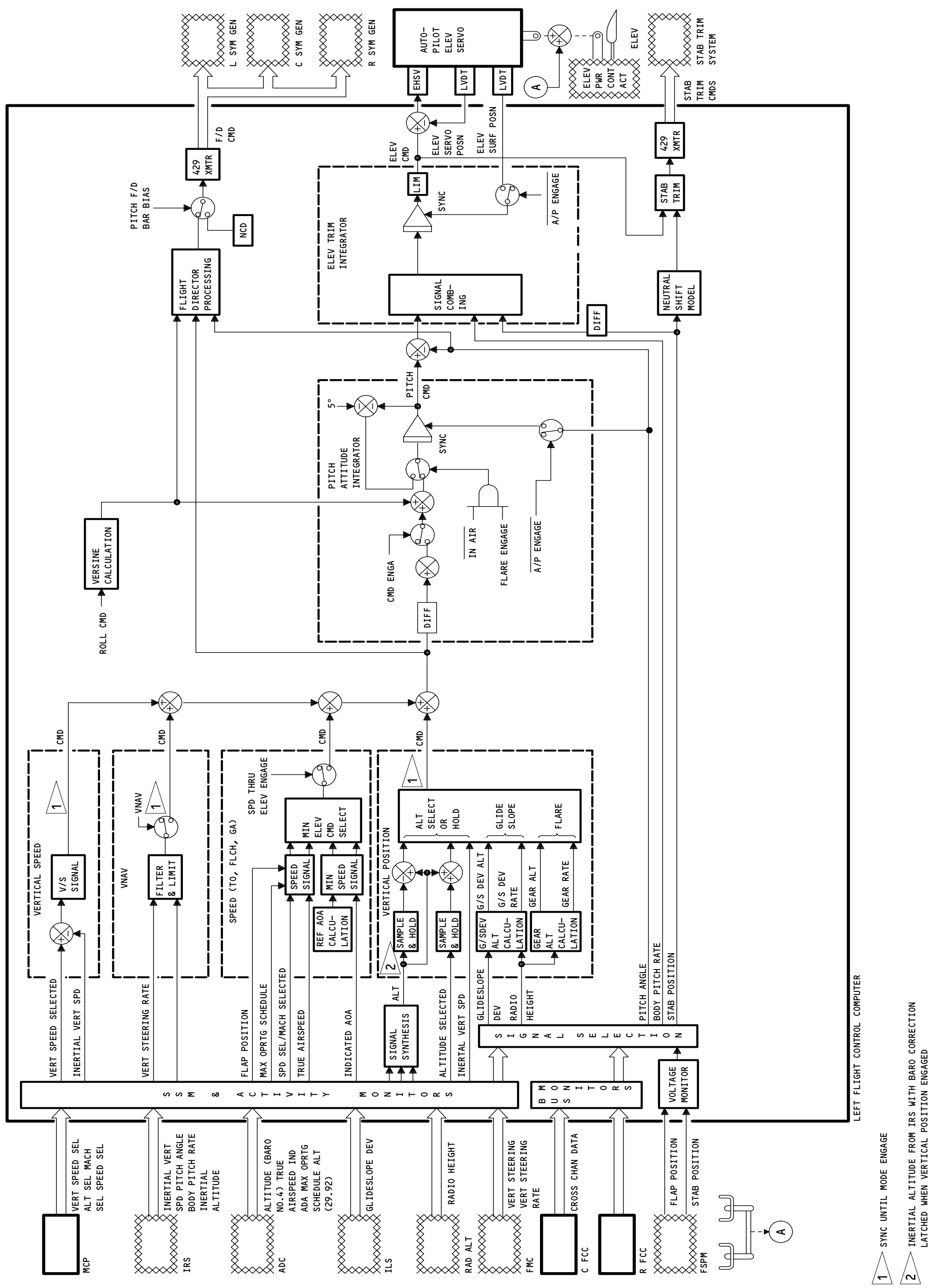

Autopilot/Flight Director System Pitch Control Signal Flow Schematic
Figure 10 (Sheet 2)

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b) Increased flap position is directly proportional and increases the signal. An increase in impact pressure is inversely proportional and decreases the signal.

(c) Speed Through Elevator

1) Speed Through Elevator Command Processing

a) The Speed Through Elevator (STE) function computes elevator commands to capture and hold the selected speed. It is normally used with autothrottle to achieve airspeed and rate of climb control. The alpha floor calculation computes elevator commands to capture and hold minimum speed (130% of stall speed) for angle of attack control. The A/T mode predicts the elevator response necessary to compensate for throttle change and adds this to the other two signals.

b) Command selection normally uses STE signals but automatically selects alpha floor signals when necessary to prevent flying below minimum speed. These signals produce the FLCH proportional command in degrees of pitch attitude.

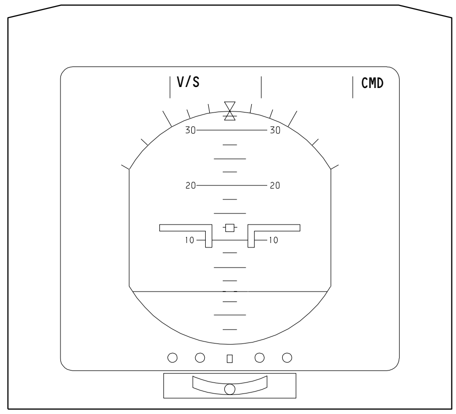
c) When minimum speed is approached, command selection automatically switches the integral and proportional commands to alpha floor commands. Rate bias is generated when alpha floor is selected. This prevents the speed command processor from building up an error when alpha floor is engaged. The total command rate is limited to avoid excessive g-loads.

d) When only the flight director is selected, the turbulence cancel command is combined with STE proportional command to further reduce F/D bar activity during turbulence. The proportional command is automatically switched to alpha floor command when minimum speed is approached. Rate bias for the speed command processor is generated when alpha floor is engaged. With STE engaged, the pitch rate integral command, pitch path command, and the pitch F/D bar command are applied to the pitch integrator.

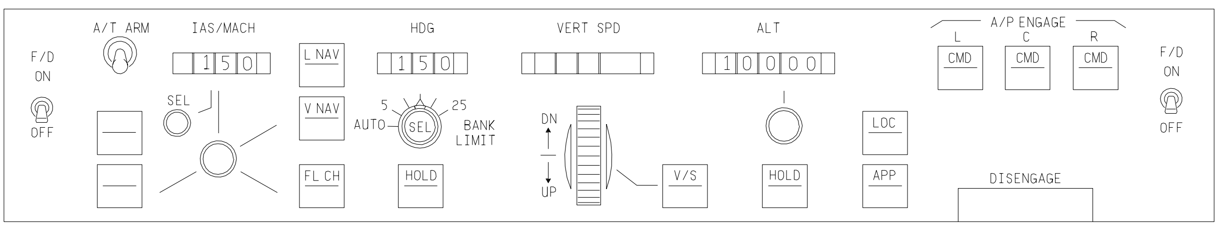
e) Three modes use Speed Through Elevator. The takeoff mode, Flight Level Change (FLCH) mode, and go-around mode use the speed through elevator function. The FLCH mode combines with throttles to reach a new flight altitude. In the go-around mode, throttles are used to gain altitude and maintain speed.

2) Takeoff Mode (Speed Engage)
   a) The takeoff mode is a F/D only mode. The speed through elevator function provides a smooth transition from rotation bar bias to speed control when in the air.
3) Flight Level Change Mode
   a) In FLCH, the pitch channel controls the elevators to capture and maintain the selected IAS/MACH. The throttles are used to control rate of climb or descent. With the A/T armed, throttles control vertical speed based on the test altitude change requested. With the A/T off, the pilots control the throttles as required. Without any change of thrust, capturing a new speed results in vertical speed change.
4) Go-Around Mode
   a) The go-around (G/A) mode is ARMED when the glideslope is captured or flaps are extended. The pitch channel controls speed and the throttles control rate of climb (either automatic or manual throttle). The airplane must achieve 400 ft radio altitude in order to exit the G/A mode.

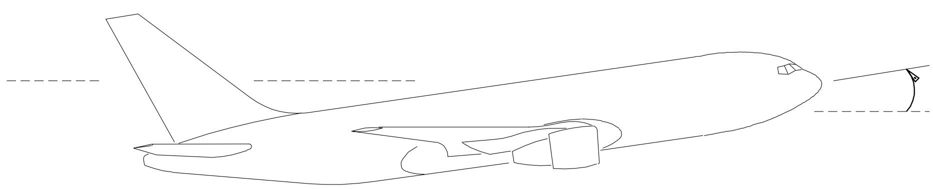
(4) Pitch Channel - Vertical Speed Mode (Fig. 11)
   (a) V/S entry and exit is possible provided transition logic and acceptance criteria are satisfied. Initial entry requires that an A/P or F/D be engaged. Entry from other modes requires that the V/S switch/light be pressed, the request accepted and no incompatable modes be in operation.
   (b) Exit from V/S to control inactive requires switching A/P and F/D to off. Exit to FLCH, ALT HOLD and G/A modes requires that the mode switch/light be pressed and the request accepted. Vertical speed is active during VNAV ARM and G/S ARM conditions. Exit to VNAV capture requires that VNAV arm is established which includes VNAV switch/light pressed and accepted, no ILS modes are established, and the airplane is within the capture zone. Exit to glideslope capture requires that the APP switch/light is pressed and accepted and that the airplane is within the glidelsope capture zone.



EADI
(EXAMPLE)



AFDS MCP



- COMMAND ENGAGE (CMD ENG)
  - AIRPLANE
    - VERTICAL SPEED (V/S) MODE
    - CLIMBOUT AT PRESENT V/S DISPLAYED
    - V/S MAY BE CHANGED WITH V/S KNOB
  - MCP
    - F/D SWITCH ON
    - A/P CMD SWITCH ENGAGED
    - ALT DISPLAY SET AT ATC CLEARED ALTITUDE
    - VERTICAL SPEED DISPLAY SYNC TO V/S AT ENGAGEMENT, THEN CHANGED AS DESIRED
    - V/S SWITCH DOT-BAR MATRIX ILLUMINATED
  - EADI
    - F/D BARS INDICATE AIRPLANE AT COMMANDED PITCH ATTITUDE
    - FMA INDICATES:
      - STATUS-CMD
      - PITCH MODE ENGAGED-V/S

Pitch Channel - Vertical Speed Mode
Figure 11

(c) Two submodes are available: VNAV ARM and G/S ARM. The airplane is controlled with V/S until reaching the capture zone of VNAV or G/S. After capture, V/S control is stopped and VNAV or G/S signals provide pitch commands. VNAV and G/S armed are annunciated on the EADI.

(d) A valid V/S mode switch request is required for acceptance into V/S mode. However, entry into V/S is automatic without pressing a V/S switch/light when an F/D or A/P is engaged from control inactive, and an autothrottle SPD or EPR/N1/THR mode is engaged.

(e) With V/S engaged, the airplane holds the existing V/S if within ±30 degrees pitch angle. If greater than 30 degrees, the airplane attitude is reduced to 30 degrees and the corresponding vertical speed is used as the reference. The EADI flight mode annunciator displays V/S when the V/S pitch mode is engaged.

(f) Vertical Speed Command Processing

1) Vertical speed command processing computes the pitch path (THETA proportional) command, the pitch rate integral command in degrees of pitch attitude per second, and versine in degrees of pitch attitude.

2) The pitch reference signals appropriate to the V/S mode are selected. Pitch rate limits, gain schedules, and filters are applied as necessary. Signals are synchronized during the non-pitch active modes.

(g) Vertical Speed Processing Logic

1) Vertical speed processing logic generates vertical speed engage and vertical path control logic for the vertical speed command processing circuits.

(h) Vertical Speed Signal Processing

1) Vertical speed signal processing provides the following gain scheduling signals:

a) Versine compensation

b) True airspeed inverse

c) Angle of Attack/true airspeed

(i) Roll Channel Signal Processing

1) Roll channel signal processing generates the cosine of the roll attitude command which is used for the versine computation.

(j) Pitch Flight Director Signal Processing

1) Pitch flight director signal processing generates the pitch F/D bar command from the pitch path command input.

(5) Vertical Speed Control Logic

(a) Vertical Speed Mode Inactive

1) The logic zero condition for vertical speed engage is generated with the following three conditions:

a) F/D ON with the CMD switch selected to CMD. This generates a F/D DISENG momentary high pulse.

b) Altitude hold mode selected. This causes the input to vertical speed engage gate to go high.

c) Vertical speed entry conditions not satisfied. This causes the input to vertical speed engage gate to go low.

2) Any of the above conditions generate a vertical speed engage logic zero which allows synchronization of the existing V/S command. The vertical speed engage valid computation is normally set to one as a configuration management interlock.

3) The logic zero condition for vertical path control is generated when:

a) Vertical speed engage is zero (disengaged)

b) Glideslope engage or altitude hold engage and a one-shot pulse for a vertical position mode change is generated.

4) Both of the above conditions generate a vertical path control logic zero. This opens the path for pitch rate compensation which is composed of wind acceleration and AOA rate filters.

(b) Vertical Speed Mode Engaged

1) Vertical speed engage logic one is generated to provide vertical speed command control with the following conditions:

a) The F/D switch ON and a REQUEST and CMD VALID pulse gone low provide a zero state for F/D disengage.

b) Altitude hold is not engaged.

c) Manual or automatic V/S mode entry conditions are satisfied for V/S mode transition generates a high.

d) V/S engage valid equals one.

(c) Gain Schedule Signal Processing

1) The cosine of the PHI CMD gain schedule shown in Fig. 11 is developed from a cosine computation of the roll attitude command.

2) The angle of attack (AOA) and true airspeed (TAS) gain schedule computation is computed from impact pressure (IAS) and TAS which have been modified by flap position.

3) The TAS inverse gain schedule computation is computed from TAS modified by flap position.

4) The versine compensation gain schedule computation is computed from impact pressure and flap position.

(6) Vertical Speed Command Processing

(a) Pitch Attitude Integrator Functions

1) The pitch attitude integrator computes four functions: pitch compensation - Versine, pitch rate integral command, pitch path command, and pitch F/D bar command. The integrator synchronizes pitch computations when the A/P is not in the V/S mode. It also selects the correct pitch reference for the V/S mode and ensures the pitch command produced is in the correct direction. Gain programs are applied as necessary.

(b) Gain Schedules

1) Four gain schedules are used. Gain schedule 1 controls versine as an inverse function of impact pressure (IAS) and as a direct function of flap position. Gain schedule 2 controls flight path acceleration as an inverse function of IAS and TAS. Gain schedule 3 controls pitch rate as a function of the cosine of roll attitude commands. The pitch rate is the summation of wind acceleration and AOA rate components. Gain schedule 4 controls V/S error as a function of inverse TAS. The V/S error is the difference between selected V/S and existing V/S.

(c) Synchronization

1) During synchronization, the V/S command from the vertical speed low pass filter goes to zero when vertical speed engage is a logic zero. The logic zero is generated as a pulse when CMD is engaged and when switching from one mode to another. During synchronization the pitch attitude integrator maintains a constant output for elevator stability.

2) The V/S error is synchronized to zero by V/S command lag filter when vertical speed engage is a logic zero. The output from V/S low pass filter is zero. With vertical path control a logic zero, the pitch rate path compensation signal is removed from the pitch rate integral command signal. With inputs zero, the pitch path integral command is zero. The pitch F/D bar command and pitch path command are zero.

(d) V/S Mode Logic/Switch Signal Status

1) After synchronization, V/S engage and vertical path control are both logic 1. Vertical speed command lag filter and vertical speed low pass filter are activated and S1 is closed. The FLCH and vertical position commands are zero. The VNAV, FLCH and vertical position integral commands are zero.

(e) V/S CMS Signal Processor Outputs

1) With V/S ENG mode, pitch attitude integrator output is proportional to filtered V/S error path command and V/S error integral command and pitch rate path compensation. The V/S error path command is provided as the pitch path command input to the pitch attitude integrator. The V/S error integral command and pitch attitude integrator are summed and provided as the pitch rate integral command input to the pitch attitude integrator.

(f) V/S Mode Reference

1) The V/S reference from V/S command lag filter is developed from the MCP selected vertical speed (altitude rate) display. The display is automatically set to the existing vertical speed at V/S ENG. The display is manually set by rotating the VERT SPD wheel.

2) Prior to V/S engage, V/S mode reference is synchronized to the IRS altitude rate value. The output is zero when V/S engage is zero.
3) After V/S engage, the V/S mode reference signal in V/S command lag filter is time delayed and integrated to dampen any rapid VERT SPD wheel changes. Rate changes are limited to prevent exceeding 0.lg during V/S capture. The output from V/S command lag filter is summed with IRS altitude rate value to provide the V/S error.

(g) V/S Error Signal Processing
1) The V/S error signal is modified as a function of inverse TAS and applied to a V/S low pass filter. The low pass filter processes the gain scheduled V/S error through an integrator with a time constant of 1.0 seconds. This provides the pitch path CMD.

(h) Pitch Path Command
1) The pitch path command provides a signal, proportional to V/S error, to change elevator deflections for short turn (immediate response) due to V/S errors from the selected or present value. It diminishes as the airplane attitude changes and so decreases the V/S error magnitude. The V/S error will not go to zero with pitch path command only, due to changed aerodynamic effects.

(i) Pitch Rate Integral Command
1) The pitch rate integral command is composed of V/S error which is reduced by a factor of 0.3 and pitch rate path compensation, which is composed of short duration wind accelerations and AOA rate components. These summed signals provide the long term integral command for elevator trim to reduce the V/S error to zero (selected V/S = actual V/S).

(j) Versine - Pitch Compensation
1) Versine pitch compensation is computed from the cosine of roll attitude and is modified by versine compensation. It is an inverse function of impact pressure and a direct function of flap position. The pitch compensation versine is applied to the pitch attitude integrator to provide a proportional pitch up signal component during a bank.

(k) Vertical Speed Flight Director Command
1) The pitch path command provides the input for developing the pitch F/D bar command.

(l) Pitch Rate Path Compensation
1) The sum of flight path acceleration and AOA rate filtered make up pitch rate path compensation. The flight path acceleration is modified as an inverse function of impact pressure and TAS. The AOA rate filtered is limited to a 1.0 deg/sec change to prevent rapid rate changes. The resultant signal is modified as a function of the cosine of roll attitude command. The pitch rate path compensation attenuates vertical accelerations and elevator activity during turbulance.

(m) Flight Path Acceleration

1) Inertial flight path acceleration from the IRS and the flight path acceleration wind signal from the wind acceleration filter are summed to make up flight path acceleration. This is then modified as an inverse function of impact pressure and TAS. The modified flight path acceleration provides a desired pitch rate representing actual flight path acceleration and a small component of wind acceleration used in computing an integral command compensation signal to the elevator. This reduces vertical acceleration in the V/S mode.

(n) AOA - Rate Filtered

1) The AOA rate filtered is computed from the computed AOA rate from the IRS. This is integrated and summed with corrected AOA (vane angle from the ADC and IRS) and applied through a low pass filter. The filter limits the signal to ±1.0 deg/sec for damping of elevator movement. The AOA rate filtered represents a component of pitch rate path compensation to the elevator to allow for vertical wind gusts on the AOA vane.

(o) Pitch Attitude Integrator Processing

1) The pitch attitude integrator establishes a pitch attitude reference. It integrates pitch commands from MCP manually selected modes, automatically selected modes, and navigation sources. It also generates pitch attitude error which is processed to provide the elevator servo command. Pitch signal summation to the pitch attitude integrator is provided from the pitch rate integral command, the pitch path command, and the versine command.

2) Vertical Speed Signal Processing

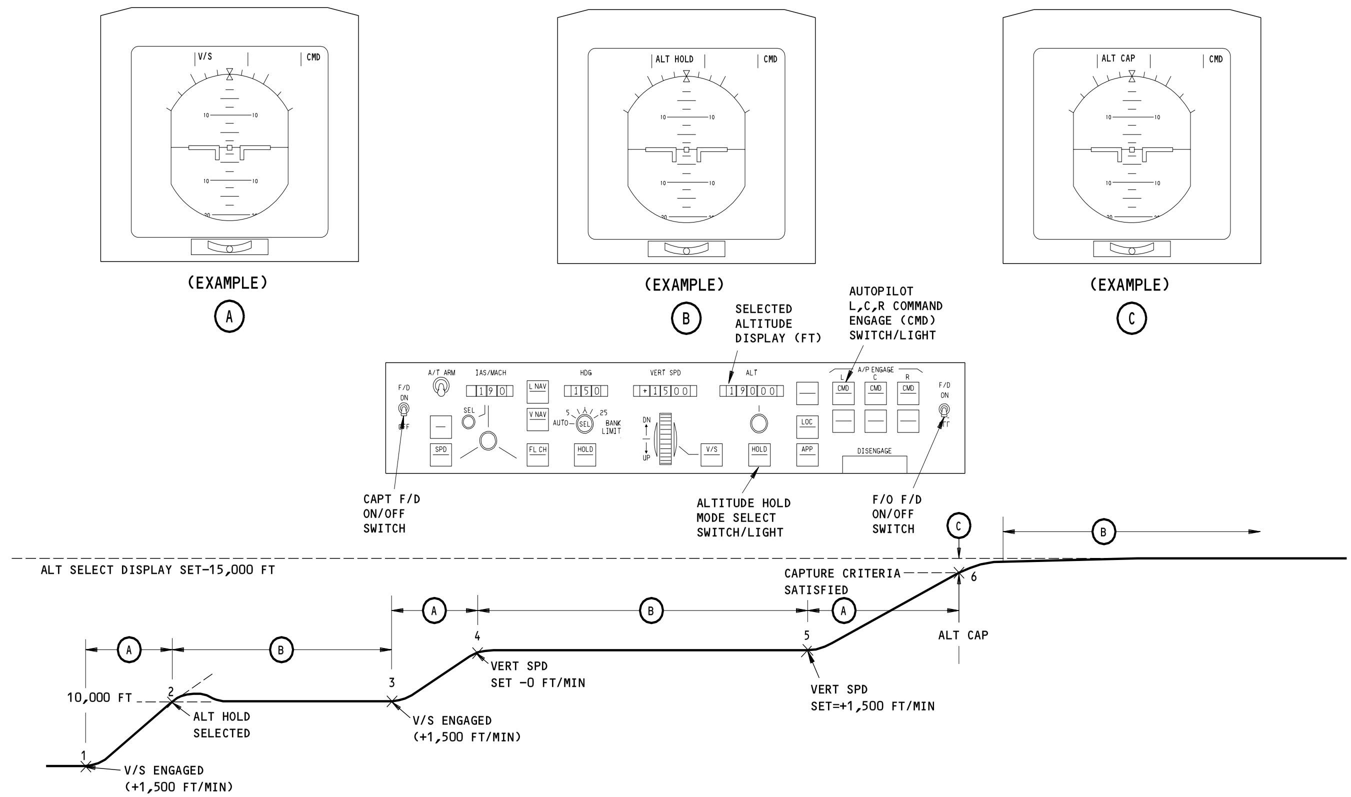
a) The pitch rate integral command is summed with pitch attitude error to provide a long standing error washout at the rate of 0.1 deg/sec/deg.

b) Pitch path (proportional) command provides the immediate response elevator command. This is differentiated and summed with pitch rate integral command. V/S ENG allowing the composite pitch command signal to be summed with any possible signal.

c) The versine command is controlled as a direct function of IAS and as an inverse function of flap position. This is differentiated and summed to form composite pitch rate command.

d) The composite pitch command is integrated and its output is limited to ±30 degrees of pitch angle. This is summed with the reference pitch attitude to provide a pitch attitude error command for processing.

(7) Pitch Channel - Altitude Select/Hold Mode (Fig. 12)



1 • ALTITUDE SELECT DISPLAY MANUALLY SET FOR ATC CLEARANCE OF 15,000 FT
- • V/S SWITCH PUSHED FOR V/S ENG
  - · V/S DOT-BAR ILLUMINATES
- • VERT SPD DISPLAY SET FOR +1,500 FT/MIN
  - · A/P CLIMBS TO POINT 2 IN V/S MODE

2 • ALT HOLD SWITCH PUSHED (AT 10,000 FT)
- · V/S DOT-BAR EXTINGUISHES
- · V/S DISPLAY BLANKS
- · ALT HOLD DOT-BAR ILLUMINATES
- · A/P LEVELS TO BAROMETRIC ALTITUDE AT TIME OF ALT HOLD SELECTED
- · AFDS UNDER ALT HOLD LOGIC CONTROL
- · FMA ON EADI IN PITCH MODE ENGAGED SECTION INDICATES ALT HOLD

3 • V/S SWITCH PUSHED FOR V/S ENG
- · ALT HOLD DOT-BAR EXTINGUISHES
- · V/S DOT-BAR ILLUMINATES
- · VERT SPD DISPLAY INDICATES 0
- • VERT SPD DISPLAY FOR +1500 FT/MIN
  - · A/P CLIMBS TO POINT 4 IN V/S MODE

4 • VERT SPD DISPLAY SET TO 0 AT SELECTED POINT
- · A/P LEVELS & HOLDS BAROMETRIC ALTITUDE
- · UNDER ALT HOLD LOGIC CONTROL
- · V/S MODE ARMED
- · FMA ON EADI IN PITCH MODE ENGAGED SECTION INDICATES V/S

5 • VERT SPD DISPLAY SET FOR +1500 FT/MIN
- · V/S MODE ENGAGED
- · V/S DOT-BAR ILLUMINATES
- · A/P CLIMBS TO POINT 6 IN V/S MODE

6 • CRITERIA SATISFIED FOR ALT CAPTURE
- · V/S DOT-BAR EXTINGUISHES
- · HOLD DOT-BAR ILLUMINATES
- · FMA ON EADI INDICATES ALT CAP UNTIL <10 FT FROM SELECTED ALT, THEN INDICATES ALT HOLD
- · A/P LEVELS OFF TO 15,000 FT
- · AFDS UNDER ALT SELECT LOGIC CONTROL

Pitch Channel - Altitude Select/Hold Mode
Figure 12

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(a) Altitude Hold Mode
1) The altitude hold mode is manually requested by pressing the HOLD switch/light on the MCP. A valid request is indicated by the lighting of the dot-bar matrix on the lower half of the switch cover. The altitude hold mode is automatically selected when in the vertical speed mode and V/S is zero. (The V/S light will remain lighted.)
2) The altitude hold mode is also selected automatically when in the VNAV mode and the FMC has commanded a target altitude in a direction away from the MCP selected altitude. Under these conditions, VNAV disengages and the VNAV switch/light extinguishes. Pitch altitude hold is active in CMD ENG or F/D on.

(b) Altitude Select Mode
1) Altitude select arm and altitude select engage are not manually selectable on the MCP. Altitude select occurs at altitude capture, which is determined by capture criteria, and may be entered from any of the following modes: V/S, FLCH, VNAV, T/O (in F/D only), and GA. Altitude select may not be entered from the G/S mode.

(8) Altitude Select/Hold Control Logic (Fig. 10)
(a) The altitude select/hold control logic processes appropriate FCC data from the data management bus to provide signal control logic. The outputs are altitude select engage, altitude hold engage, flare engage, vertical position approach, and vertical position engaged. These output signals provide the switching control for modes of operation, synchronization, and gain schedule.
(b) Flare Engage Logic
1) Flare engage logic is generated by computation of the appropriate data from the data management bus.
(c) Vertical Position Approach Logic
1) The vertical position approach logic is generated from flare engage or glideslope engage. This logic is required for switching between flare engage/glideslope engage circuitry and ALT HOLD/SELECT circuitry.
(d) Vertical Position Engage Logic
1) Vertical position engage logic is generated by either flare engage, glideslope engage, altitude select engage and altitude hold engage and generates vertical position engage. Whenever a vertical position mode change occurs, vertical position engage is momentarily interrupted. The vertical position engage logic is required for switching from a synchronization mode to active flare engage, glideslope engage, altitude select engage, and altitude hold engage mode in the vertical position command processor.

(e) Altitude Select Engage

1) The altitude select engage logic is generated and is momentarily zero at F/D disengage. The input requires altitude select engage valid and altitude select arm to be logic one. The altitude select arm logic requires that the ALT HOLD switch not be pressed and that the FMC does not command a target altitude in a direction away from the MCP altitude when in the VNAV mode. The ALT SELECT/HOLD mode must be active for altitude select arm to be logic one.

2) The altitude select engage logic is required to establish the altitude select mode in the ALT SEL/HOLD signal development.

(f) Altitude Hold Engage

1) The altitude hold engage logic is generated and is momentarily zero at F/D disengage. The altitude input can be generated three ways (provided altitude hold/engage valid is logic one).

a) If the ALT HOLD switch/light dot bar matrix on the MCP is lighted and ALT SEL/HOLD mode is active.

b) When the FMC (in VNAV mode and not in ALT SEL ENG) commands the airplane to fly to a target altitude in a direction away from the MCP set altitude.

c) In the V/S mode, the V/S display set to zero will also generate altitude hold engage.

(9) Altitude Select/Hold Control Signal Flow and Monitoring (Fig. 10)

(a) Vertical Position Time Constant (Natural Frequency)

1) The vertical position time constant is generated for the three regions in which altitude capture can occur, to provide a safe and smooth transition path to the selected altitude. The three regions of concern are overshoot (region I), acceleration limit (region II) and no acceleration limit (region III). Additional requirements for the time constant calculation are vertical acceleration command limit, delta altitude command and delta altitude rate command. The delta altitude command provides the initial path command and the delta altitude rate command provides the initial path rate command at ALT CAP or when the airplane achieves the proper delta altitude and rate to smoothly achieve the selected altitude.

2) The three regions in which ALT CAP occurs defines the logic which activates time constant calculation for one iteration. The time constant value is limited between 0.01 and 0.50 radian/second for a smooth transition path generation.

(b) Vertical Position Limit
1) The vertical position limit is generated by a logic 1 input from ALT CAP in either overshoot limit region or acceleration limit region. The vertical position limit is required for switching logic in the vertical command processor to input vertical acceleration command limit when the altitude is captured in the overshoot or acceleration limit regions.

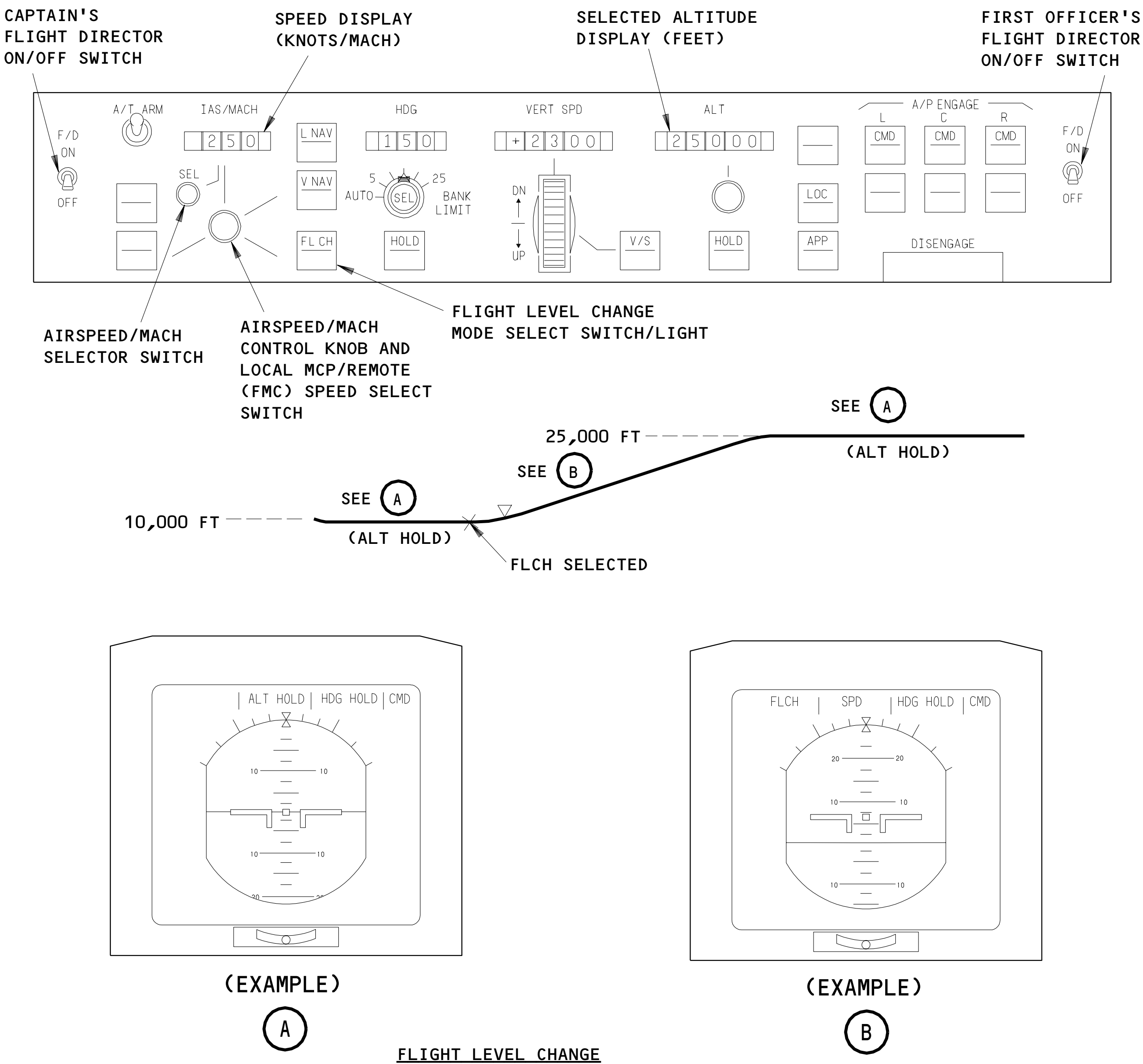
(c) Vertical Acceleration Command Limit
1) The vertical acceleration command limit is a fixed value of vertical acceleration from the IRS at the time of ALT CAP, or it is 3.2 ft/sec (0.1g) prior to ALT CAP. Either of these fixed values will control the airplane until the altitude rate is zero or the absolute value of the rate of approach to selected altitude is reduced to the normal capture limit.
2) The sign of HDDCC is opposite that of delta altitude rate command. This provides an acceleration that will bring the airplane closer to the desired altitude. The altitude capture (ALT CAP) logic is required for vertical acceleration command limit control.

(10) Pitch Channel - Flight Level Change Mode (Fig. 13)

(a) The flight level change mode is entered when the FL CH switch/light is pressed with the autopilot system in any mode except control inactive. Flight level change mode can exit into any autopilot mode except takeoff mode by pressing the appropriate mode button.

(b) If a different autothrottle mode is entered, the AFDS exits FLCH and enters the V/S mode. Two submodes, VNAV ARM and G/S ARM, are possible in FLCH mode. Automatic transition to G/S occurs if G/S is ARMED when G/S is captured. The FMA on the EADI will show G/S in the pitch mode armed annunciation section.

(c) In the FLCH mode, the AFDS commands the elevators to capture and hold the selected airspeed/mode, climb/descent rate controlled by the throttle. This occurs whether in autothrottle or manual operation and is automatically limited to maximum speed limits. The FMA shows SPD as the pitch mode engaged if no limits exist. If a maximum limit exists, the FMA shows FLAP LIM or SPD LIM.



FLIGHT LEVEL CHANGE

- USED TO CLIMB OR DESCEND TO THE NEW FLIGHT LEVEL
- COMBINED THE A/T AND FCC MODES
- THROTTLES ADVANCE/RETARD TO ACHIEVE A CLIMB/DESCENT RATE BASED ON THE ALTITUDE CHANGE REQUESTED
- IF THE CREW OVERIDES THE THROTTLES, A/T GOES TO THROTTLE HOLD
- SPEED THROUGH ELEVATOR CONTROLS THE SPEED TO THE SET IAS/MACH
- AUTOMATIC TRANSITION TO THE VERTICAL POSITION AND THE A/T SPEED WHEN THE NEW ALTITUDE IS APPROACHED

Pitch Channel - Flight Level Change
Figure 13

(11) Pitch Channel - Vertical Navigation (VNAV) Mode (Fig. 14)

(a) The VNAV mode is active in CMD ENG or F/D ON status. It is manually requested by pressing the VNAV switch/light on the MCP. A valid VNAV request is indicated by the illumination of the dot-bar matrix on the lower half of the switch/light. The flight mode annunciator on the EADI indicates VNAV as the pitch mode engaged. The IAS/MACH speed display and the VERT SPD display are both blank when VNAV is engaged. The control source for the VNAV mode is the Flight Management Computer (FMC).

(b) VNAV cannot be entered directly from glideslope arm mode. The glideslope mode is a valid submode of VNAV. The flight mode annunciator FMA on the EADI shows G/S in the pitch mode armed section. A valid VNAV mode switch request is required to engage the VNAV mode (except from control inactive and G/S active).

(c) The VNAV mode is armed by pressing the VNAV switch/light on the MCP. The FMA on the EADI shows VNAV in the pitch mode armed section. The mode is engaged for control of the vertical axis. The current vertical axis mode is disengaged when the FMC outputs valid vertical steering commands. If VNAV steering signals become NCD (no computed data) after engagement, VNAV will disengage but remain ARMED. The VNAV mode is disengaged if the A/P is disconnected (disengaged) and both F/Ds are OFF, or if any other vertical axis mode is engaged.

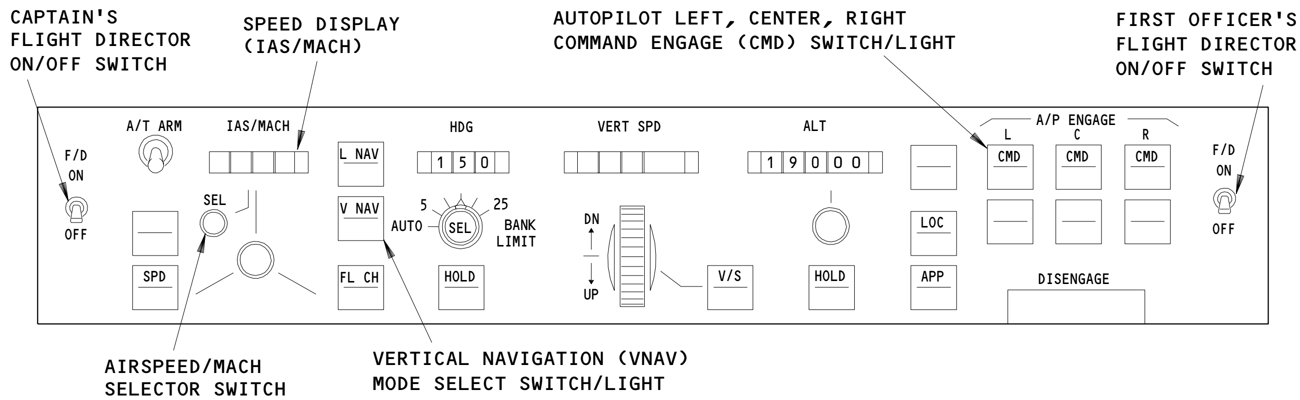
(12) VNAV Control Logic (Fig. 10)

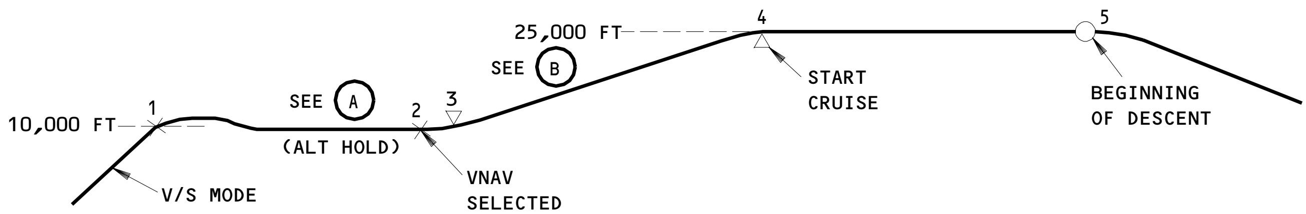
(a) Vertical Navigation Mode Engage

1) The VNAV mode engage logic is generated by three input conditions: F/D not disengaged, VNAV mode active, and VNAV engaged valid. The VNAV mode active means that the VNAV switch/light dot bar matrix is lighted and the FMC is sending valid data.

(b) TAS Inverse Gain Schedule

1) The TAS inverse gain schedule control amplitude is inversely proportional to TAS and is used for gain schedule control.





- ALT SELECT DISPLAY MANUALY SET FOR AN ALTITUDE OF 30,000 FEET (HIGHER THAN FMC ROUTE)

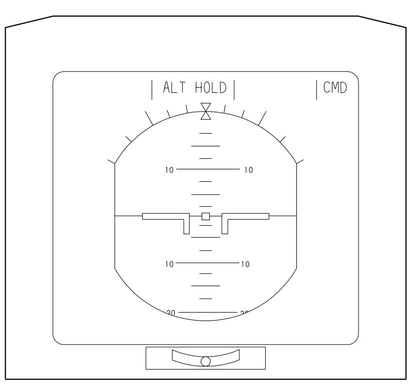
1 • ALT HOLD PUSHED AT 10,000 FEET
- V/S SWITCH DOT-BAR GOES OUT
- V/S DISPLAY IS EMPTY
- HOLD SWITCH DOT-BAR COMES ON
- A/P LEVELS TO BARO-ALT AT TIME OF ALT HOLD
- FMA ON EADI IN PITCH MODE ENGAGES SECTION SHOWS ALT HOLD

2 • VNAV SWITCH PUSHED
- HOLD SWITCH DOT-BAR GOES OUT
- VNAV SWITCH DOT-BAR COMES ON
- IAS/MACH DISPLAY IS EMPTY
- FMA ON ADI SHOWS VNAV
- AFDS UNDER FMC CONTROL

3 • FMC COMMANDS ELEVATORS FOR CLIMB TO 25,000 FEET AT THE GIVEN RATE

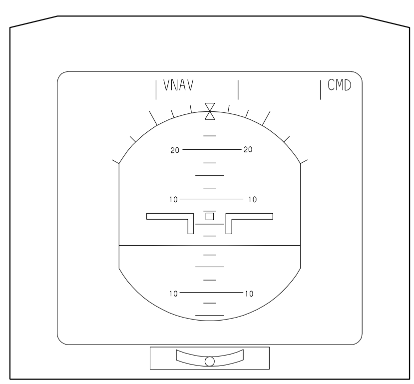
4 • FMC COMMANDS ELEVATORS FOR LEVEL CRUISE

5 • FMC COMMANDS ELEVATORS FOR DESCENT



(EXAMPLE)

A



(EXAMPLE)

B

Pitch Channel - Vertical Navigation (VNAV) Mode
Figure 14

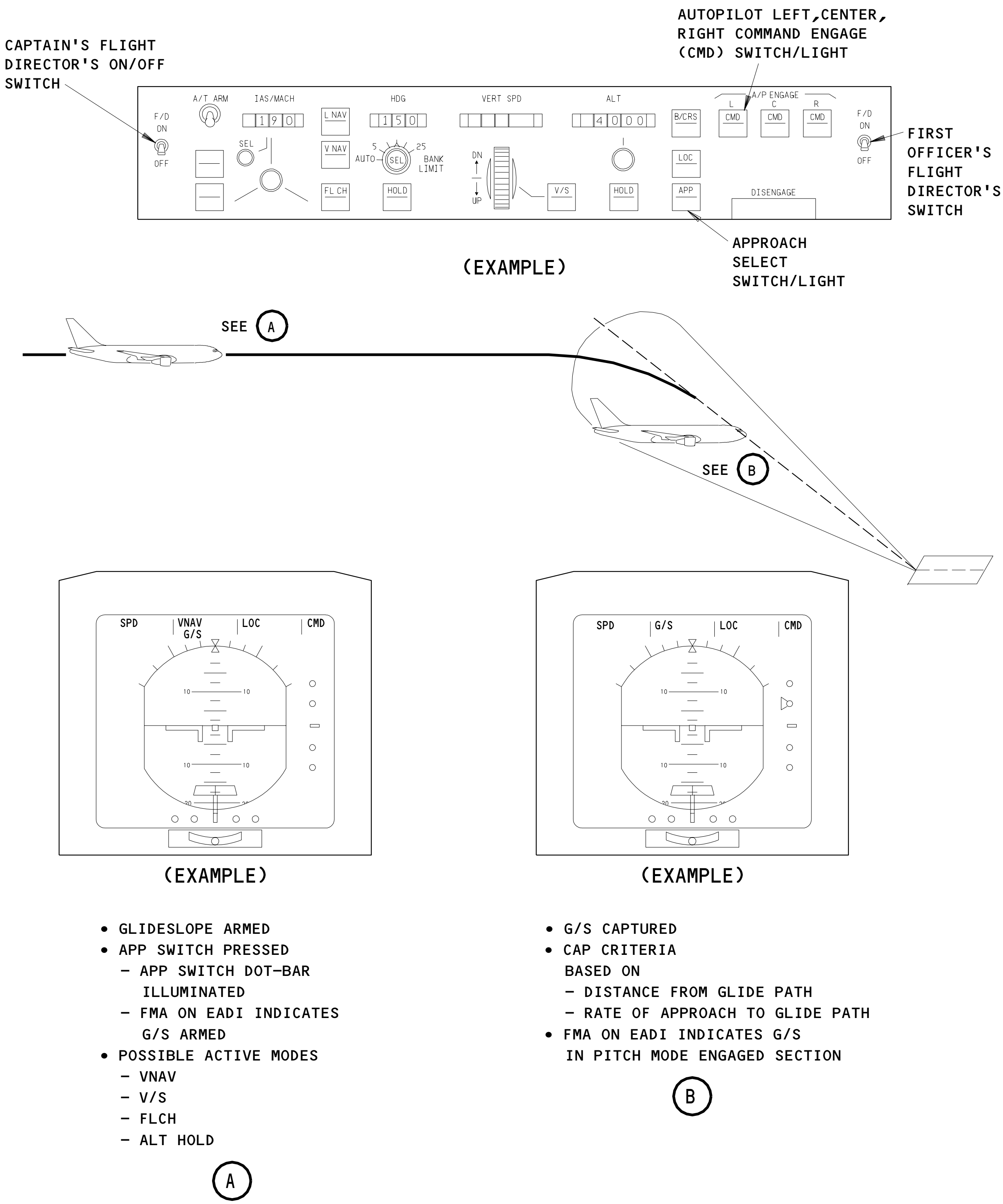
(c) Transition Control Logic From VNAV to ALT HOLD
1) The altitude hold auto engage logic is produced, for one iteration when it is enabled by VNAV mode engage and the following three conditions exist:
a) The airplane is above or below the MCP selected altitude.
b) The FMC is in the process of having the target altitude changed.
c) The FMC is notifying the AFDS that the next target altitude is farther away from the MCP selected altitude.

(d) Transition Control Logic From VNAV to ALT SELECT
1) The altitude select auto engage logic is produced for one sampling when it is enabled by VNAV mode engage and the following four conditions exist:
a) The airplane is above or below the MCP selected altitude.
b) Altitude capture criteria are met.
c) The FMC notifies the AFDS that the next target altitude is such that the airplane will pass through the MCP selected altitude.
d) The FMC target altitude is not equal to the MCP altitude.

(e) VNAV Override
1) If altitude hold auto engage and altitude select auto engage are generated, VNAV override is generated which automatically disengages the VNAV mode and engages ALT HOLD or ALT SEL.

(13) VNAV Control Signal Flow and Monitoring (Fig. 10)
(a) Output Requirements to the Pitch Attitude Integrator
1) Inputs to the VNAV processor are VNAV command and VNAV proportional limit. The VNAV command VNAVS is the VNAV steering command to be processed under VNSLIM limitation to provide TCMD. The VNSLIM is developed from a 4.83 ft/sec^2 (0.15g) constant acceleration command and controlled by inverse TAS.

(14) Pitch Channel - Approach (Glideslope) Mode (Fig. 15)



Pitch Channel - Approach Mode (Glideslope)
Figure 15

(a) The approach mode is armed by pressing the APP switch/light. This also arms the glideslope. Glideslope will engage when the glideslope capture criteria are satisfied. Command engage configuration can be single channel or multi-channel. Glideslope is a valid submode in single or multi-channel engage. Flare is a valid submode in multi-channel engage only.

(b) Manual entry from any permissible mode is possible when the APP switch/light dot bar matrix is lighted. If the switch/light is pressed again, prior to LOC or G/S capture, the APP mode is disarmed.

(c) Automatic exit from the APP mode can occur three ways: selecting a different roll mode after LOC capture but prior to glideslope capture; selecting a different pitch mode after glideslope capture but before LOC capture; and when go-around is selected. There is a one-sampling time interruption of the APP mode if only the F/D is engaged and one channel is initially engaged in CMD.

(d) Two submodes are permissible in APP ARM: glideslope ARM and glideslope engaged. The flight mode annunciator on the EADI shows G/S (glideslope) as the pitch mode armed and pitch mode engaged respectively.

(15) Glideslope Control Logic (Fig. 10)

(a) Glideslope Capture

1) Glide slope capture is armed by APP mode ARM. A bias is added to gear deviation altitude to reference the deviation back to the glide slope antenna. When deviation and rate meet the capture criteria and the airplane is approaching the beam, a capture signal is generated.

2) If deviation and rate reach minimum values without capture criteria being met, capture is forced by a second capture signal. The glideslope engage logic is latched until APP ARM is removed, or Flare is engaged, or until the airplane is on the ground.

(b) Vertical Position Logic

1) The vertical position engage logic goes invalid for one iteration after a vertical position mode change. This allows synchronization to new values. Vertical position engage transition logic goes valid for one frame during the next iteration time. This logic is used by acceleration forecast circuits for acceleration limit determination. One time frame later, vertical position engage delayed logic becomes valid and enables the acceleration limit reset.

(16) Glideslope Control Signal Flow and Monitoring (Fig. 10)

(a) Signal Selection

1) The vertical position approach logic selects filtered glideslope deviation altitude and filtered glideslope deviation rate inputs to command processing. Since the deviation altitude is the distance from the main landing gear to the glideslope beam centerline, a bias of 19 feet is added to reference it to the glideslope antenna.

2) The vertical position approach logic selects the gains for glideslope operation. The position and rate errors are combined to form the integral and proportional commands.

(b) Acceleration Forecast

1) At glideslope capture, the capture region is determined by the relationship between deviation altitude and deviation rate. If no acceleration limit is required, the output is a no acceleration limit region computation pulse. If some acceleration limit is required, acceleration limit region computation latch is set. This is reset when exceeding the acceleration limit is no longer forecast. The overshoot limit region computation latch is set if overshoot is forecast. When the airplane is parallel to the glideslope beam centerline, overshoot limit region computation is reset.

(c) Vertical Acceleration Command Limit

1) A natural frequency (time constant) is calculated for the command processor at capture if no limit is imposed or when an imposed limit is removed. It is held, limited and sent to command processing.

2) When a limit is imposed, the acceleration commanded by the command processor will be set at this value. For glideslope it is a fixed value of 2.5 ft/sec/sec.

3) If no acceleration limits are imposed, the natural frequency sets the gains in the second order filter. The commanded deviation and rate decay from the initial values at a time constant set by the natural frequency.

4) If acceleration limits are imposed, vertical position limit logic operates and the commanded rate is reduced at the fixed 2.5 ft/sec/sec acceleration limit. When the limit is removed, the commanded position and rate decay to zero with the calculated time constant.

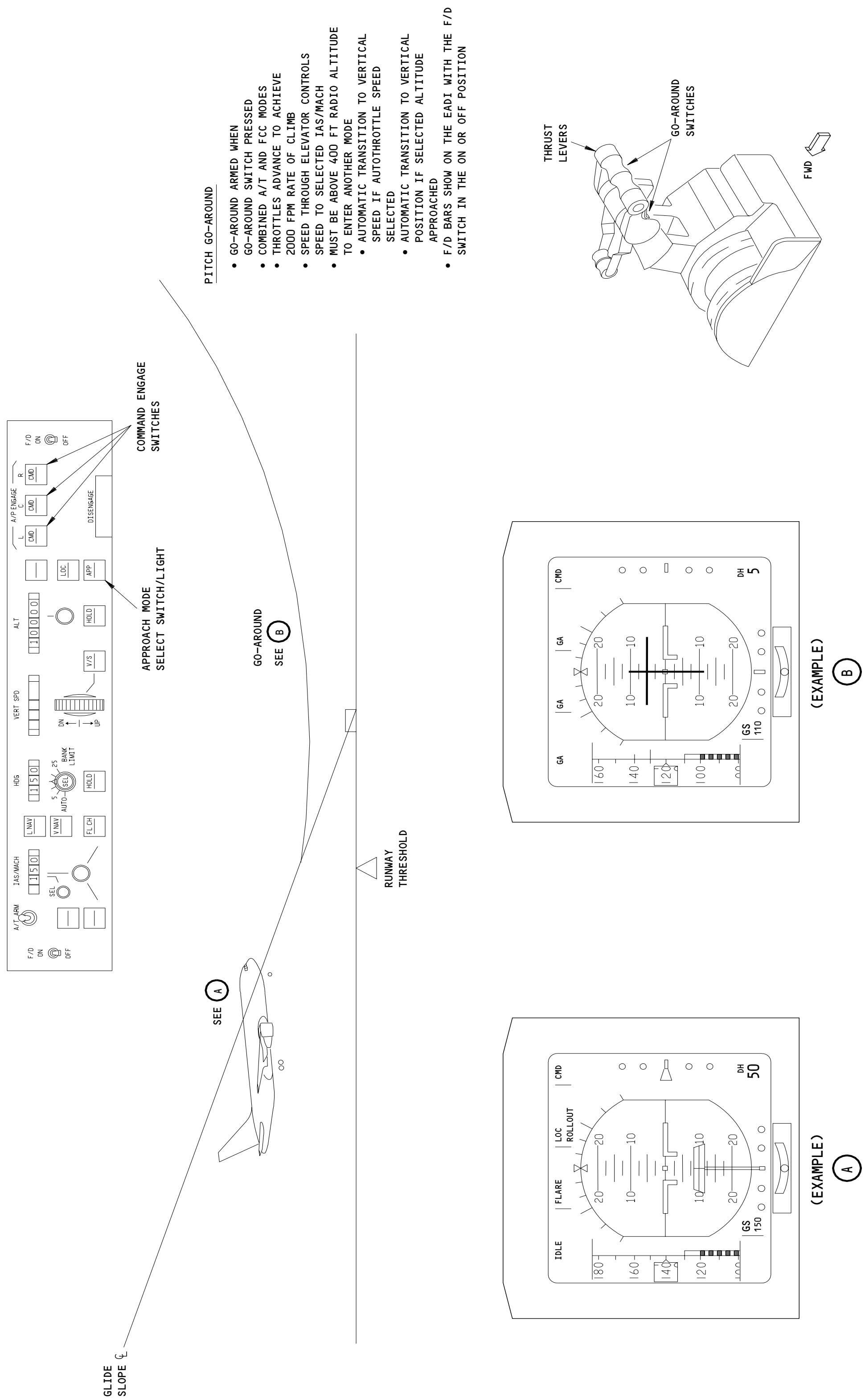
5) Vertical position error and vertical position rate error are the differences between commanded and actual position and rate respectively.

(17) Pitch Channel - Go-Around Mode (Fig. 16)

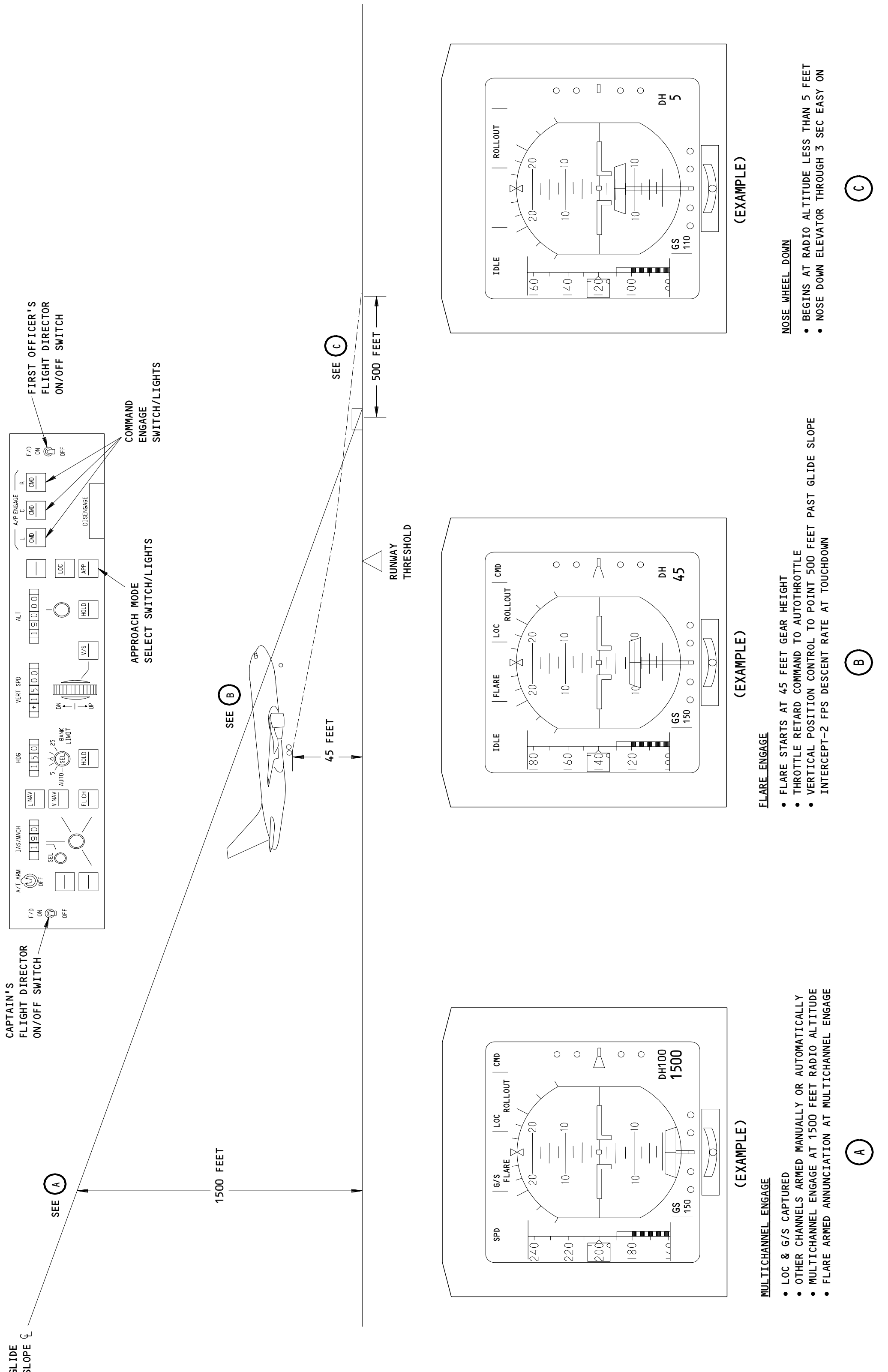
(a) Transition to Go-Around mode is possible from all modes except control inactive and takeoff. Go-around mode can exit into all modes except takeoff and glideslope. The VNAV ARM submode is permissible for go-around. A valid go around switch request is required to enter go-around mode.

(b) On entry into go-around mode the elevators are commanded to capture and hold selected airspeed and the autothrottle (if engaged) commands a 2000 FPM climb rate. A pitch up signal to the elevators is provided to compensate for the increased thrust. The flight mode annunciator on the EADI displays GA as the go-around mode is engaged. The flight director is displayed on the EADI during G/A regardless of F/D switch position.

(18) Pitch Channel - Autoland (Fig. 17)



Pitch Channel - Go-Around Mode
Figure 16



Pitch Channel - Autoland
Figure 17

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(a) Autoland status can only be achieved when two or more A/P channels are ARMED with APP selected. One channel must already be in approach (APP) command to initiate multi-channel autoland. Each additional A/P ENGAGE channel is automatically armed. The Autoland Status Annunciator (ASA) shows channel engagement. The ASA display indications assume FULL TIME NO LAND 3 option is active.

(b) Pitch Multi-Channel Engage

1) Pitch multi-channel engage occurs at 1500 feet radio altitude if G/S and LOC are captured and two or more channels are armed. The flight mode annunciator on the EADI shows FLARE as the pitch ARMED mode.

a) Flare is engaged when the main gear is 45 feet above the ground. A throttle retard command is sent to the autothrottle to control vertical position to a point 500 feet past the glideslope intercept point. The descent rate at touchdown is 2 feet per second.

b) At radio altitude of less than 5 feet, a nose-down command is applied to the elevators to ensure nose wheel steering is effective.

(19) Wind shear

(a) The autoflight wind shear system causes the airplane to climb out of a wind shear when in the TO or GA modes, or when both flight director switches are off and autopilot is not in CMD. The system is started when one of the go-around switches is pushed. If the usual AFDS pitch commands do not cause a sufficient rate of climb (at least 1200 fpm), the AFDS smoothly changes to a 15 degree pitch attitude or a little less than the pitch limit indicator, whichever is less.

(b) The autothrottle sets the go-around thrust unless it is during takeoff when in the THR HOLD mode. When the go-around switch is pushed while in the TO mode, all the performance derates are cancelled and the EPR pointers move to maximum takeoff thrust. When THR HOLD is on, the throttles must be operated manually.

(c) When the wind shear conditions go away, the autoflight system smoothly changes back to the usual takeoff or go-around mode.

(d) If the wind shear system turns on with the flight directors off, the flight director command bars come on automatically and give the applicable commands, and the ADIs show GA. When the wind shear condition is gone, the flight directors can be reset (command bars off) by selecting an AFDS mode, or by setting the flight director switches off, then on.

(20) Autoland Flight Path Control Logic (Fig. 10)

(a) Upon receipt of the APP mode ARMED or OPERATIVE discrete, the APP switch dot-bar matrix is illuminated to indicate request accepted. Each CMD ENGAGE switch initiates a CMDENG request discrete which goes to its corresponding FCC. A valid request generates a REQUEST ACCEPTED discrete which illuminates the CMD switch dot-bar matrix in the up position. Each F/D switch generates an ON/OFF discrete, but there is no visual feedback to the switches.

(b) Command Engage Sequence Detection Logic
1) Each FCC receives CMDENG REQUEST ACCEPTANCE logic from the other two FCCs via the cross-channel bus and records the order in which each become CMDENG. One of the discretes (L FIRST, C FIRST, R FIRST) is supplied to the first-in-command detection logic. This logic determines which FCC will supply FMA signals to the EFIS. It identifies LOCAL FIRST IN COMMAND and FOREIGN channels during multi-channel engage. On detection of a failure in the first-in-command channel, the fail sequence logic selects a different channel to be first-in-command.

(c) Autoland Processing Logic
1) The APP mode request generates the APPROACH ARM logic when the F/D disengage logic is zero. This is applied to the localizer logic as an alternate to the LOC mode arm logic and is also used in autoland logic.

(d) Transition to Rollout Arm
1) When Autoland status is ROLLOUT ARM and the left FCC is LOCAL FIRST IN CMD, and the yaw channel remains in sync.
2) Each of the remaining FCCs (R and C) detects that the LFCC is CMDENG and each must receive individual CMDENG requests. The LOCAL FIRST IN CMD logic for the C and R FCCs is zero. The MULTIARM 1/2 logic is generated in the CFCC and RFCC. Both C and R multichannel engage (MCHENG) logic outputs are zero until MCHENG occurs. The R and C FCCs remain ARMED for CMDENG and are also ROLLOUT ARMED.
3) Multi-channel engage requires G/S and LOC capture and an indication of less than 1500 feet from a valid radio altimeter. Engage logic in the LFCC generates MCHENG. In the C and R FCCs, engage logic generates MCHENG and C or R CMDENG logic. At MCHENG, the LFCC causes the yaw channel to be in CMD. The CFCC and RFCC cause roll, pitch and yaw to be in CMD. All three FCCs are now in ROLLOUT ENGAGE.

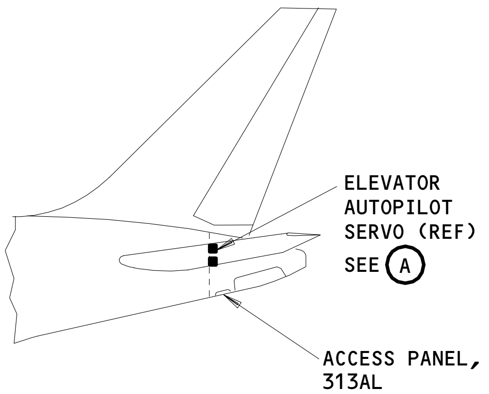
(e) Rollout Engage Default to OFF
1) Rollout engage defaults to OFF occurs when an engage removal fault is detected from the engage interlocks (Ref 22-11-00). If the LFCC defaults to OFF, for example, L CMDENG is logic zero to all FCCs. Sequence detection logic then selects a new LOCAL FIRST IN CMD from a prescribed sequence. This channel controls the ASA display.

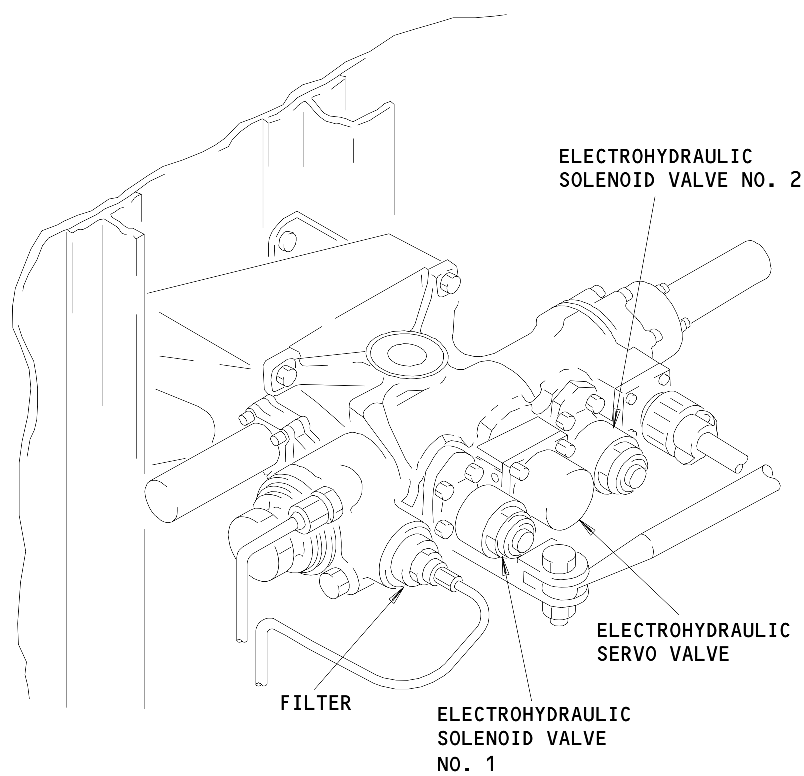
(21) Autoland Flight Path Control Signal Flow and Monitoring (Fig. 10)
(a) Autoland pitch signals provide control during the following positions of the flight profile:
1) Glideslope (less than 1500 feet)
2) Flare (from gear altitude at 45 feet to touchdown)
3) Nose wheel let down
4) Elevator Equalization
(b) At multi-channel engage, elevator equalization logic becomes valid. The elevator commands are adjusted to remove any small errors between surface position feedback and servo position feedback.

AUTOPILOT/FLIGHT DIRECTOR PITCH CHANNEL

COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
FILTER - EAS	--	1	313AL, EA ELEVATOR AUTOPILOT SERVO	22-12-02
SERVO -				
C ELEVATOR AUTOPILOT, M272	--	1	313AL, SECT 48	22-12-01
L ELEVATOR AUTOPILOT, M271	--	1	313AL, SECT 48	22-12-01
R ELEVATOR AUTOPILOT, M273	--	1	313AL, SECT 48	22-12-01
VALVE -				
ELECTROHYDRAULIC SERVO	--	1	313AL, EA ELEVATOR AUTOPILOT SERVO	22-12-02
ELECTROHYDRAULIC SOLENOID	--	2	313AL, EA ELEVATOR AUTOPILOT SERVO	22-12-02

Autopilot/Flight Director Pitch Channel - Component Index
Figure 101





LEFT,CENTER OR RIGHT ELEVATOR AUTOPILOT SERVO,
M271,M272 OR M273

A

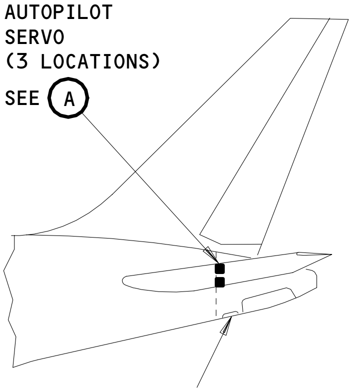
Autopilot/Flight Director Pitch Channel - Component Location
Figure 102

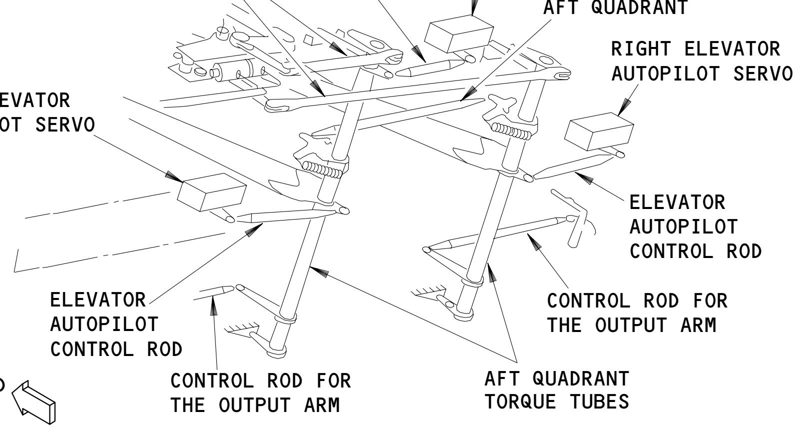
ELEVATOR AUTOPILOT SERVO - REMOVAL/INSTALLATION

1. General (Fig. 401)
   A. The three elevator autopilot servos are on the bulkhead aft of the horizontal stabilizer. You can access the servos and control rods through access panel 313AL (AMM 06-41-00).
   B. The 767-300 Elevator Autopilot Servo can be used on the 767-200 airplanes, but the 767-200 Elevator Autopilot Servo cannot be used on the 767-300 airplanes. An angle bracket on the 767-300 Elevator Autopilot Servo mounting bracket (Fig. 401) prevents the installation of 767-200 servos.
   C. Two adjustment procedures are given for the Autopilot Pitch Control Servo:
      (1) Adjustment with the Maintenance Control Display Panel (MCDP)
      (2) Adjustment with the Phase Synchronous Voltmeter, Model 101A

   TASK 22-12-01-024-001
2. Remove the Elevator Autopilot Servo
   A. Equipment
      (1) Rig Pins from Set A20004-XX (AMM 20-10-24/201)
         (a) E3 - P/N A20004-17
         (b) E4 - P/N A20004-17
      (2) Service Platform, Control Bay Access Door - A51001-19
   B. Consumable Materials
      (1) A00247 Sealant, BMS 5-95
   C. References
      (1) AMM 06-42-00/201, Empennage (Major Zones 300) Access Doors and Panels
      (2) AMM 24-22-00/201, Electrical Power - Control
      (3) AMM 29-11-00/201, Main (Left, Center, and Right) Hydraulic Systems
   D. Access
      (1) Location Zone
          313 Stabilizer Torsion Box Compartment (Left)

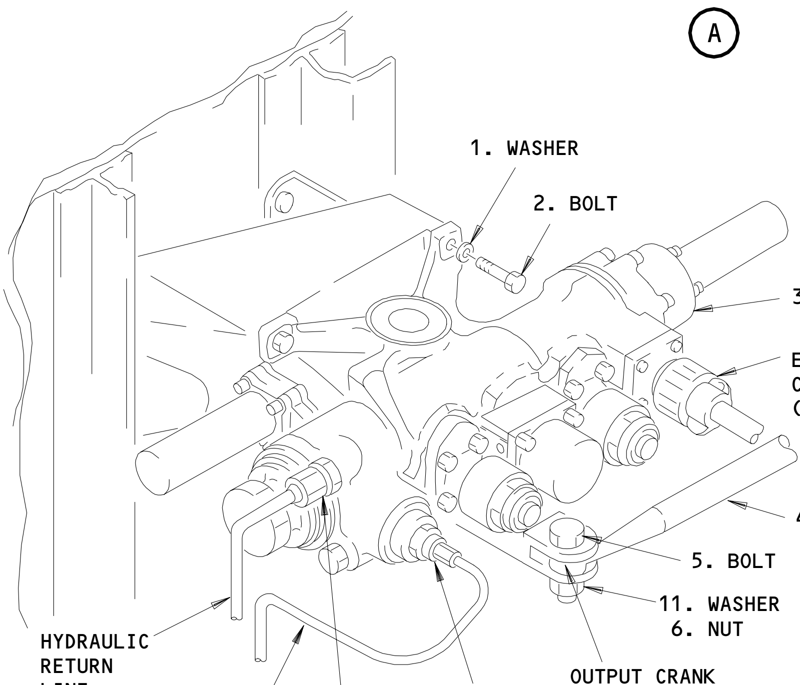
      (2) Access Panel
          313AL Controls Bay Access





ELEVATOR AUTOPILOT SERVO LOCATIONS
AND LINKAGES

A

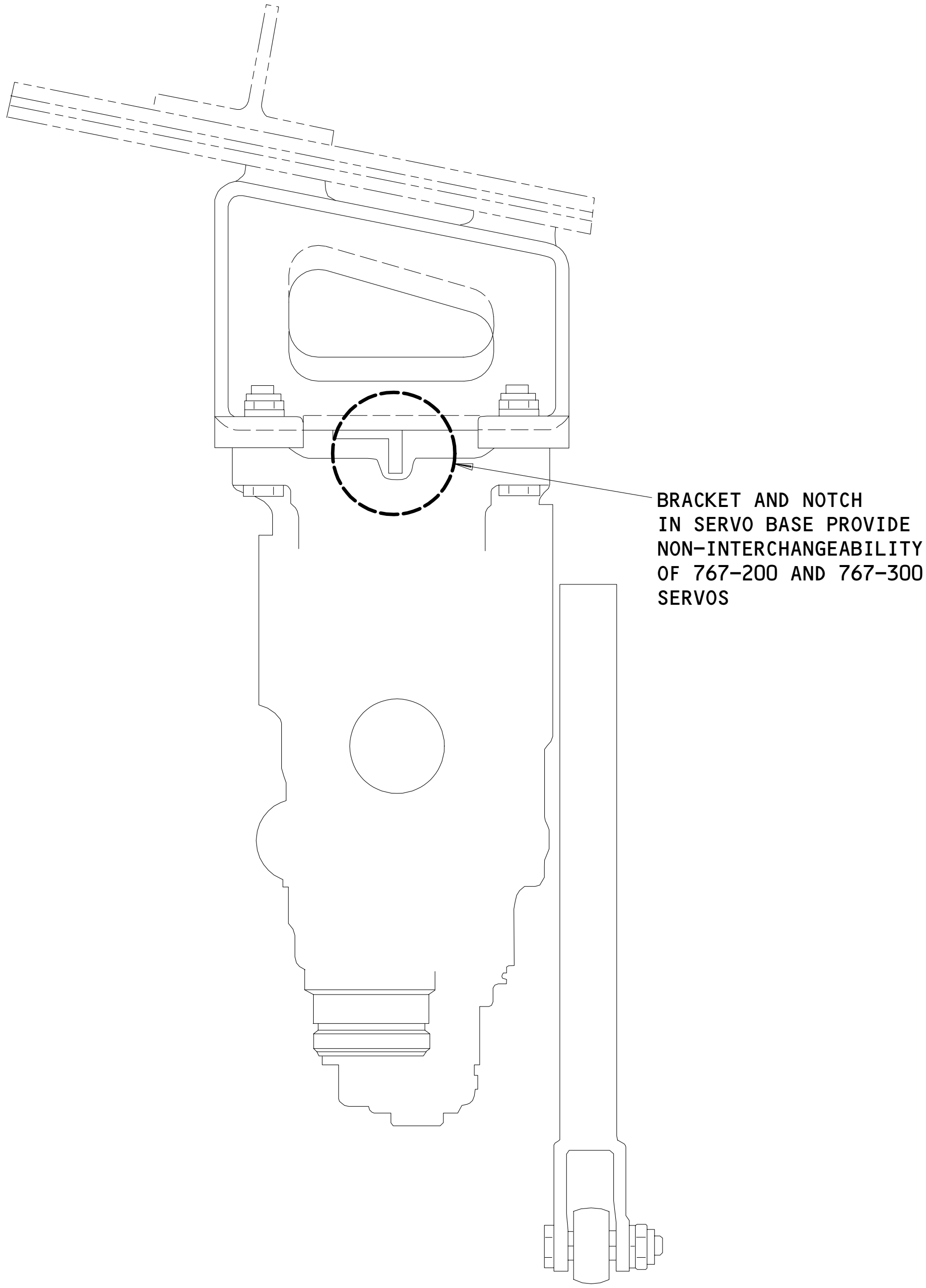


ELEVATOR AUTOPILOT SERVO (EXAMPLE)

B

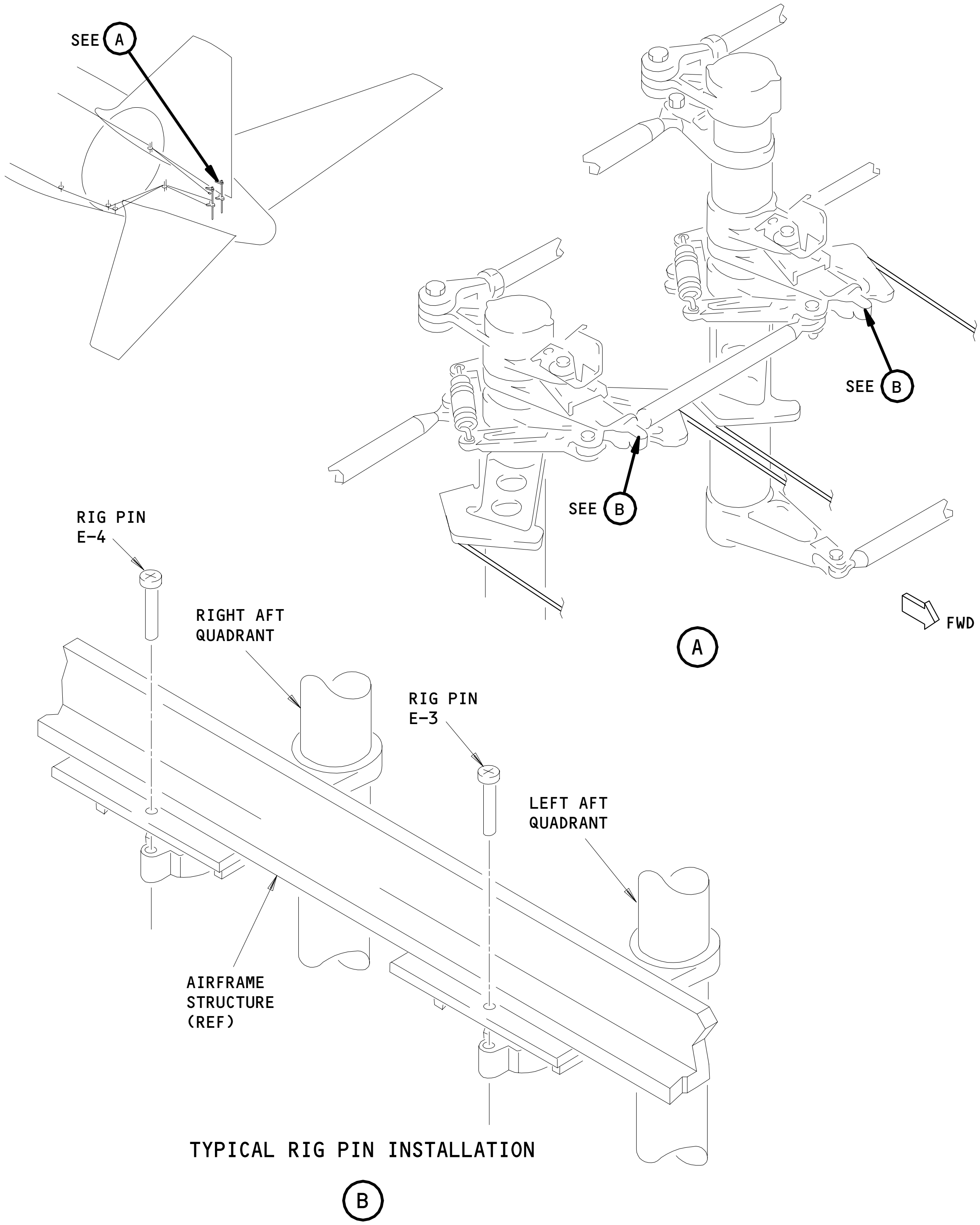
Elevator Autopilot Servo Installation
Figure 401 (Sheet 1)

693006



767-300 SERVO INSTALLATION
C

Elevator Autopilot Servo Installation
Figure 401 (Sheet 2)



Elevator Autopilot Servo Installation
Figure 402

E. Prepare for Removal

S 864-004

(1) Supply electrical power (AMM 24-22-00/201).

S 864-005

(2) Remove the hydraulic pressure from the applicable hydraulic system (AMM 29-11-00/201).
   (a) Left EAS - left hydraulic system
   (b) Center EAS - center hydraulic system
   (c) Right EAS - right hydraulic system

S 864-006

(3) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches, on the right side panel, P61, in the OFF position and attach DO-NOT-OPERATE tags.
   (a) Make sure that the FLT CONT SHUTOFF TAIL VALVE indicator lights (left, center, and right) are amber.

S 864-007

(4) Put the two STAB TRIM switches on the control stand panel, P10, in the CUTOUT position and attach DO-NOT-OPERATE tags.

S 864-008

(5) Set the two F/D switches on the MCP in the OFF position.

S 864-009

(6) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
   (a) 11C12, STAB TRIM SHUTOFF L
   (b) 11C13, STAB TRIM SHUTOFF CENTER
   (c) 11H17, FLT CONT SHUTOFF TAIL L
   (d) 11H18, FLT CONT SHUTOFF TAIL CENTER
   (e) 11H27, FLT CONT SHUTOFF TAIL R
   (f) For the Left Elevator Autopilot Servo
      1) 11E17, FLT CONT CMPTR PWR L
      2) 11E18, FLT CONT CMPTR SERVO L
   (g) For the Center Elevator Autopilot Servo
      1) 11E20, FLT CONT CMPTR PWR C
      2) 11E21, FLT CONT CMPTR SERVO C
   (h) For the Right Elevator Autopilot Servo
      1) 11E35, FLT CONT CMPTR PWR R
      2) 11E36, FLT CONT CMPTR SERVO R

S 864-057

(7) Attach DO-NOT-OPERATE tags on the two control columns.

S 864-010

(8) Open the controls bay access door, 313AL (AMM 06-42-00/201).

S 494-053

WARNING: STAY OFF THE SERVICE ACCESS DOOR, 312AR, AND THE ACCESS DOOR, 313AL, TO THE CONTROLS BAY. YOUR WEIGHT CAN RELEASE THE SPRING-LOADED LATCHES ON THE DOOR. IF YOU FALL THROUGH THE DOOR, INJURIES CAN OCCUR.

(9) Install the service platform over the controls bay access door, 313AL.

S 494-012

(10) Install rig pins E3 and E4 (Fig. 402).

F. Remove the Elevator Autopilot Servo

S 034-014

(1) Disconnect the electrical connector from the servo.

S 034-015

(2) Remove the bolt (5), the nut (6), and the washer (11) to disconnect the control rod (4) from the servo output crank.

S 034-056

WARNING: MAKE SURE YOU REMOVE THE HYDRAULIC PRESSURE FROM THE APPLICABLE LEFT (CENTER, RIGHT) HYDRAULIC SYSTEM BEFORE YOU DISCONNECT THE HYDRAULIC LINES. FAILURE TO REMOVE THE HYDRAULIC PRESSURE CAN CAUSE INJURY TO PERSONS.

(3) Disconnect the hydraulic lines.

S 434-017

(4) Cap the hydraulic lines and the servo hydraulic ports.

S 024-018

(5) Hold the servo (3), and remove the servo mounting bolts (2), the washers (1), and the servo (3).

TASK 22-12-01-424-019

3. Install the Elevator Autopilot Servo

A. Equipment

(1) Rig Pins from Set A20004-XX (AMM 20-10-24/201)

(a) E3 - P/N A20004-17

(b) E4 - P/N A20004-17

(2) Bonding Meter (AMM 20-10-21)

(3) Service Platform, Control Bay Access Door - A51001-19

(4) Voltmeter, Phase Synchronous Model 101A - 101-AC6 or 101-AC7 kit
Electronic Aviation Systems
PO Box 1388 Raccoon Point Road
Eastsound, WA 98245
(Necessary when you do not use the MCDP for adjustment)
(a) Voltmeter, Phase Synchronous Model 101A
(b) Adapter Cable Assembly 1014002-5

B. Consumable Materials
(1) A00247 Sealant, BMS 5-95

C. Parts

AMM		NOMENCLATURE	AIPC		
FIG	ITEM		SUBJECT	FIG	ITEM
401	1	Washer, Flat	22-12-01	01	75
	2	Bolt, Hex Head left and right			65
		center			70
	3	Autopilot Servo Assy, Elevator			105
	4	Control Rod left and right			20
		center			25
	5	Bolt, Hex Head			5
	6	Nut, Self-Locking			15
	7	Union (Return Line)			90
	8	Packing (O-Ring)			100
	9	Union (Pressure Line)			225
	10	Packing (O-Ring)			220
	11	Washer, Flat			10

D. References
(1) 06-42-00/201, Empennage (Major Zones 300) Access Doors and Panels
(2) 22-00-02/201, Autoflight BITE
(3) 24-22-00/201, Electrical Power - Control
(4) 29-11-00/201, Main (Left, Center, and Right) Hydraulic Systems
(5) 34-11-00/201, Pitot-Static System

E. Access
(1) Location Zone
313 Stabilizer Torsion Box Compartment (Left)

(2) Access Panel
313AL Controls Bay Access

F. Install the Elevator Autopilot Servo

S 754-020

CAUTION: DO NOT INSTALL 767-200 ELEVATOR AUTOPILOT SERVOS ON THE 767-300 AIRPLANES. THE 767-300 AIRPLANES MOUNTING BRACKETS HAVE AN ANGLE BRACKET ATTACHED TO PREVENT INSTALLATION OF 767-200 SERVOS. INSTALLATION OF A 767-200 ELEVATOR AUTOPILOT SERVO ON A 767-300 CAN DAMAGE THE SERVO AND THE MOUNTING BRACKET.

(1) Make sure the correct servo is to be installed.

S 144-021

(2) Fully clean the external areas of the valve and around the surfaces of the valve where it attaches to the servo. Clean with a rotary bonding brush or a rotary abrasive disk (AMM 20-10-22/701).

S 034-022

(3) Remove the caps from the hydraulic lines and the servo hydraulic ports.

S 434-023

(4) Connect the hydraulic lines to servo and tighten with your fingers.

S 424-024

(5) Install the servo (3) with the mounting bolts (2) and the washers (1).

S 434-025

(6) Tighten the hydraulic lines.

S 434-026

(7) Connect the control rod (4) to the servo output crank with the bolt (5), the washer (11), and the nut (6).

S 824-027

(8) Use the bonding meter to make sure that the maximum resistance between the servo body and the airframe is 3.0 milliohms (0.0030 ohms).

S 394-028

(9) Apply a fillet seal around the servo assembly where the servo touches the mounting surface. Use BMS 5-95 sealant.

S 394-021

(10) Apply a fillet seal around the bolt (2) heads. Use BMS 5-95 sealant.

S 434-029

(11) Connect the electrical connector to the servo.

S 094-030
(12) Remove rig pins E3 and E4.

S 824-031
(13) If replacement of the control rod (4) is not necessary, adjust the servo with the Maintenance Control Display Panel (MCDP) or the Model 101A Phase Synchronous Voltmeter.

G. Replacement of the Control Rod

S 284-032
(1) Make sure that rig pins E3 and E4 are inserted as shown in Fig. 402.

S 034-033
(2) Remove the bolt (5), the washer (11) and the nut (6) to disconnect the rod assembly (4) from the servo.

S 034-034
(3) Remove the bolt, the washer and the nut to disconnect the rod assembly (4) from aft quadrant.

S 284-035
(4) For the left or the right servo control rod (4), make sure that the replacement rod assembly has a nominal length of 23 11/32 inches from bolt center to bolt center.

<u>NOTE</u>: The nominal length is used to reduce the amount of adjustment necessary to adjust the servo to null.

S 284-036
(5) For the center servo control rod (4), make sure that the replacement control rod assembly has a nominal length of 18 13/16 inches bolt center to bolt center.

<u>NOTE</u>: The nominal length is used to reduce the amount of adjustment necessary to adjust the servo to null.

S 434-037
(6) Connect the rod end (4) to the aft quadrant with the bolt, washer and nut.

S 434-038
(7) Connect the rod end (4) to the servo assembly (3) with the bolt (5).

S 094-039
(8) Remove rig pins E3 and E4.

S 834-022
(9) Adjust the servo with the Maintenance Control Display Panel (MCDP) or the Model 101A Phase Synchronous Voltmeter.

H. Adjust the servo with the Maintenance Control Display Panel (MCDP)

S 864-047
(1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11E17, FLT CONT CMPTR PWR L
(d) 11E18, FLT CONT CMPTR SERVO L
(e) 11E20, FLT CONT CMPTR PWR C
(f) 11E21, FLT CONT CMPTR SERVO C
(g) 11E35, FLT CONT COMPTR PWR R
(h) 11E36, FLT CONT CMPTR SERVO R
(i) 11H17, FLT CONT SHUTOFF TAIL L
(j) 11H18, FLT CONT SHUTOFF TAIL CENTER
(k) 11H27, FLT CONT SHUTOFF TAIL R

S 864-048
(2) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches in the ON position.

S 864-049
(3) Put the two STAB TRIM switches in the NORM position.

S 864-003

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(4) Apply pressure to the L, C, and R hydraulic systems (AMM 29-11-00).

S 984-050
(5) Manually operate the elevator system three or more times.

S 984-051
(6) Set the stabilizer trim to six units of trim.

S 864-052
(7) Pressurize the Elevator Feel Pitot System to 350 knots (AMM 34-11-00/201).

NOTE: Pressurization of the Elevator Feel Pitot System is recommended to accurately center the system for adjustment. However, pressurization of the Elevator Feel Pitot System is not necessary if the rig pins can be installed with finger pressure before pressurization.

S 284-053
(8) Make sure that rig pins E3 and E4 can be installed with finger pressure only. Do not leave rig pins installed.

S 744-054
(9) Start MCDP Ground Test 66-XDCR OUTPUTS (AMM 22-00-02/001).

S 744-050
(10) Push the YES/ADV switch until the top line of the MCDP display shows 66 ELEV SURF DEG.

S 824-055
(11) Adjust the servo control rod (4) until the MCDP bottom line value is 00.0 ±0.2 degrees with the control rod (4) in its installed position on the servo output crank.

S 824-094
(12) Make sure that the difference between channels at the zero position is 0.4 degrees or less.

S 434-056
(13) Connect the control rod assembly (4) with the bolt (5), the washer (11), and the nut (6).

S 434-057
(14) Tighten the control rod (4) jamnuts.

S 864-058
(15) Remove the pressure from the Elevator Feel Pitot System.

S 284-059
(16) Examine the installation for leaks and repair as necessary.

S 714-060
(17) Do the Servo Installation Test

I. Adjustment of the Servo with the Model 101A Phase Synchronous Voltmeter

S 484-061
(1) Connect the Phase Synchronous voltmeter to the power source specified on the voltmeter placard.

S 984-062
(2) Set the ON/OFF (S1) switch in the ON position.

S 484-051
(3) Stop for 10 minutes to let the voltmeter prepare for operation.

S 984-063
(4) Put the CAL/RUN (S2) Switch in the CAL CHK position.
(a) Make sure that the IN-PHASE VOLTS are 9.990 to 10.010.

S 984-064
(5) Put the CAL/RUN (S2) switch in the Run position.

S 034-065
(6) Disconnect the electrical connector from the servo.

S 434-066
(7) Connect the -5 cable assembly to the Autopilot Pitch Control Servo and the Phase Synchronous Voltmeter.

S 864-067
(8) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11H17, FLT CONT SHUTOFF TAIL L
(d) 11H18, FLT CONT SHUTOFF TAIL CENTER
(e) 11H27, FLT CONT SHUTOFF TAIL R

S 864-068
(9) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches in the ON position.

S 864-060
(10) Put the two STAB TRIM switches in the NORM position.

S 864-061

WARNING: KEEP PERSONS AND EQUIPMENT CLEAR OF THE FLIGHT CONTROL SURFACES, THE THRUST REVERSERS, AND THE LANDING GEAR. THESE COMPONENTS CAN MOVE SUDDENLY WHEN YOU SUPPLY HYDRAULIC POWER. THIS CAN CAUSE INJURIES TO PERSONS AND DAMAGE TO EQUIPMENT.

(11) Supply hydraulic power (AMM 29-11-00/201).

S 984-070
(12) Manually operate the elevator system three or more times.

S 984-071
(13) Set the stabilizer trim to two units of trim.

S 864-072
(14) Pressurize the Elevator Feel Pitot System to 350 knots (AMM 34-11-00/201).

NOTE: Pressurization of the Elevator Feel Pitot System is recommended to accurately center the system for adjustment However, pressurization of the Elevator Feel Pitot System is not necessary if rig pins can be installed with finger pressure before pressurization.

S 284-073
(15) Make sure that rig pins E3 and E4 can be installed with finger pressure only. Do not leave rig pins installed.

S 824-074
(16) Adjust the Autopilot Servo Control rod (4) until the Phase Synchronous voltmeter shows between 2.465 and 2.525 IN-PHASE VOLTS with the control rod (4) connected to the servo.

S 434-075
(17) Connect the servo control rod assembly (4) with the bolt (5), the washer (11), and the nut (6).

S 434-076
(18) Tighten the control rod (4) jamnuts.

S 034-077
(19) Disconnect the Phase Synchronous Voltmeter as follows:
(a) Put the ON/OFF switch in the OFF position.
(b) Disconnect the -5 cable assembly from the Autopilot Servo.
(c) Disconnect the voltmeter from the power source.

S 434-078
(20) Connect the airplane electrical connector to the autopilot servo.

S 864-079
(21) Remove the pressure from the Elevator Feel Pitot System (AMM 34-11-00/501).

S 284-080
(22) Examine the installation for leaks and repair as necessary.

S 714-081
(23) Do the Servo Installation Test.

J. Servo Installation Test

S 864-046
(1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11E17, FLT CONT CMPTR PWR L
(d) 11E18, FLT CONT CMPTR SERVO L
(e) 11E20, FLT CONT CMPTR PWR C
(f) 11E21, FLT CONT CMPTR SERVO C
(g) 11E35, FLT CONT COMPTR PWR R
(h) 11E36, FLT CONT CMPTR SERVO R
(i) 11H17, FLT CONT SHUTOFF TAIL L
(j) 11H18, FLT CONT SHUTOFF TAIL CENTER
(k) 11H27, FLT CONT SHUTOFF TAIL R

S 864-045
(2) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches in the ON position.

S 864-044
(3) Put the two both STAB TRIM switches in the NORM position.

S 864-062

WARNING: KEEP PERSONS AND EQUIPMENT CLEAR OF THE FLIGHT CONTROL SURFACES, THE THRUST REVERSERS, AND THE LANDING GEAR. THESE COMPONENTS CAN MOVE SUDDENLY WHEN YOU SUPPLY HYDRAULIC POWER. THIS CAN CAUSE INJURIES TO PERSONS AND DAMAGE TO EQUIPMENT.

(4) Supply hydraulic power (AMM 29-11-00/201).

S 984-043
(5) Manually operate the elevator system three or more times.

S 984-042
(6) Set the stabilizer trim to six units of trim.

S 864-041
(7) Pressurize the Elevator Feel Pitot System to 350 knots (Ref 34-11-00).

NOTE: Pressurization of the Elevator Feel Pitot System is recommended to accurately center the system for adjustment However, pressurization of the Elevator Feel Pitot System is not necessary if the rig pins can be installed with finger pressure before pressurization.

S 284-040
(8) Make sure that rig pins E3 and E4 can be installed with finger pressure only. Do not leave rig pins installed.

S 744-082
(9) Start MCDP Ground Test 66-XDCR OUTPUTS (Ref 22-00-02).
  (a) Push the YES/ADV switch until the top line of the MCDP display shows 66 ELEV SURF DEG.
  (b) Make sure that the MCDP bottom line value is 00.0 ±0.2 degrees for the L, C, and R servos.

S 864-055
(10) Remove the pressure from the elevator feel pitot system (Ref 34-11-00) if it is pressurized.

S 744-086
(11) Do MCDP Ground Test 08-SERVO ELEV (AMM 22-00-02).
  (a) Make sure that no fault messages are shown after test is engaged.

S 744-045
(12) Do the MCDP Ground Test 68-ELEV SURF LIM (AMM 22-00-02) without pitot pressure applied.
  (a) When the MCDP shows "68 A/P ENG LCR TO CMD" engage only the autopilot (A/P) channel that corresponds to the elevator autopilot servo (EAS) that was replaced (engage the right A/P channel if the right EAS was replaced; engage the center A/P channel if the center EAS was replaced; engage the left A/P channel if the left EAS was replaced).
    1) Do these steps:
      a) Make sure the elevator positive limit is more than 7.5 degrees.
      b) Make sure the elevator negative limit is more than -7.5 degrees (-7.6 or subsequent).

K. Put the Airplane Back to Its Usual Condition

S 094-087
(1) Remove the service platform and close the controls bay access door, 313AL.

S 984-088
(2) Set the MCDP to off.

S 864-089
(3) Remove the DO-NOT-OPERATE tags from the control columns.

S 864-090
(4) Remove the pressure from the L, C and R hydraulic systems if it is not necessary (AMM 29-11-00).

S 864-091
(5) Remove electrical power if is not necessary (AMM 24-22-00).

EAS ELECTROHYDRAULIC SERVOVALVE AND SOLENOID VALVES - REMOVAL/INSTALLATION

1. General (Fig. 401)
   A. There are three elevator autopilot servos. They are attached to the bulkhead aft of the horizontal stabilizer. You can access the servos through the controls bay access panel, 313AL (Ref 06-42-00). Each servo has two LRU solenoid valves and one LRU electrohydraulic servovalve. The removal/installation and the adjustment/test procedures are the same for the two type of valves.

TASK 22-12-02-024-045

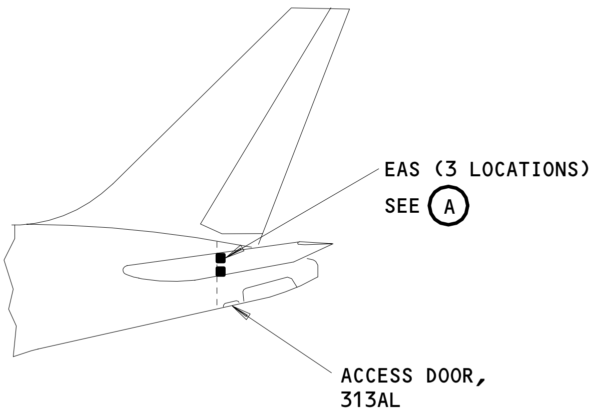
2. Remove the EAS Electrohydraulic Servovalve or Solenoid Valve
   A. Equipment
      (1) Bonding Meter (Ref 20-10-21).
      (2) Service Platform, Control Bay Access Door - A51001-19
   B. Consumable Materials
      (1) D00153 Hydraulic Fluid, Fire Resistant BMS 3-11 (AMM 20-30-04/201)
      (2) D00290 Lubricant - MCS 352B (AMM 20-30-04/201)
   C. References
      (1) AMM 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
      (2) AMM 22-00-02/201, Autoflight BITE
      (3) AMM 24-22-00/201, Electrical Power, Control
      (4) AMM 29-11-00/201, Main Left, Center, and Right Hydraulic Systems
   D. Access
      (1) Location Zone
          313 Stabilizer Torsion box Compartment (Left)

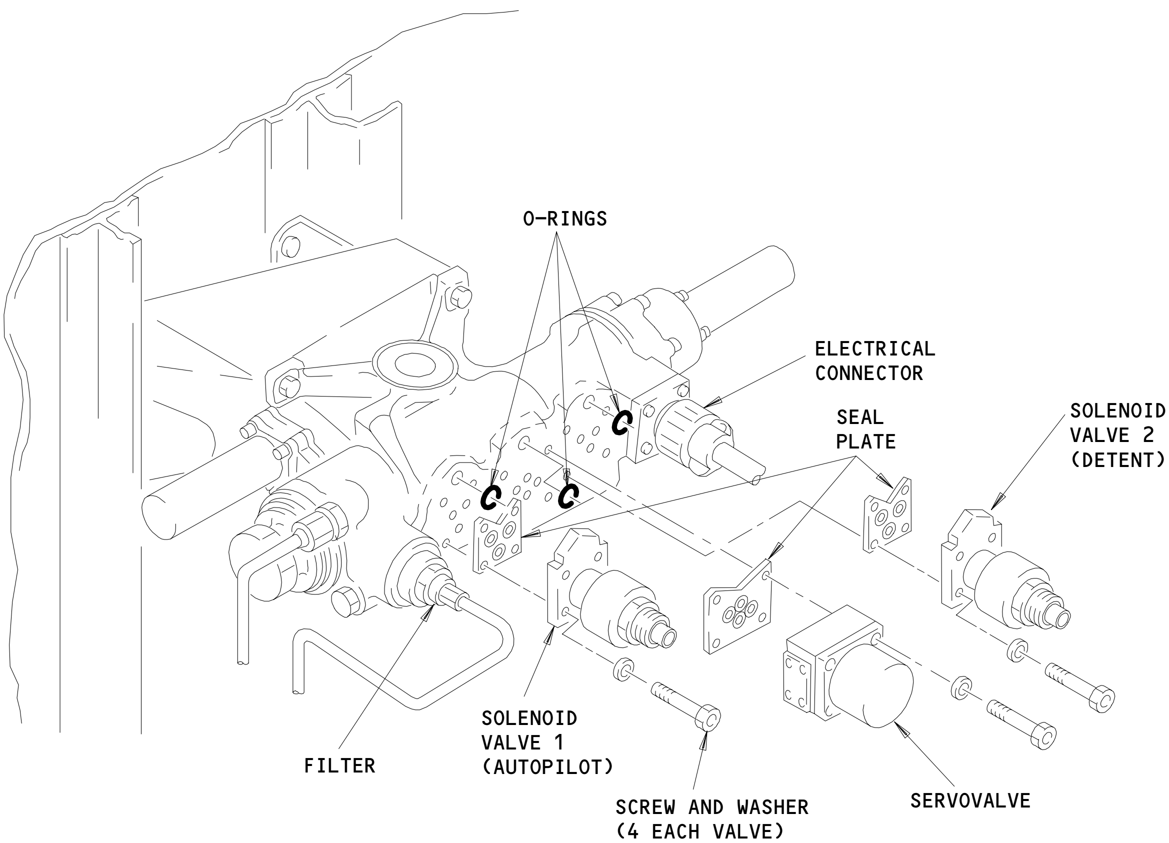
      (2) Access Panel
          313AL Controls Bay Access

   E. Prepare for Removal

      S 864-047
      (1) Supply electrical power (AMM 24-22-00/201).

      S 864-048
      (2) Remove pressure from the applicable hydraulic system (AMM 29-11-00/201).
         (a) The Left hydraulic system powers left elevator autopilot servo.
         (b) The Center hydraulic system powers center elevator autopilot servo.
         (c) The Right hydraulic system powers right elevator autopilot servo.





A

EAS Electrohydraulic Servovalve and Solenoid Valves Installation
Figure 401

S 864-049
(3) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches on the right side panel, P61, to OFF.

S 864-050
(4) Put the two STAB TRIM switches on the control stand panel, P10, in the CUTOUT position.

S 864-051
(5) Put the two F/D switches on the MCP in the OFF position.

S 864-052
(6) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11H17, FLT CONTROL SHUTOFF TAIL L
(d) 11H18, FLT CONTROL SHUTOFF TAIL CENTER

S 864-053
(7) For the left elevator autopilot servo, open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11E17, FLT CONT CMPTR PWR L
(b) 11E18, FLT CONT CMPTR SERVO L

S 864-054
(8) For the center elevator autopilot servo, open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11E20, FLT CONT CMPTR PWR C
(b) 11E21, FLT CONT CMPTR SERVO C

S 864-055
(9) For the right elevator autopilot servo, open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11E35, FLT CONT CMPTR PWR R
(b) 11E36, FLT CONT CMPTR SERVO R

S 014-056
(10) Open the controls bay access door, 313AL (AMM 06-42-00/201).

S 414-057

WARNING: STAY OFF THE SERVICE ACCESS DOOR, 312AR, AND THE ACCESS DOOR, 313AL, TO THE CONTROLS BAY. YOUR WEIGHT CAN RELEASE THE SPRING-LOADED LATCHES ON THE DOOR. IF YOU FALL THROUGH THE DOOR, INJURIES CAN OCCUR.

(11) Install the service platform over the controls bay access door, 313AL.

F. Procedure

S 014-044

CAUTION: DO NOT LET ANY CONTAMINATION GET INTO THE ELEVATOR AUTOPILOT SERVO AND VALVES. CONTAMINATION CAN CAUSE EQUIPMENT DAMAGE AND FAILURE.

(1) Fully clean the external areas of the valve and around the surface of the valve where it attaches to the servo.

S 034-058

(2) Remove the lockwire.

S 034-059

(3) Remove the mounting screws, washers, and seal plate from the elevator autopilot servo assembly.

S 024-060

(4) Remove the valve assembly.

S 024-061

(5) Remove the O-ring.

S 424-062

(6) Put covers on all the openings.

TASK 22-12-02-424-046

3. Install the EAS Electrohydraulic Servovalve or Solenoid Valve

A. Equipment
   (1) Bonding Meter (Ref 20-10-21).
   (2) Service Platform, Control Bay Access Door - A51001-19

B. Consumable Materials
   (1) D00153 Hydraulic Fluid, Fire Resistant BMS 3-11 (AMM 20-30-04/201)
   (2) D00290 Lubricant - MCS 352B (AMM 20-30-04/201)

C. References
   (1) AMM 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
   (2) AMM 22-00-02/201, Autoflight BITE
   (3) AMM 24-22-00/201, Electrical Power, Control
   (4) AMM 29-11-00/201, Main Left, Center, and Right Hydraulic Systems

D. Access
   (1) Location Zone
       313 Stabilizer Torsion Box Compartment (Left)

   (2) Access Panel
       313AL Controls Bay Access

E. Procedure

S 014-063
(1) Remove the covers from all openings.

S 104-064
(2) Clean the surfaces of the valve that will attach to the servo, the new seal plate and the O-ring with clean hydraulic fluid. Do not allow hydraulic fluid to go to the electrical connector.

S 644-065
(3) Apply a layer of MCS 352B lubricant to the new seal plate and the O-ring.

S 424-014
(4) Install the new O-ring on the valve.

S 424-015
(5) Install the seal plate and the valve with the screws and washers.

S 434-016
(6) Tighten the screws to 60 pound-inches.

S 764-018
(7) AIRPLANES WITH A -4 AND ON ELEVATOR AUTOPILOT SERVO;
Use the bonding meter to make sure that the resistance between the valve assembly and the servo body is 3.0 milliohms (0.0030 ohms) or less.

S 434-019
(8) Put lockwire on the mounting screws.

F. Test Elevator Autopilot Electrohydraulic Servovalve and Solenoid Valves

S 864-020
(1) Supply electrical power (AMM 24-22-00/201).

S 864-021
(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11E17, FLT CONT CMPTR PWR L
(d) 11E18, FLT CONT CMPTR SERVO L
(e) 11E20, FLT CONT CMPTR PWR C
(f) 11E21, FLT CONT CMPTR SERVO C
(g) 11E35, FLT CONT CMPTR PWR R
(h) 11E36, FLT CONT CMPTR SERVO R
(i) 11H17, FLT CONT SHUTOFF TAIL L
(j) 11H18, FLT CONT SHUTOFF TAIL CENTER
(k) 11H27, FLT CONT SHUTOFF TAIL R

S 984-022

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(3) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches in the ON position.

S 864-023
(4) Put the two STAB TRIM switches in the NORM position.

S 864-024
(5) Pressurize the L, C, and R hydraulic systems (AMM 29-11-00/201).

S 984-025
(6) Operate the elevator system several times.

S 744-026
(7) Do MCDP ground test 08-SERVO ELEV.
(a) Make sure that no fault messages are shown after the test starts (AMM 22-00-02/201).

G. Inspect the valves for leakage.

S 864-027

(1) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches in the OFF position.

S 864-028

(2) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
   (a) 11H17, FLT CONT SHUTOFF TAIL L
   (b) 11H18, FLT CONT SHUTOFF TAIL CENTER
   (c) 11H27, FLT CONT SHUTOFF TAIL R

S 214-029

(3) Examine the valves for hydraulic leakage and correct as necessary.

H. Put the Airplane Back to Its Usual Condition

S 864-030

(1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
   (a) 11H17, FLT CONT SHUTOFF TAIL L
   (b) 11H18, FLT CONT SHUTOFF TAIL CENTER
   (c) 11H27, FLT CONT SHUTOFF TAIL R

S 864-031

(2) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches in the ON position.

S 094-032

(3) Remove the service platform and close the controls bay access door, 313AL.

S 864-033

(4) Remove hydraulic power if it is not necessary (AMM 29-11-00/201).

S 864-034

(5) Turn off the MCDP.

S 864-035

(6) Remove electrical power if it is not necessary (AMM 24-22-00/201).

ELEVATOR CONTROL FORCE TRANSDUCER - REMOVAL/INSTALLATION

1. General (Fig. 401)
   A. Each control column has an elevator control force transducer on the crank and arm assembly under the flight compartment. You can access the transducers through the forward equipment bay door, 113AL, and the main electrical/electronics bay access door 119AL (Ref 06-41-00). The removal/installation and adjustment procedures are the same for the two transducers.

TASK 22-12-03-024-001

2. Remove the Elevator Control Force Transducers
   A. Equipment
      (1) A20004-XX Rig Pin Set (AMM 20-10-24/201)
         (a) Rig Pin E-1
         (b) Rig Pin E-2
   B. References
      (1) 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
      (2) 20-10-24/201, Rig Pins
      (3) 22-00-02/201, Autoflight BITE
      (4) 24-22-00/201, Electrical Power - Control
      (5) 29-11-00/201, Main (Left, Center, and Right) Hydraulic Systems
   C. Access
      (1) Location Zones
         113 Area Forward of NLG Wheel Well (Left)
         119 Main Equipment Center (Left)

      (2) Access Panels
         113AL Flight/Landing Gear/Engine Control Components
         119BL Main Equipment Center

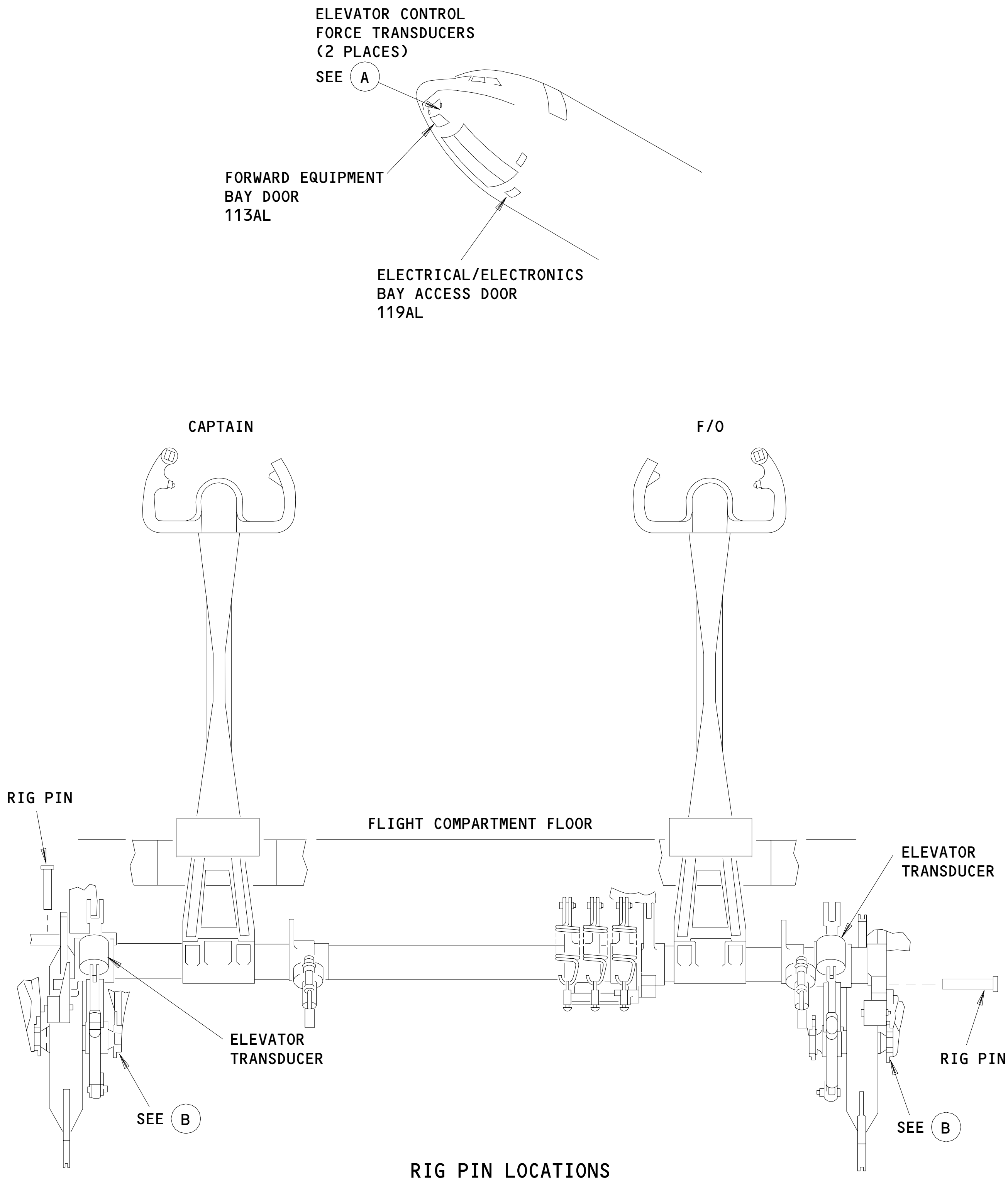
   D. Prepare For Removal

      S 864-002
      (1) Supply electrical power (Ref 24-22-00).

      S 864-003
      (2) Put the L, C, and R TAIL FLIGHT CONTROL SHUTOFF switches, on the P61 right side panel, in the OFF position.

      S 864-004
      (3) Put the two F/D switches on the MCP in the OFF position.

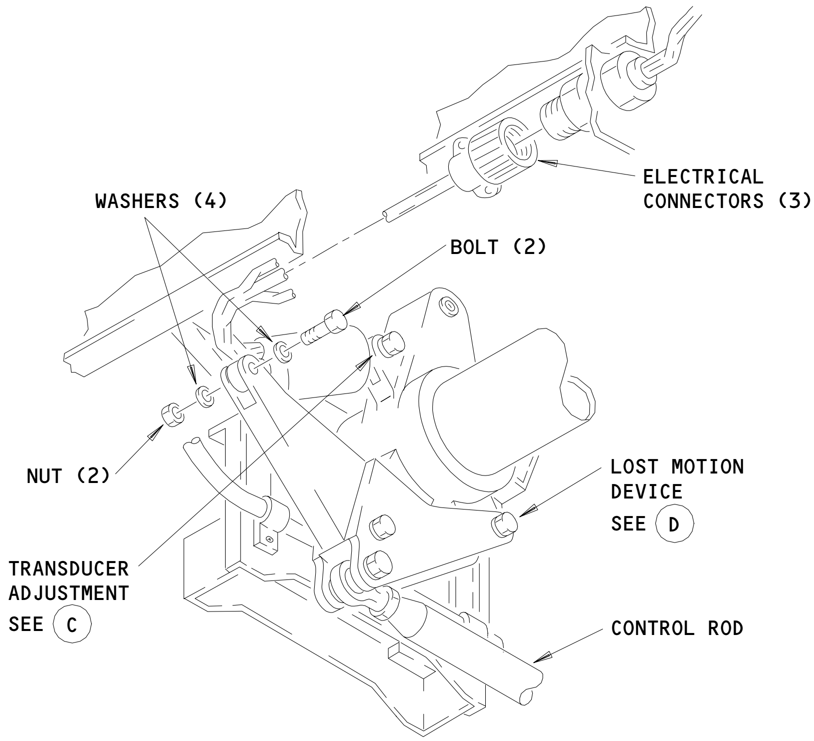
      S 864-005
      (4) Open these circuit breakers on the overhead circuit breaker panel, P11, and install DO-NOT-CLOSE tags:
         (a) 11E16, MODE CONT PNL L
         (b) 11E17, FLT CONT CMPTR PWR L
         (c) 11E18, FLT CONT CMPTR SERVO L
         (d) 11E20, FLT CONT CMPTR PWR C



RIG PIN LOCATIONS

A

Elevator Control Force Transducer
Figure 401 (Sheet 1)



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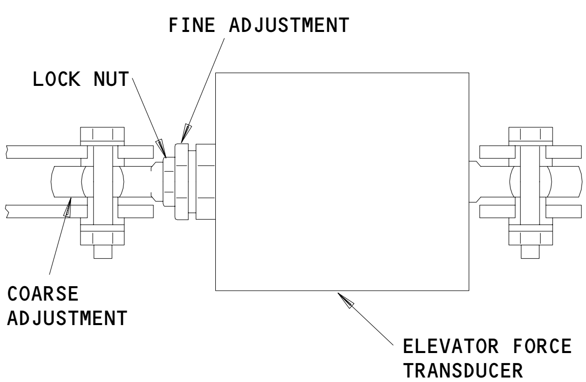

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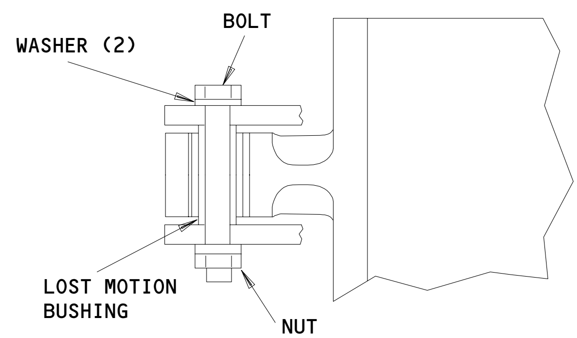


D

Elevator Control Force Transducer
Figure 401 (Sheet 2)

(e) 11E21, FLT CONT CMPTR SERVO C
(f) 11E34, MODE CONT PNL R
(g) 11E35, FLT CONT CMPTR PWR R
(h) 11E36, FLT CONT CMPTR SERVO R
(i) 11H17, FLT CONT SHUTOFF TAIL L
(j) 11H18, FLT CONT SHUTOFF TAIL CENTER
(k) 11H27, FLT CONT SHUTOFF TAIL R

S 014-006
(5) Open the access panel, 113AL (Ref 06-41-00).

S 494-007
(6) For the left transducer, install rig pin E-1 in the left column torque tube.

S 494-008
(7) For the right transducer, install rig pin E-2 in the right column crank assembly.

S 864-009
(8) Attach DO-NOT-OPERATE tags on the pilots' elevator control columns.

E. Procedure

S 034-010
(1) Disconnect the transducer's three electrical connectors.

S 014-011
(2) Remove the transducer wire bundle clamp.

S 014-013
(3) Remove the bolts, washers, and nuts from transducer rod-ends.

S 024-012
(4) Remove the transducer from the arm and crank assembly.

TASK 22-12-03-424-014

3. Install the Elevator Control Force Transducer

A. Equipment
(1) A27025-1 Force Transducer Rigging Equipment
(a) Rod

(b) Bushing
(2) A20004-XX Rig Pin Set (AMM 20-10-24/201)
(a) Rig Pin E-1
(b) Rig Pin E-2

B. References
(1) 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
(2) 20-10-24/201, Rig Pins
(3) 22-00-02/201, Autoflight BITE
(4) 24-22-00/201, Electrical Power - Control
(5) 29-11-00/201, Main (Left, Center, and Right) Hydraulic Systems

C. Access
(1) Location Zones

113	Area Forward of NLG Wheel Well (Left)
119	Main Equipment Center (Left)

(2) Access Panels

113AL	Flight/Landing Gear/Engine Control Components
119BL	Main Equipment Center

D. Procedure

S 424-015
(1) Install the transducer non-adjustable rod-end to the arm assembly with the bolt, washers, and nut.

S 024-040

CAUTION: DO NOT APPLY ANY EXTERNAL LOADS TO THE TRANSDUCER DURING INSTALLATION AND RIGGING.

(2) Loosen the locknut on the transducer.

S 984-017
(3) Adjust the rod-end length and install the transducer on the crank assembly with the bolt (Detail C).

S 014-018
(4) Remove the bolt, washer, and nut from arm and crank assembly.

S 014-019
(5) Remove the lost-motion bushing (Detail B).

S 494-020
(6) Replace the lost-motion bushing with the rigging set bushing.

S 824-021
(7) Adjust the transducer rod-end so that the rigging set pin can be passed freely through the crank assembly and the rigging bushing.

S 984-022
(8) Lock the adjustment and make sure that the pin can still pass freely.

S 754-039
(9) Make sure that there is no load on transducer assembly.

S 024-023
(10) Remove the rigging bushing and install the lost-motion bushing.

S 434-024
(11) Connect the crank and arm assemblies with bolts, washers, and nuts.

S 434-025
(12) Connect the transducer's electrical connectors to airplane wiring.

S 414-026
(13) Install the wire bundle clamp.

S 014-027
(14) Remove rig pins E-1 and E-2.

S 864-028
(15) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11E16, MODE CONT PNL L
(b) 11E17, FLT CONT CMPTR PWR L
(c) 11E18, FLT CONT CMPTR SERVO L
(d) 11E20, FLT CONT CMPTR PWR C
(e) 11E21, FLT CONT CMPTR SERVO C
(f) 11E34, MODE CONT PNL R
(g) 11E35, FLT CONT CMPTR PWR R
(h) 11E36, FLT CONT CMPTR SERVO R
(i) 11H17, FLT CONT SHUTOFF TAIL L
(j) 11H18, FLT CONT SHUTOFF TAIL CENTER
(k) 11H27, FLT CONT SHUTOFF TAIL R

S 864-029
(16) Put the L, C, and R TAIL, FLT CONTROL SHUTOFF switches in the ON position.

S 864-030
(17) Remove the DO-NOT-OPERATE tags from the control columns.

S 864-031
(18) Pressurize the L, C, and R Hydraulic Systems (Ref 29-11-00).

E. Test Elevator Control Force Transducer.

S 744-032
(1) Perform MCDP Test 66 XDCR output and press YES/ADV switch until MCDP top line displays 66 (L or R) COL-LBS/LCR.

S 284-033
(2) Check that MCDP lower line displays 0 ±2 lbs.

<u>NOTE</u>: If the above tolerances are not met, repeat above test with Elevator Feel Pitot System pressurized to 400 knots, otherwise continue.

S 284-034
(3) For left elevator control force transducer, do the MCDP ground test 14-L COL XDCR.
(a) Make sure that no fault messages are shown after test is engaged (Ref 22-00-02).

S 284-035
(4) For right Elevator Control Force Transducer, do the MCDP ground test 15-R COL XDCR.
(a) Make sure that no fault messages are shown after test is engaged (Ref 22-00-02).

F. Put the airplane back to its initial condition.

S 864-036
(1) Turn off the MCDP.

S 864-037
(2) Remove pressure from the L, C and R hydraulic systems if no longer required (Ref 29-11-00).

S 864-038
(3) Remove electrical power if it is not necessary (Ref 24-22-00).

EAS FILTER - REMOVAL/INSTALLATION

1. General
   A. The three elevator autopilot servos (EASs) are mounted on the bulkhead aft of the horizontal stabilizer. The servos are accessed through the access panel 313AL (Ref 06-42-00). Each servo has a LRU hydraulic filter element.
   B. This procedure supplies data for the two piece filter cover that uses a high pressure filter screen and a filter adapter.

TASK 22-12-04-004-002

2. Remove the EAS Filter (Fig. 401)
   A. Equipment
      (1) Service Platform, Control Bay Access Door - A51001-19
   B. References
      (1) 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
      (2) 24-22-00/201, Electrical Power, Control
      (3) 29-11-00/201, Main Left, Center, and Right Hydraulic Systems
   C. Access
      (1) Location Zone
         313 Stabilizer Torsion Box Compartment (Left)

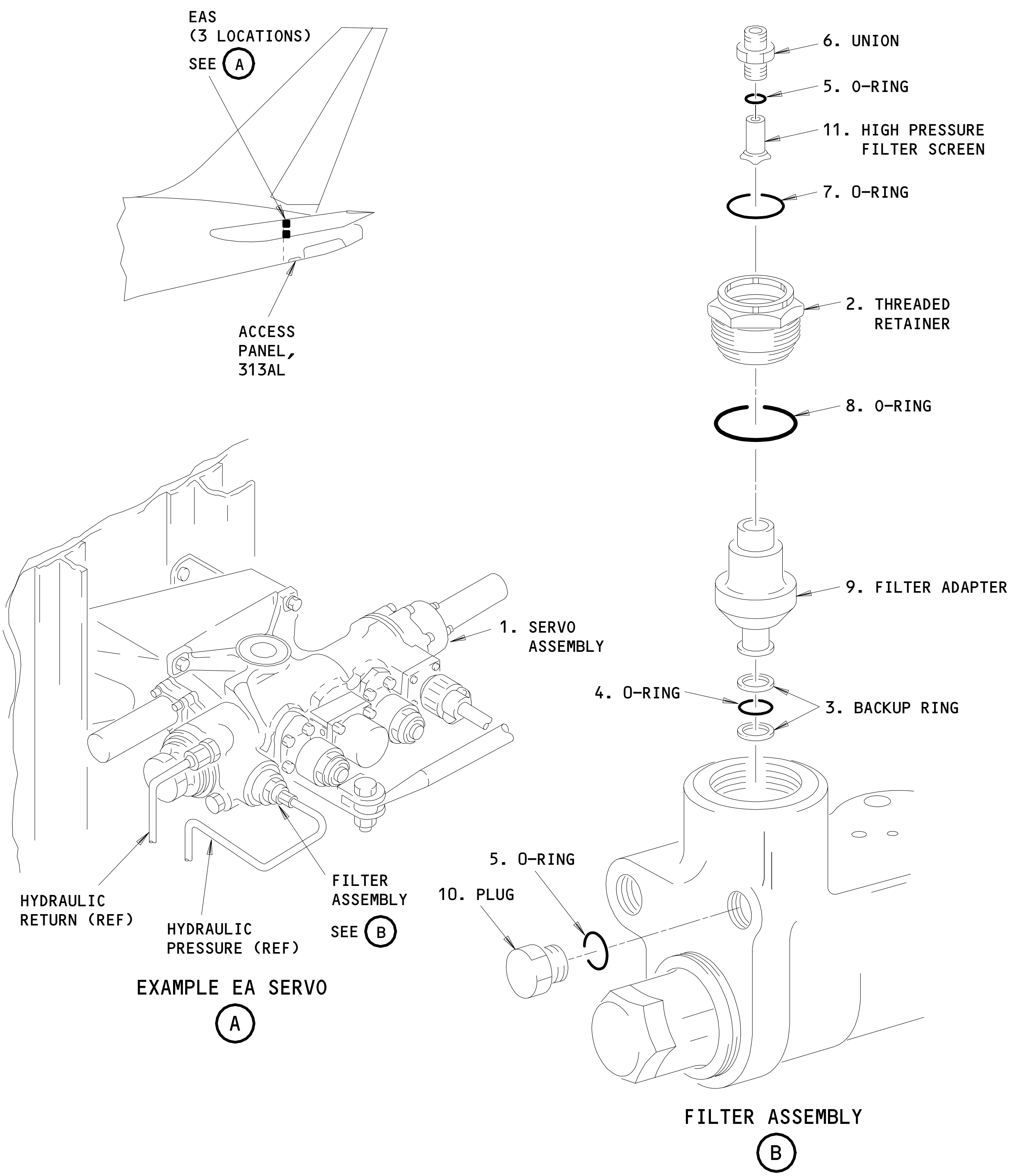
      (2) Access Panel
         313AL Elevator Mechanical Linkages

   D. Prepare for Removal

      S 864-003
      (1) Supply electrical power (Ref 24-22-00).

      S 864-004
      (2) Remove the pressure from the applicable hydraulic system (Ref 29-11-00).
         (a) The left hydraulic system supplies power to the left elevator autopilot servo.
         (b) The center hydraulic system supplies power to the center elevator autopilot servo.
         (c) The right hydraulic system supplies power to the right elevator autopilot servo.

      S 864-005
      (3) Set the L, C, and R TAIL FLT CONTROL SHUTOFF switches on the right side panel, P61, to OFF.



EAS Filter Installation
Figure 401

S 864-006

(4) Set the two STAB TRIM switches on control stand panel, P10, to CUTOUT.

S 864-007

(5) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
   (a) 11C12, STAB TRIM SHUTOFF L
   (b) 11C13, STAB TRIM SHUTOFF CENTER
   (c) 11H17, FLT CONT SHUTOFF TAIL L
   (d) 11H18, FLT CONT SHUTOFF TAIL CENTER
   (e) 11H27, FLT CONT SHUTOFF TAIL R

S 014-058

(6) Open the access door, 313AL (Ref 06-42-00).

S 914-057

WARNING: STAY OFF THE ACCESS DOOR FOR THE CONTROLS BAY, 313AL. YOUR WEIGHT CAN CAUSE THE SPRING-LOADED LATCHES TO RELEASE. IF YOU FALL THROUGH THE DOOR, INJURY CAN OCCUR.

(7) Install a service platform over the access door, 313AL.

E. Remove the Filter

S 164-020

CAUTION: DO NOT LET CONTAMINATION GET INTO THE ELEVATOR AUTOPILOT SERVO AND VALVES. THE DIRECTIONAL AUTOPILOT SERVO AND VALVES ARE VERY SENSITIVE ELECTROHYDRAULIC DEVICES. CONTAMINATION IN THE SYSTEM CAN CAUSE DAMAGE TO THE EQUIPMENT.

(1) Clean the external areas of the filter cover and the servo assembly.

S 034-021

(2) Remove the lockwire.

S 024-022

(3) Remove the filter cover threaded retainer (2) and filter adapter (9) from servo assembly (1).

S 434-023

(4) Put a cover on the opening in the servo to prevent contamination of the hydraulic passages.

S 034-024

(5) Remove the O-rings (7 and 8) from the threaded retainer (2).

S 034-025
(6) Remove the two back-up rings (3) and the O-ring (4) from the filter adapter (9).

S 034-026
(7) Remove the union (6) from the filter adapter (9).

S 034-027
(8) Remove the filter screen (11) from the filter adapter (9).

S 034-028
(9) Remove the O-ring (5) from the union (6).

TASK 22-12-04-404-001

3. Install the EAS Filter (Fig. 401)

A. Equipment
(1) Service Platform, Control Bay Access Door - A51001-19

B. Consumable Materials
(1) D00054 Lubricant - Hydraulic System O-rings, Backup Rings, Fittings MCS 352B

C. Parts

MM		NOMENCLATURE	IPC		
FIG	ITEM		SUBJECT	FIG	ITEM
401	1	Servo Assembly	22-12-01	01	105
	4	O-Ring			185
	5	O-Ring			220
	7	O-Ring			167
	8	O-Ring			165

D. References
(1) 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
(2) 22-00-02/201, Autoflight BITE
(3) 24-22-00/201, Electrical Power, Control
(4) 29-11-00/201, Main Left, Center, and Right Hydraulic Systems

E. Access
(1) Location Zone
313 Stabilizer Torsion Box Compartment (Left)

(2) Access Panel
313AL Elevator Mechanical Linkages

F. Install the Filter

NOTE: Apply MCS 352B lubricant to all O-rings and back-up rings as the parts are installed and assembled.

S 434-043
(1) Install the O-ring (5) onto the union (6).

S 434-044
(2) Install the filter screen (11) into the filter adapter (9).

S 434-045
(3) Install the union (6) into the filter adapter (9).

CAUTION: TIGHTEN THE UNION (6) ON A SURFACE WHICH WILL NOT CAUSE DAMAGE TO THE FILTER ADAPTER (9).

(a) Tighten the union (6) to 110 ±10 pound-inches.

S 434-046
(4) Install the two back-up rings (3) and the O-ring (4) on the filter adapter (9).

S 434-047
(5) Install the O-rings (7 and 8) into the filter cover threaded retainer (2).

S 424-048
(6) Install the filter adapter (9) into the filter "boss" cavity of the servo assembly (1).

S 424-049
(7) Install the filter cover threaded retainer (2) into the servo assembly (1).
(a) Tighten the filter cover threaded retainer (2) to 20 ±2 pound-feet.

G. Elevator Autopilot Servo Test

S 864-051

(1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
   (a) 11C12, STAB TRIM SHUTOFF L
   (b) 11C13, STAB TRIM SHUTOFF CENTER
   (c) 11H17, FLT CONT SHUTOFF TAIL L
   (d) 11H18, FLT CONT SHUTOFF TAIL CENTER
   (e) 11H27, FLT CONT SHUTOFF TAIL R

S 864-052

(2) Set the L, C, and R TAIL FLT CONTROL SHUTOFF switches to ON.

S 864-053

(3) Set the two STAB TRIM switches to NORM.

S 864-054

(4) Pressurize the left, center, and right hydraulic systems (Ref 29-11-00).

S 984-055

(5) Manually operate the elevator system three or more times.

S 714-056

(6) Do MCDP ground test 08-SERVO ELEV (Ref 22-00-02).
   (a) Make sure no failure messages show during the test.

S 794-057

(7) Examine the filter assembly for leaks and correct as necessary.

NOTE: Pressurize the servo for a minimum of 2 minutes before you do a check for leaks.

S 434-062

(8) Attach the filter cover threaded retainer (2) and the plug (10) to the servo assembly (1) with a lockwire.

H. Put the Airplane Back to Its Usual Condition.

S 094-059

(1) Remove the service platform from the access panel 313AL.

S 414-060

(2) Close the access panel 313AL.

S 864-061

(3) Set the MCDP to the OFF position.

S 864-062

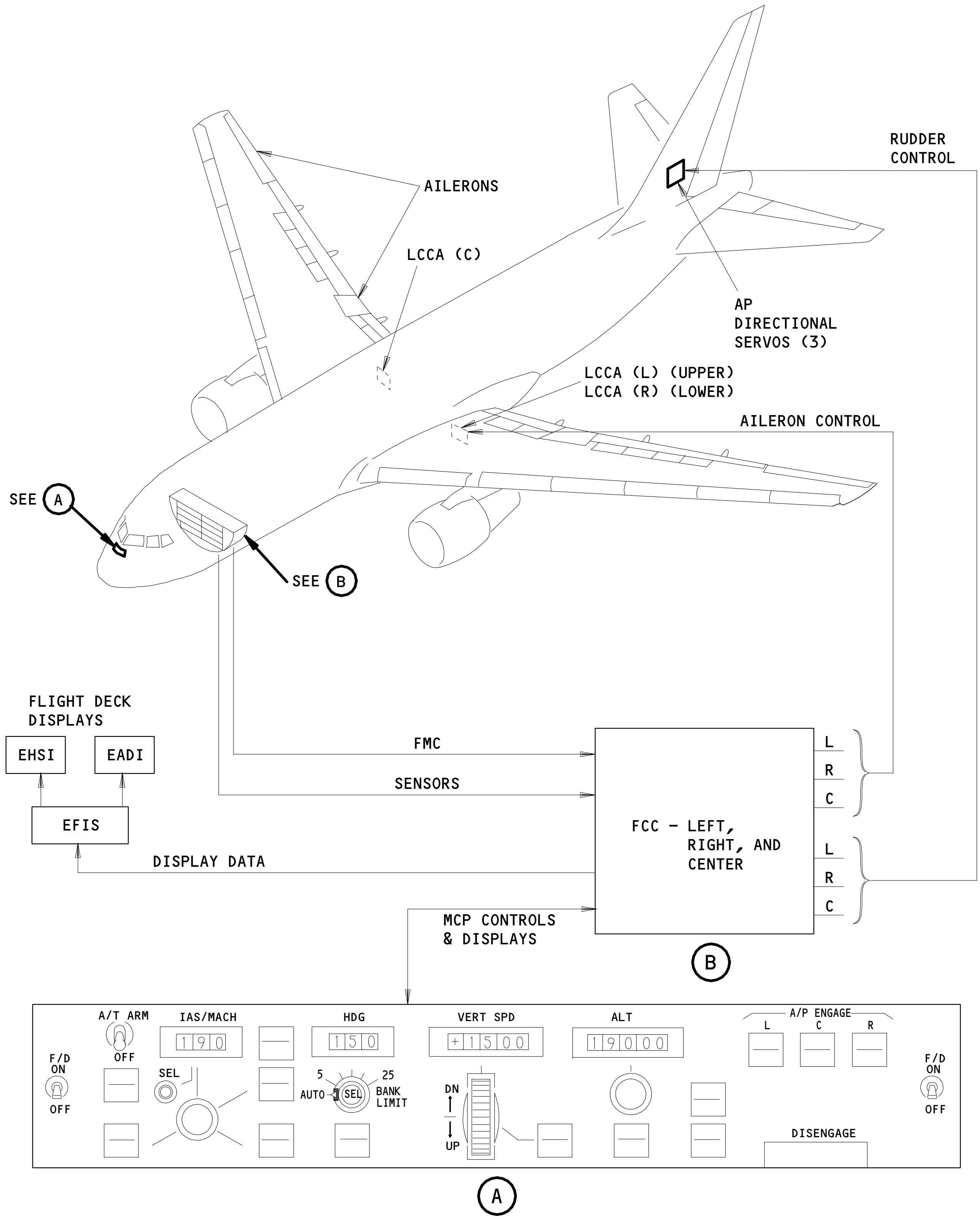
(4) Remove hydraulic power if it is not necessary (Ref 29-11-00).

S 864-063

(5) Remove electrical power if it is not necessary (Ref 24-22-00).

AUTOPILOT/FLIGHT DIRECTOR ROLL AND YAW CHANNEL - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The roll and yaw channels consist of three flight control computers (FCCs), three lateral central control actuators (LCCAs) and three directional autopilot servos (DASs). Each servo is dedicated to a single FCC. Primary control of the A/P is through selections made on the AFCS Mode Control Panel (MCP). Mode and Flight Director (F/D) displays are presented on the Electronic Attitude Director Indicators (EADIs) of the Electronic Flight Instrument System (EFIS).
   B. Each FCC generates F/D commands independently from A/P commands. The F/D commands are supplied to the EFIS symbol generators for display on the EADI. The F/D displays are enabled by the F/D switches on the MCP.
   C. With the A/P not engaged, control wheel rotation provides input to the aileron power control units; rudder pedals and rudder trim control provide inputs to the rudder power control actuators. The output Linear Variable Differential Tranducers (LVDTs) of the LCCA and DAS sense control surface movement. They generate aileron and rudder servo feedback commands for synchronizing the FCC to the control surface positions.
   D. With the A/P engaged in single channel CMD, sensor signals are combined in the FCC and autopilot roll command signals drive the dedicated LCCA. Off-line FCCs synchronize to existing conditions.
   E. Two or three FCCs in CMDENG constitute the multi-channel engage (MCHENG) configuration for Autoland. In MCHENG, each engaged FCC receives dedicated sensor data and drives the dedicated LCCA and DAS with roll and yaw servo commands.
   F. Roll Axis Control Requirements
      (1) Roll axis control functions of the FCC provide roll attitude hold, heading/track hold, and programmed roll attitude control of the airplane. The LCCAs, ailerons, and spoilers provide for roll axis maneuvering, but only the LCCAs and ailerons are controlled by the autopilot.
      (2) If a flight director is on, roll attitude hold and heading/track hold provide F/D commands to maintain the roll attitude of the airplane during aerodynamic disturbances.
      (3) Programmed attitude control is provided automatically:
         (a) To commands from the Flight Management Computer System (FMCS) during lateral steering (LNAV mode).
         (b) During localizer capture and centerline tracking (LOC and APP modes).
         (c) During automatic compensation for wind drift (track conditions).
         (d) During runway alignment maneuvering prior to touchdown (APP mode).



Roll and Yaw Axis Control
Figure 1

G. Yaw Axis Control

(1) Yaw axis control functions provide directional control of the airplane during an autopilot triple channel engaged controlled touchdown and rollout. During an ILS approach with three autopilots engaged, an FCC directional control module aligns the airplane on the centerline prior to touchdown and maintains the airplane on runway centerline after touchdown. During high speed rollout, the rudder is used for steering. Through an interconnect with the rudder pedals, nose wheel steering is used during low speed rollout.

H. Appropriate rate limiting and compensation signals are provided to the roll and yaw channels in each A/P mode. The validity of input data used in each mode is monitored and appropriate actions taken in the event a no computed data (NCD) bit is detected.

I. FCC Roll and Yaw Axes - Inputs and Outputs (Fig. 2)

(1) Figure 2 shows the FCC inputs and outputs related to the roll and yaw axes. The inputs include mode requests (MCP), servo and surface position (LCCA & DAS), flight reference data (IRU and ADC), and navigation data (ILS and RA). Runway length and horizontal steering commands are received from the FMC. The go-around switches on the thrust levers provide on input for engagement of G/A mode.

(2) Outputs are servo arm and engage commands (LCCA and DAS), mode status (MCP), F/D roll commands (EADI), and Flight Mode Annunciations (EADI FMA). Cross-channel data is provided to and from each of the other FCCs.

2. Component Details

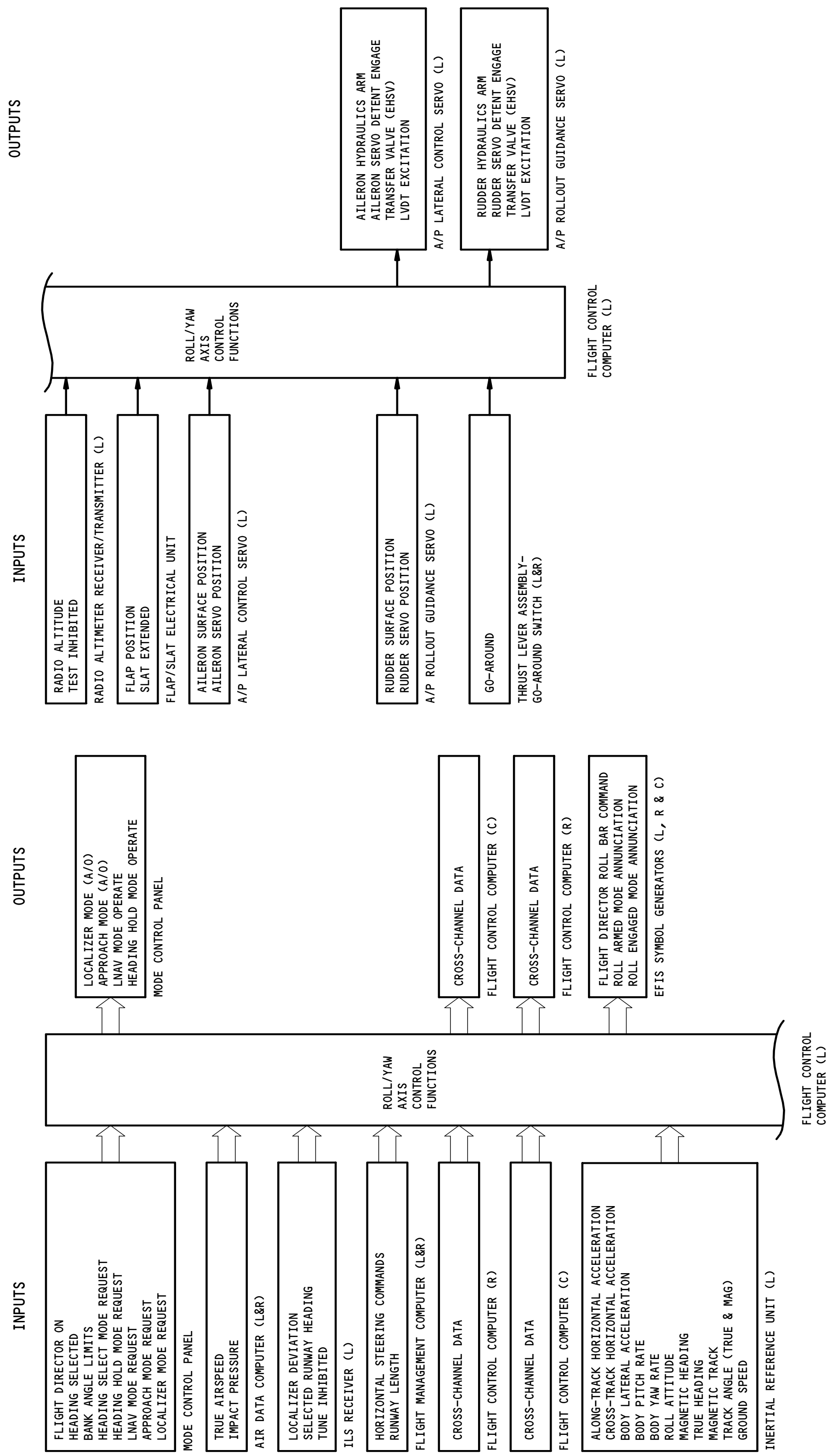
A. Lateral Central Control Actuator - Location and Linkages (Fig. 4)

(1) Locations

(a) The left and right LCCAs are located in the left wing root (left above right). The center LCCA is in the right wing root. All are mounted on the aft face of the rear spar and installed with 4 mounting bolts (Ref 27-11-14, Lateral Central Control Actuator). Two hydraulic lines and an electrical connector complete the installation.

(2) Linkages

(a) The left and right LCCA output shafts are both connected to the left LCCA output quadrant which drives the control cables out to the aileron power control actuators (PCAs) and associated linkage on the left side. The LCCA input cranks are connected to the left torque tube. The torque tube is driven through the feel, centering and trim mechanism, by the control cables from the drive drum on the left quadrant.



FCC Roll and Yaw Axes - Control Inputs and Outputs
Figure 2

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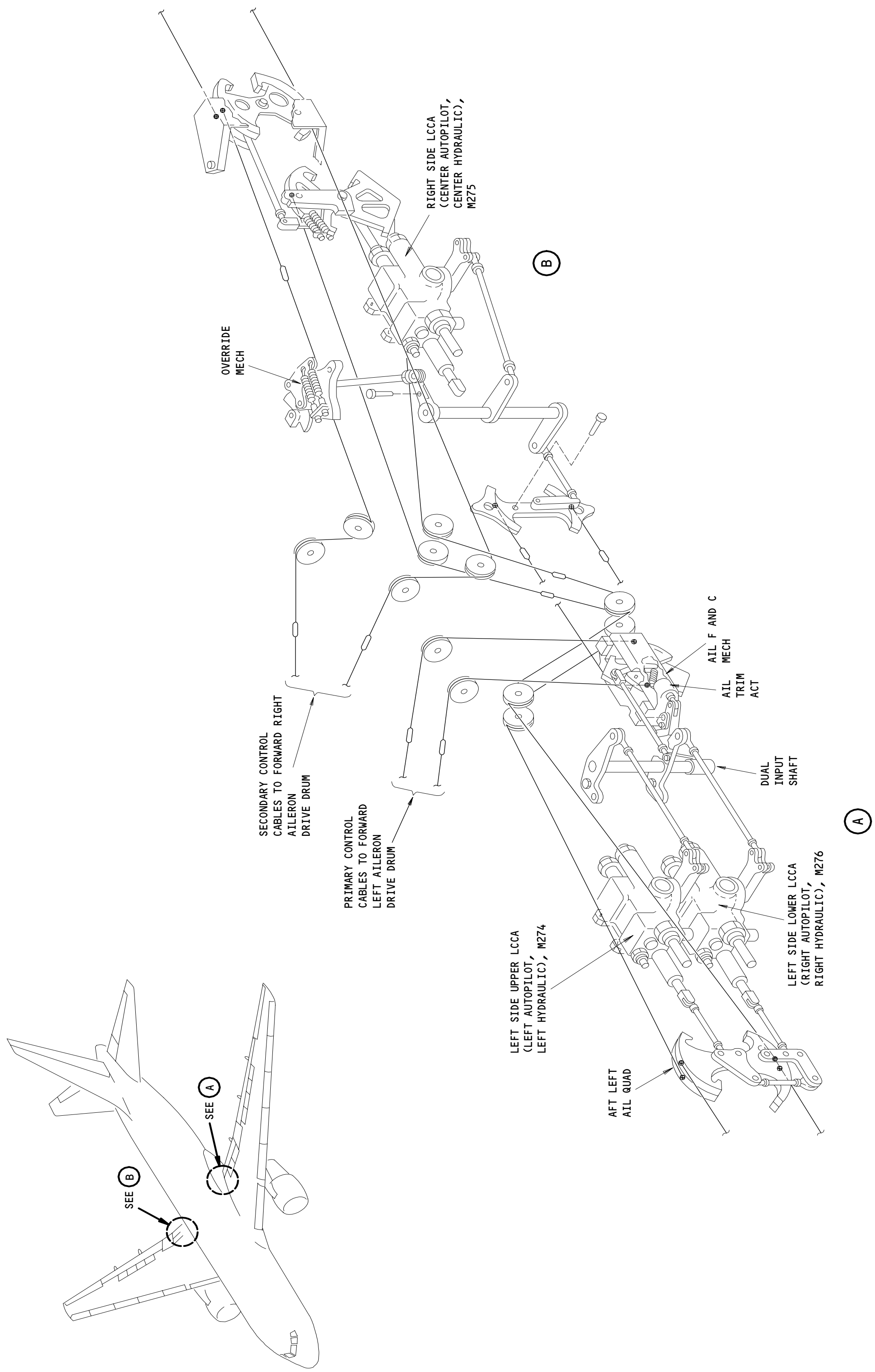
Not Used
Figure 3



LCCA Locations and Linkages
Figure 4

18110

(b) The center LCCA output shaft is connected to the right LCCA output quadrant which drives the control cables to the aileron PCAs and associated linkages on the right side. The right side input crank is connected to the right torque tube which is also driven through the feel, centering and trim mechanism.

(3) Mechanical Drive

(a) The first officer's control wheel cables are connected to the right forward quadrant. The right and left forward quadrants are interconnected by a control rod. Therefore, the first officer operates the ailerons thru the left quadrant. A second set of cables from the right quadrant are driven thru a load limiter on the quadrant. They provide a manual backup in the event any aileron linkages jam. The backup cables bypass the feel, centering and trim mechanism, and LCCAs to operate the aileron PCAs directly.

B. Lateral Central Control Actuator Components (Fig. 5)

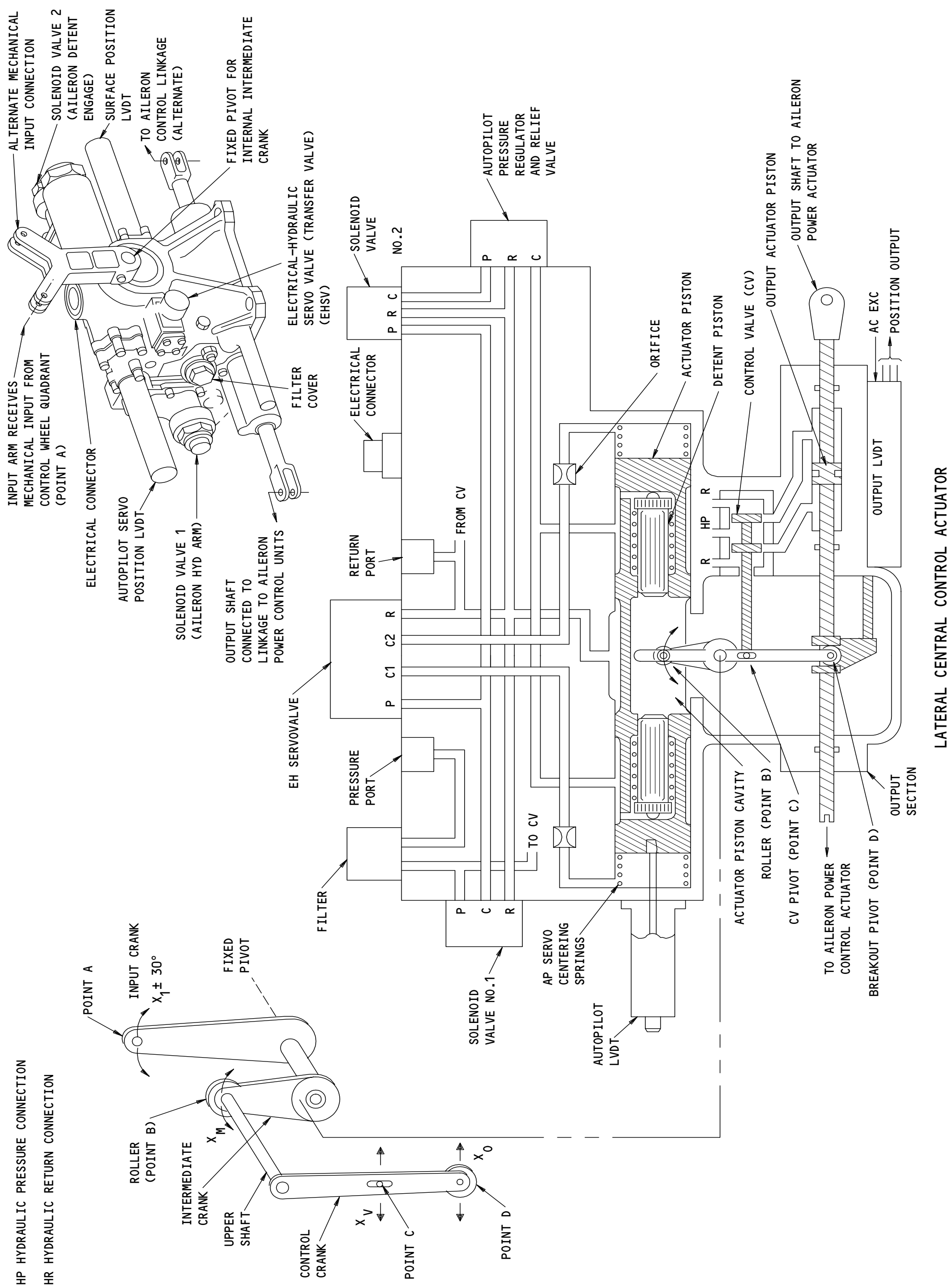
(1) The electrohydraulic control valves (servovalve (1) and solenoid valves (2)) are considered as autopilot LRUs. The following description of LCCA components includes items normally described in Chapter 27. There are repeated here to provide a better understanding of autopilot operation of the LCCA.

(2) Electrohydraulic Control Valves

(a) Solenoid Valves

1) Two solenoid valves (SV1 and SV2) are installed on the LCCA. Each solenoid valve is an electrically operated open-close valve which completes hydraulic pressure through the LCCA when autopilot arm and engage logic circuits are completed. The valve is installed with 4 bolts and sealed with a gasket plate. Electrical pins are mated when the solenoid valve is bolted in position.

2) Solenoid valve number 1 (SV1) opens when AIL HYD ARM DC is applied (autopilot is armed). It ports hydraulic pressure to solenoid valve number 2 (SV2), the electrohydraulic servo valve (EHSV) and the control valve (CV). Solenoid valve number 2 opens when AIL DETENT ENG DC is applied (autopilot is engaged in CMD) and ports hydraulic pressure to the detent pistons. The detent pistons clamp the output linkage crank and transmit actuator piston position to the output linkage.



Lateral Central Control Actuator Components
Figure 5

(b) Electrohydraulic Servovalve (EHSV)

1) One electrohydraulic servovalve is installed on each LCCA. The EHSV contains a sealed torque motor, a feedback spring, a projector jet and a piston. Hydraulic pressure through the valve can be applied to either of two output ports. When no error signal is applied to the torque motor, a small amount of hydraulic fluid flows through a flexpipe attached to the torque motor armature and out of the projector jet. From the jet, equal pressure is applied to opposite ends of the piston holding it at center and closing both output ports.

2) When an error signal is applied, the motor armature rotates in proportion to the magnitude and direction of the input signal and moves the projector jet accordingly. The jet directs more hydraulic pressure to one end of the piston than the other causing it to move and open the corresponding output port to complete hydraulic pressure through the LCCA. When the error signal is nulled, the motor armature and jet return to center. This equalizes the pressure on the piston and, with the aid of the feedback spring, causes the piston to recenter and close both output ports.

3) The EHSV is installed with four bolts and sealed with a gasket plate. Electrical pins are mated when the EHSV is bolted in position.

4) The EHSV is controlled by the aileron error signal (AIL SERVO CMD) from the FCC. In response to this command, the EHSV ports hydraulic pressure to the right or left side of the actuator piston. Feedback from the actuator piston LVDT (linear variable differential transducer) nulls the error signal at the FCC and piston movement stops.

(3) Linear Variable Differential Transducer (LVDT)

(a) The LCCA has two LVDTs. The actuator piston (servo position) LVDT functions as a linear follow-up transmitter for closing the loop around the EHSV. It is operated by the actuator piston. The output position (surface position) LVDT is operated by the intermediate crank, which is connected via the booster actuator piston to the aileron control linkage. Both LVDTs are variable reluctance transformers with an output that varies directly with linear motion. The LVDT uses 26vac excitation from the associated FCC. The LVDTs are not considered line replaceable units. They require nulling adjustments to be completed which match actuator piston with booster piston position when the LCCA is on the bench.

(4) Pressure Regulator and Relief Valves

(a) The pressure regulator and relief valve regulates and limits hydraulic pressure applied to the actuator piston and detent piston (through the EHSV) and the output piston (through the CV). The pressure relief function allows manual inputs from the control linkage to override autopilot control.

(5) Actuator Piston Assembly
  (a) The actuator piston translates autopilot input commands through the EHSV into mechanical positioning of the ailerons. The actuator piston is normally centered by two springs and is moved right or left by hydraulic pressure from the EHSV when the autopilot is armed or engaged. The springs center the actuator piston when pressurization is released.
  (b) Two detent pistons inside the actuator piston are normally retracted by disengage springs. When the autopilot is engaged in CMD, SV2 opens. Hydraulic pressure through SV2 overrides spring tension and locks the detent pistons against the roller of the intermediate crank. Detent pistons may be forced back if sufficient force to overcome hydraulic pressure is applied manually through the control linkage (camout).
  (c) The actuator piston LVDT provides an electrical signal proportional to actuator piston position. This signal nulls out the autopilot aileron position error signal from the FCC to stop movement of the actuator piston.

(6) Cranks
  (a) The input crank, external to the LCCA, receives mechanical input from the control wheel quadrant (point A). It shares a common pivot point with the intermediate crank.
  (b) The intermediate crank is internal to the LCCA and has a roller at the top (point B) which is located between the detent pistons. The crank has limited freedom when the detent pistons are not engaged and rotates about the common pivot point. When the autopilot is engaged, point B becomes a fixed point due to the action of the detent pistons. Actuator piston motion moves the crank with point D as the pivot. The output piston LVDT is connected to the intermediate crank at point E and produces a signal proportional to aileron position.
  (c) The control crank is internal to the LCCA. The lower end (point D) has a roller located between spacers attached to the output shaft. This pivot point moves with piston motion and provides feedback to the control valve connected at point C.

(7) Control Valve
  (a) The control valve is deflected left or right by the motion of the control crank and controls the hydraulic pressure to the output piston. Feedback through the control crank re-centers the control valve and stops piston motion.

(8) Output Piston Assembly
  (a) The output piston (booster actuator) assembly is connected to the aileron linkage. It moves when hydraulic pressure is ported from the control valve. Mechanical feedback is through the control crank to the control valve. The LVDT, connected to the intermediate crank at point E, generates an electrical signal proportional to output shaft displacement. The LVDT is not a line replaceable unit.

(9) Mechanical Control Sequence - Autopilot Disengaged

(a) Initially, the actuator piston is fixed by the centering springs. With the autopilot on but not engaged, SV1 is open, SV2 is closed, the detent pistons are disengaged and the intermediate crank is free to move in the piston cavity. When the input crank is moved clockwise (CW) by control wheel input, the intermediate crank and control crank also move clockwise with the pivot point at D. This moves the control valve to the right in an amount proportional to control wheel angle and ports hydraulic pressure to the right side of the output piston. This moves the output piston and control crank pivot point (D) to the left; mechanical feedback moves the control valve to the left. When the CV has recentered, hydraulic pressure to the output piston is cut off and motion stops. The ailerons, connected through linkages to the output shaft of the booster actuator, are deflected establishing airplane roll rate in accordance with control wheel position.

(b) The output position LVDT transmits output piston position to the dedicated FCC for autopilot synchronization. The FCC transmits a command to the EHSV, and the actuator piston follows the FCC input to keep the intermediate crank centered in the cavity.

(10) Autopilot Control Sequence - Autopilot Engaged

(a) With autopilot engaged, SV1 and SV2 are both open, the detent pistons pressurized and the intermediate crank is clamped in the center position. When the EHSV receives a command from the dedicated FCC, hydraulic pressure is ported to the actuator piston. The detent pistons carry the intermediate crank along with the actuator piston. Action of the control valve and booster actuator piston is the same as described for the mechanical sequence. The control wheel follows the motion of the ailerons as commanded by the FCC input. The two LVDTs transmit piston position to null the error signal at the FCC and stop piston movement.

(11) Camout

(a) Camout occurs when the output crank position (surface position) does not correspond to the actuator piston position (servo position) as sensed by differing LVDT outputs. Causes of camout are mechanical jamming of output linkage, pilot override of an engaged autopilot channel, or mechanical override of one autopilot channel by the remaining two during multi-channel operation.

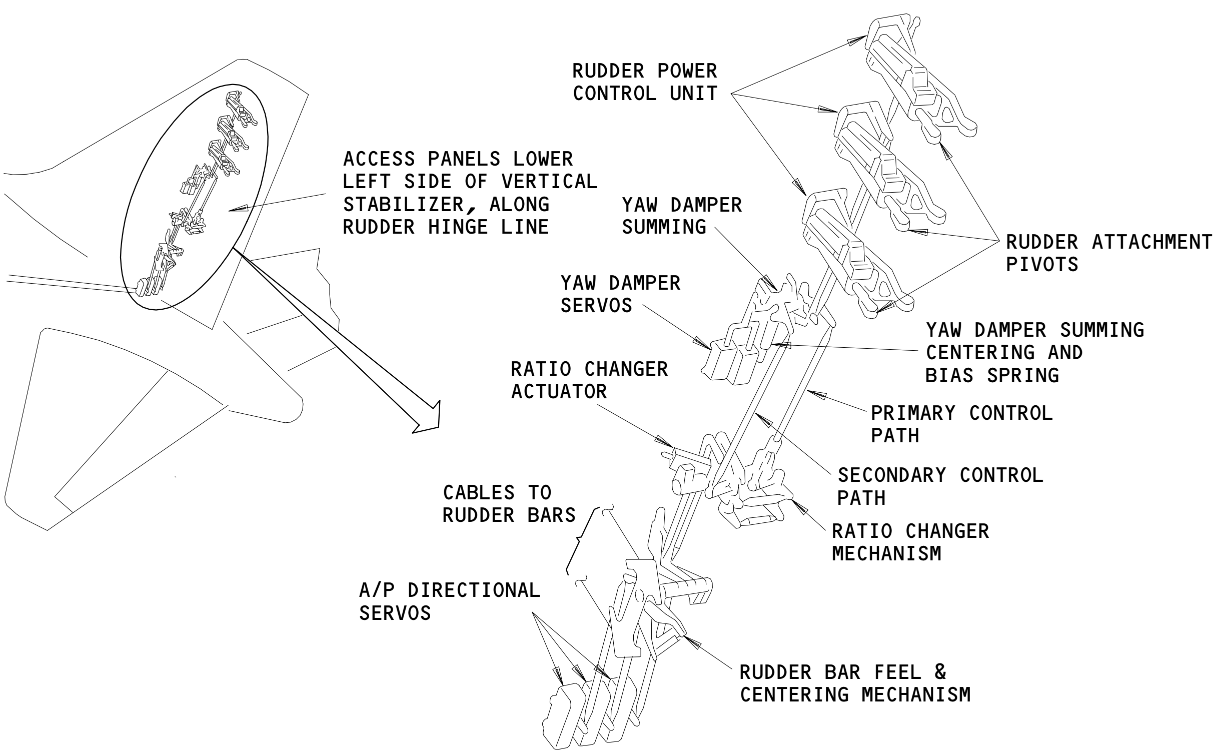
C. Directional Autopilot Servos - Location and Linkage (Fig. 6)

(1) Three directional autopilot servos (DASs) are side by side at the base of the vertical stabilizer immediately forward of the rudder hinge line. Each servo has an output rod connected to the rudder feel, centering and trim mechanism on the aft quadrant. Access to the servos is through a removable panel on the lower left side of the vertical stabilizer.

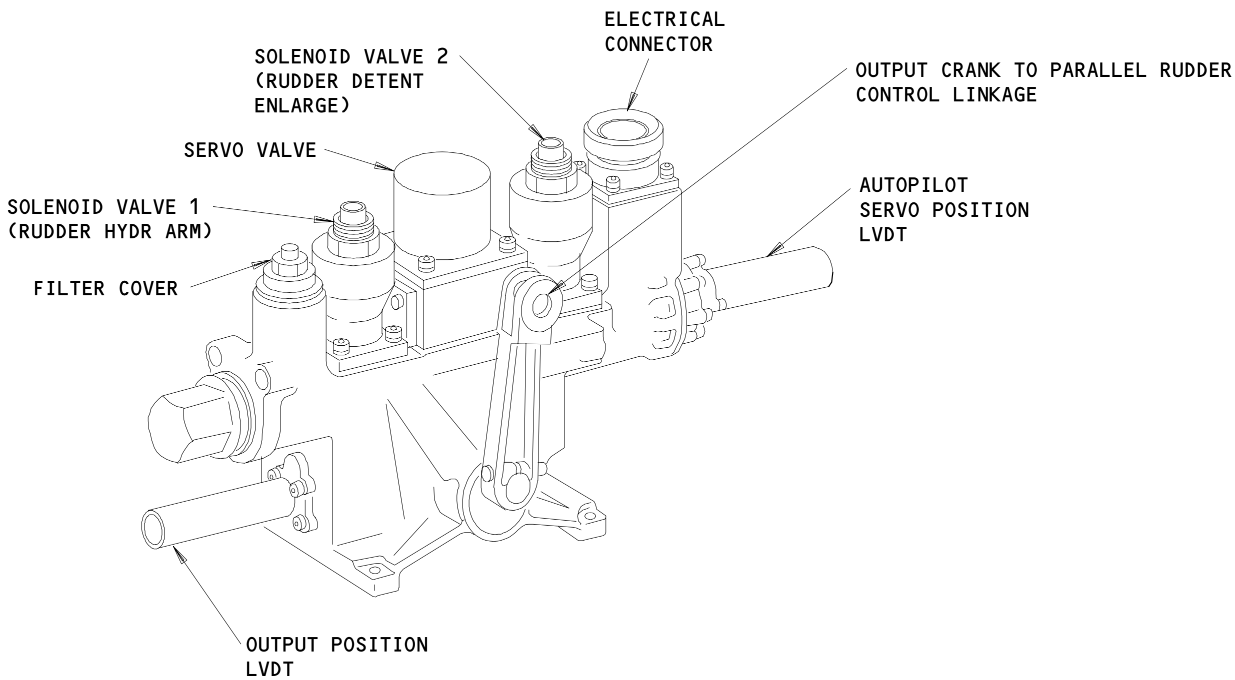
(2) Each DAS has its own drive linkage to the aft quadrant which provides feedback to the rudder pedal. The rudder control system has dual mechanical drive links to the ratio changer mechanism with primary and backup linkages to the yaw damper summing lever. There is no feedback from the yaw damper to the rudder pedals. Three power control actuators (PCAs) use a common mechanical input to deflect the rudder surface.

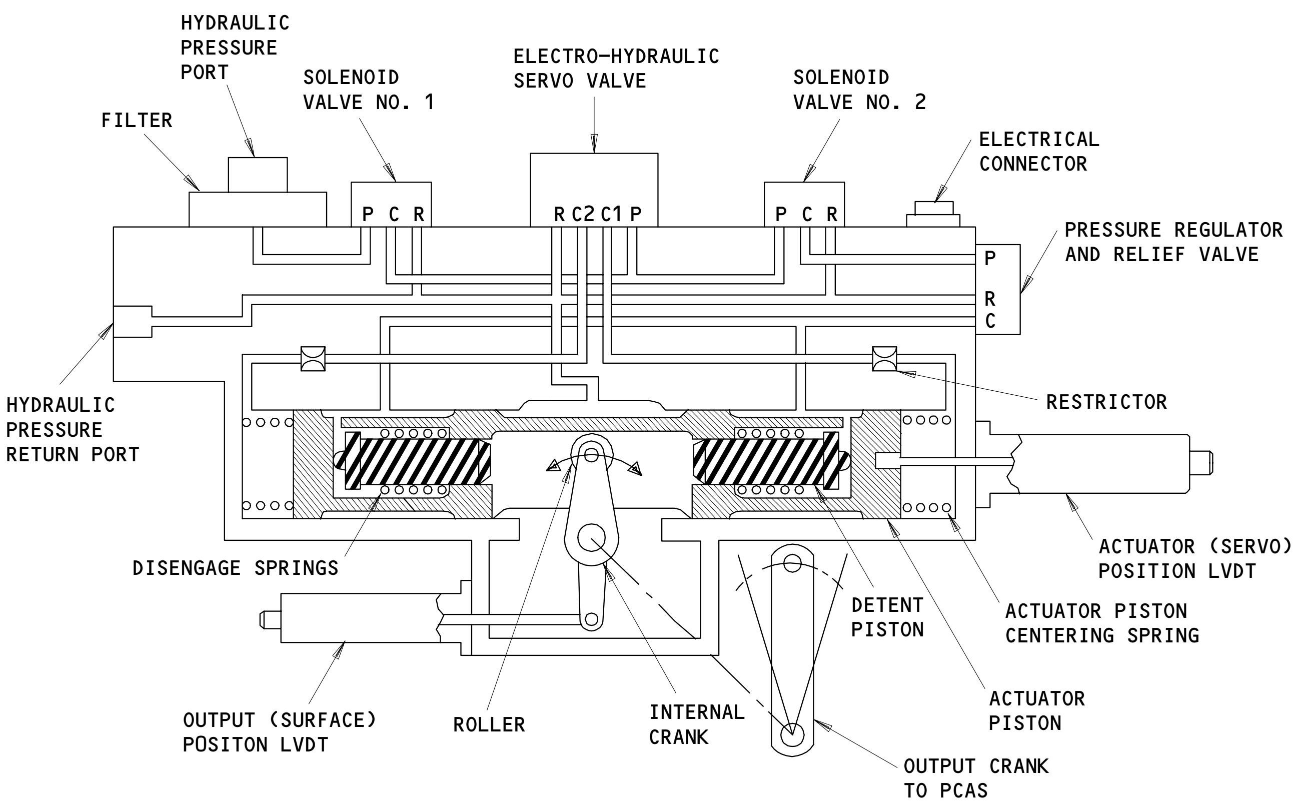
D. Directional Autopilot Servo Components (Fig. 7)

(1) Each Directional Autopilot Servo (DAS) contains two solenoid valves, an electrohydraulic servo valve (EHSV), two linear variable differential transducers (LVDTs), and a pressure regulator/relief valve. Hydraulic power is applied through pressure and return ports at one end of the servo. A 25-micron filter is built into each unit.



Directional Autopilot Servos - Location and Linkages
Figure 6





Directional Autopilot Servo Components
Figure 7

(2) Solenoid Valves
   (a) Each solenoid valve (SV1 and SV2) is an electrically operated open-close valve which completes hydraulic pressure through the servo when autopilot arm and engage logic circuits are completed. The valve is installed with 4 bolts and sealed with a gasket plate. Electrical pins are mated when the solenoid valve is bolted in position.
   (b) Solenoid valve number 1 (SV1) opens when the autopilot is armed. It ports hydraulic pressure to solenoid valve number 2 (SV2) and the electrohydraulic servo valve (EHSV). Solenoid valve number 2 opens when the autopilot is engaged, and ports hydraulic pressure to the detent pistons. The detent pistons clamp the output linkage crank and transmit actuator piston position to the output linkage.

(3) Electrohydraulic Servovalve (EHSV)
   (a) The EHSV is controlled by the output command signal from the FCC. The EHSV contains a sealed torque motor, a feedback spring, a projector jet and a piston. Hydrualic pressure through the valve can be applied to either of two output ports. When no command signal is applied to the torque motor, a small amount of hydraulic fluid flows through a flexpipe attached to the torque motor armature and out of the projector jet. From the jet, equal pressure is applied to opposite ends of the piston holding it at center.
   (b) When a command signal is applied, the motor armature rotates in proportion to the magnitude and direction of the input signal and moves the projector jet accordingly. The jet directs more hydraulic pressure to one end of the piston than the other causing it to move and open the corresponding output port to complete hydraulic pressure through the servo. When the command signal is nulled, the motor armature and jet return to center. This equalizes the pressure on the piston and, with the aid of the feedback spring, causes the piston to recenter and close both output ports.
   (c) In response to the FCC output command, the EHSV ports hydraulic pressure to the right or left side of the actuator piston. Feedback from the actuator piston LVDT nulls the command signal at the FCC and piston movement stops.
   (d) The EHSV is installed with four bolts and sealed with a gasket plate. Electrical pins are mated when the EHSV is bolted in position.

(4) Linear Variable Differential Transducer (LVDT)
   (a) Each servo has two identical LVDTs. The actuator piston (servo position) LVDT functions as a linear follow-up transmitter for closing the loop around the EHSV. It is operated by the actuator piston. The output position (surface position) LVDT is operated by the intermediate crank, which is connected to the surface control linkage. Both LVDTs are variable reluctance transformers with an output that varies directly with linear motion. The LVDT uses 26vac excitation from the associated FCC. The LVDTs are not considered line replaceable units. They require nulling adjustments to be completed which match actuator piston with piston position when the servo is on the bench.

(5) Pressure Regulator and Relief Valves
   (a) The pressure regulator and relief valve regulates and limits hydraulic pressure applied to the actuator piston and detent pistons. The pressure relief function allows manual inputs from the control linkage to override autopilot control.

(6) Actuator Piston Assembly
   (a) The actuator piston translates autopilot input commands through the EHSV into mechanical positioning of the control surface. The actuator piston is normally centered by two springs and is moved right or left by hydraulic pressure from the EHSV when the autopilot is armed or engaged. The springs center the actuator piston when pressurization is released.
   (b) Two detent pistons inside the actuator piston are normally retracted by disengage springs. When the autopilot is engaged, SV2 opens. Hydraulic pressure through SV2 overrides spring tension and locks the detent pistons against the roller of the internal crank. Detent pistons may be forced back if sufficient force to overcome hydraulic pressure is applied manually through the control linkage (camout).
   (c) The actuator piston LVDT provides an electrical signal proportional to actuator piston position. This signal nulls the autopilot command signal from the FCC to stop movement of the actuator piston.

(7) Cranks
   (a) The internal crank roller is clamped by the detent pistons when the autopilot is engaged. The crank moves with the actuator piston. Motion of the internal crank moves the output position LVDT. One end of the output crank is directly connected to the internal crank at a common pivot point. The other end of the output crank is connected to the control surface linkage.

(8) Mechanical Control Sequence - Autopilot Disengaged
(a) Initially, the actuator piston is fixed by the centering springs. With the autopilot not engaged (armed), SV1 is open, SV2 is closed, detent pistons are disengaged, and the internal crank is free to move within the piston cavity. The output position LVDT provides internal crank position to the FCC for autopilot synchronization to surface position. The FCC commands through the EHSV cause the actuator piston to follow the internal crank so that the crank remains centered within the cavity. The actuator piston position LVDT nulls the command signal to stop the actuator piston.
(9) Autopilot Control Sequence - Autopilot Engaged
(a) With the autopilot engaged, SV1 and SV2 are open, the detent pistons are pressurized and the internal crank is clamped in the center of the actuator piston. When the EHSV receives a command from the FCC, hydraulic pressure is ported to one end of the actuator piston. The detent pistons carry the internal crank with the actuator piston to its commanded position. The output crank moves the linkage to the surface power control acutators and the output position LVDT sends position information back to the FCC to null the command signal and stop surface movement.
(10) Camout
(a) Camout occurs when the output crank position (surface position) does not correspond to the actuator piston position (servo postion) as sensed by differing LVDT outputs. Causes of camout are mechanical jamming of output linkage, pilot override of an engaged autopilot channel, or mechanical override of one autopilot channel by the remaining two during multi-channel operation.

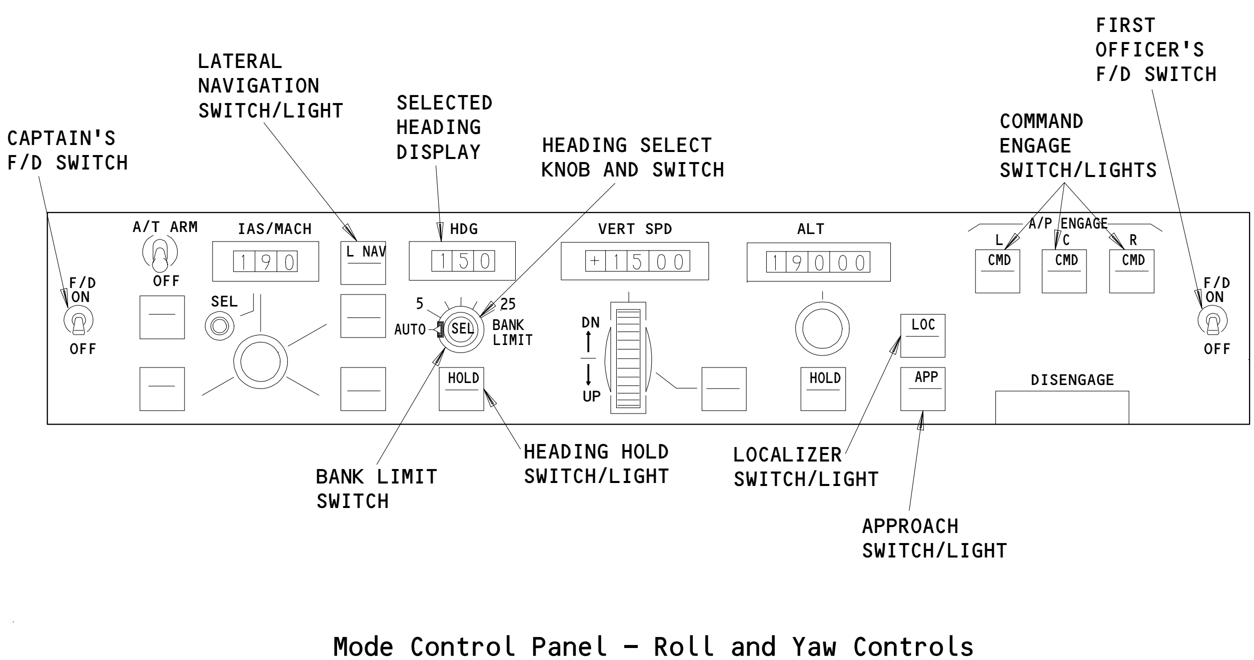
3. <u>Operation</u>
A. Functional Description
(1) Mode Control Panel - Roll and Yaw Controls (Fig. 8)
(a) Flight Director ON/OFF Switches
1) Two F/D toggle switches control the left and right flight director displays.
(b) Command Engage Switches
1) Three CMD ENGAGE switches (L, C, R) provide inputs for command engage logic to dedicated FCCs.
(c) Other MCP controls associated with the roll and yaw modes.
1) LNAV switch/light
2) HDG HOLD switch/light
3) HDG SEL knob
4) HDG SEL switch (within HDG SEL knob)
5) Selected HDG display
6) LOC switch/light
7) APP switch/light
(2) Command Roll Mode Functions (Fig. 9)

(a) Simultaneous Modes

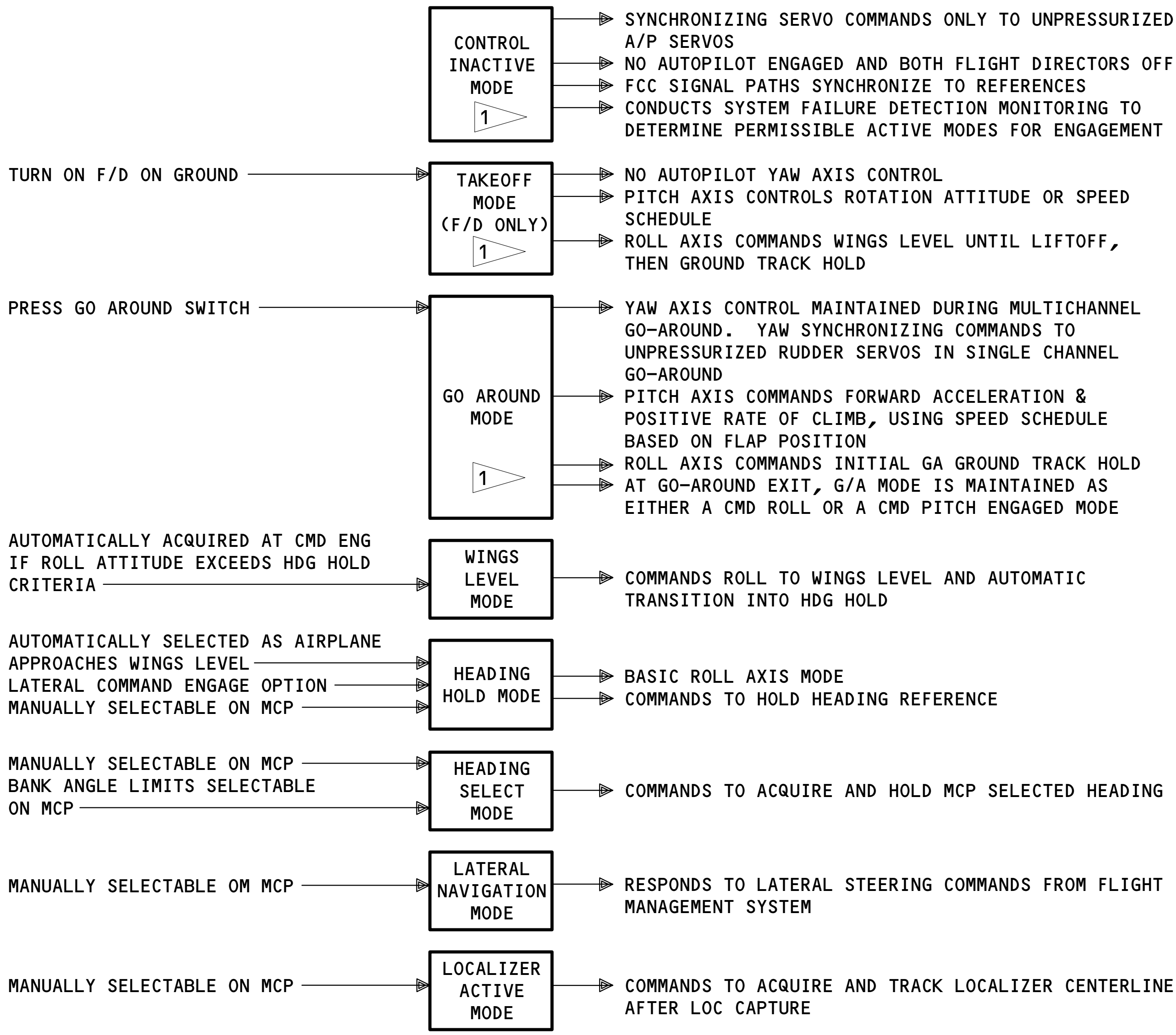
1) Simultaneous modes are those in which both pitch and roll control laws engage at the same time. These modes are Control Inactive, Takeoff (T/O), and Go-Around (G/A).

(b) Control Inactive

1) Control Inactive is the configuration of the AFDS when no autopilot or flight director is engaged. During control inactive, the FCC processes roll angle commands from Runway Alignment, Heading/Track, and Localizer Processors to produce the Roll Attitude Command. This maintains a roll attitude reference when the A/P is not engaged and generates a roll to wings level command at CMDENG. The aileron servo command synchronizes to aileron position when the A/P is not engaged. This permits a smooth transition to A/P control at CMDENG. Processors not active in command computations synchronize to maintain a zero output.



Mode Control Panel - Roll and Yaw Controls
Figure 8



1 SIMULTANEOUS PITCH AND ROLL MODES.

Command Roll Modes
Figure 9

(c) Takeoff Mode (F/D only)
   1) The takeoff mode has two submodes: on-ground and in-air. In the on-ground submode, the F/D provides an 8 degree nose-up command for initial climbout. The F/D also provides a wings level roll command.
   2) In the in-air submode, the F/D pitch command is referenced to the speed schedule with flap position and engine failure corrections. The F/D roll commands provide ground track hold after lift-off. There is no F/D yaw axis control.

(d) Go Around
   1) The AFDS provides commands for a safe positive rate of climb using a speed reference schedule based on flap position. Roll channel commands initiate go-around (G/A) ground track hold. This mode is manually selected by pressing the go-around switch on the throttles.

(e) Lateral Command Engage Option (LCEO)
   1) The lateral command engage option (LCEO) is selected by grounding a dedicated pin on the FCC. The baseline configuration is LCEO inoperative (pin ungrounded). In this configuration, the A/P engages into HDG HLD mode at CMDENG regardless of the bank angle (B/A).

(f) Autopilot Mode Engage Option (APMEO)
   1) The APMEO is enabled by grounding a dedicated pin on the FCC. The baseline configuration is APMEO enabled. This option causes the A/P to couple smoothly into the path command modes being used by the F/D when the A/P engages in CMD (except during takeoff or go-around). The G/S ACTIVE, LOC ACTIVE, and G/A modes transition to V/S and HDG HLD modes and rearm the G/S and LOC if they were armed or active before A/P CMD ENGAGE.

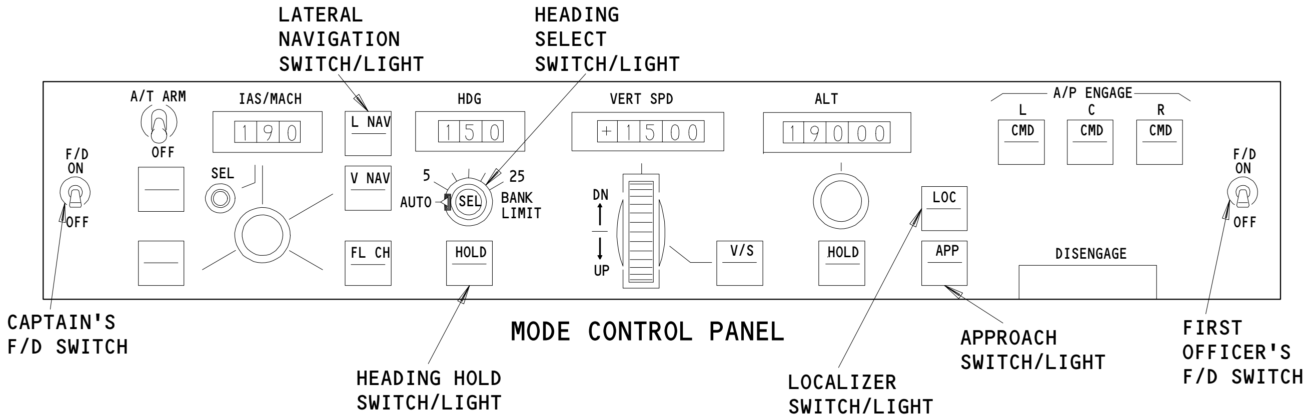
(g) Heading Hold
   1) Heading hold is the basic roll axis mode. It is automatically selected when the F/D only mode is initially chosen or when any A/P channel is engaged in single-channel CMD. Heading hold can also be manually selected by pressing the HOLD switch/light on the MCP.

(h) Heading Select Mode
   1) Heading select mode acquires and maintains the heading selected on the MCP. It has six selectable bank angle limits: AUTO and 5-25 degrees in 5-degree steps. The AUTO bank angle limit varies from 15 to 25 degrees as a function of true airspeed. The heading select mode can be manually called by pushing the HDG SEL switch on the MCP.

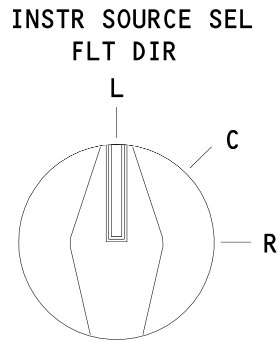
(i) Lateral Navigation Mode (LNAV)
   1) The LNAV mode responds to FMC lateral steering commands to acquire and track selected flight plan paths. It is manually selected by pressing the LNAV switch/light on the MCP.

(j) Localizer Active Mode
1) The LOC active mode provides aileron commands to acquire and track the centerline of the localizer beam during all phases of single or multi-channel ILS approach. The A/P yaw channel provides rudder commands during multi-channel aproachs to maintain runway alignment during rollout.

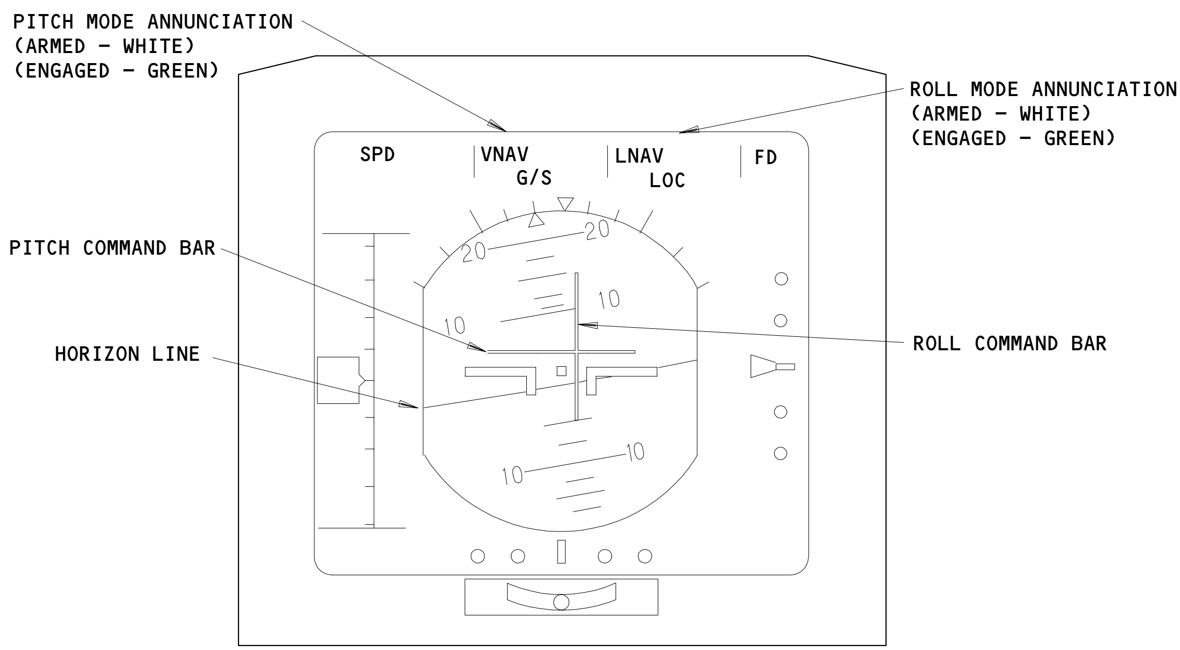
(3) Roll Flight Director Operation (Fig. 10)
(a) The Flight Director (F/D) permits the pilot to fly the airplane without engaging the autopilot (A/P). The FCC computes F/D commands and displays these on the EADIs. The pilot controls the airplane by control column and control wheel movements in response to these commands.
(b) When on the ground, the F/D automatically engages into the takeoff-wings level mode when it is first switched on. It changes to the takeoff-track hold mode when the airplane reaches five feet radio altitude. During flight, if the A/P is engaged when the F/D is first switched on, the F/D engages into the current A/P mode.
(c) Manually selectable F/D modes are:
1) Heading Select (HDG SEL)
2) Heading Hold (HDG HLD)
3) Lateral Navigation (LNAV)
4) Localizer (LOC)
5) Approach (APP)
6) Go-Around (G/A)
(d) Flight Director Controls and Displays
1) Flight Director operation and mode selection are accomplished by switches and switch/lights on the Mode Control Panel. The captain's and first officer's F/D switches are paralleled, so either can be used to turn the F/D on. Both must be off to disable the F/D. Each F/D switch, however, independently controls the F/D displays on the corresponding EADI. The L, C, or R FCC may be selected as data source for the captain's or first officer's F/D displays by the respective FLT DIR switch (P1-1 or P3-3 panels).



MODE CONTROL PANEL



CAPTAIN'S FLIGHT DIRECTOR SWITCH
(F/O'S SIMILAR)



ELECTRONIC ATTITUDE DIRECTOR INDICATOR

Flight Director Controls and Displays
Figure 10

(4) Takeoff Mode (F/D only) (Fig. 11)

(a) Preparatory Operations

1) Before the takeoff mode can be entered, either or both flight directors must be on and all A/P CMD ENGAGE switches must be off. The F/D only configuration is verified by the AFDS status FMA on the EADI. The IAS/MACH indicator is set at V2 (climb speed); HDG is set to the heading of the initial flight leg after takeoff; and ALT is set for the initial clearance altitude.

(b) Flight Director Display

1) Initially (A) the EADI shows wings level and rotation attitude of 8 degrees nose up. The status FMA shows F/D and both the pitch and roll operative mode FMAs show TO.

2) On reaching 5 feet altitude (B), roll channel commands provide track hold and the pitch rotation attitude command reduces linearly toward zero within 3 seconds. The roll operative mode FMA changes to TRK HLD.

3) The F/D enters the takeoff speed engage mode (C) which produces pitch attitude commands to maintain a safe climbout speed.

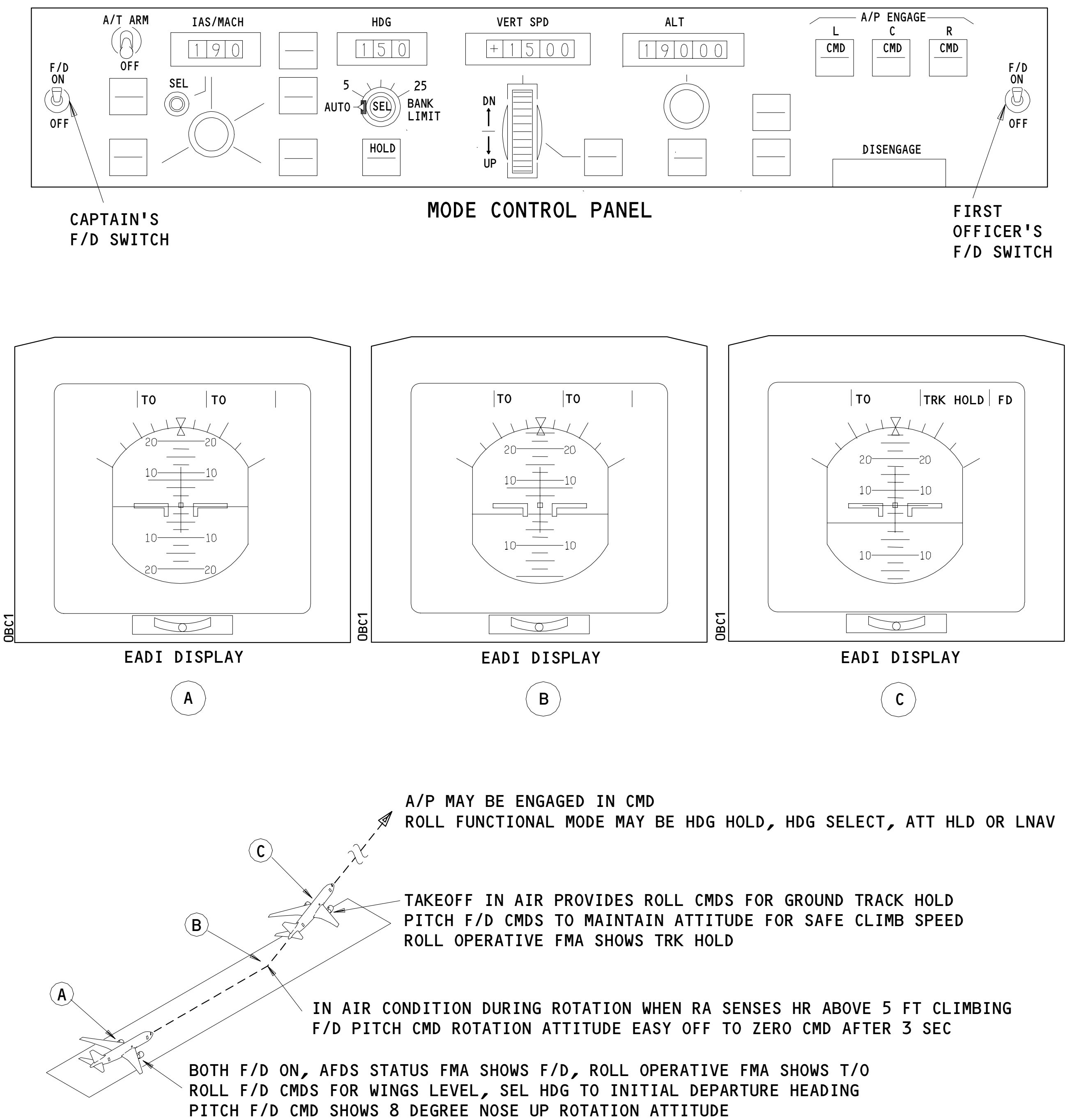
(c) Mode Transitions

1) From the takeoff mode, transition into the following modes is possible: roll channel, HDG HLD, HDG SEL, LNAV ARM, and LNAV capture; pitch channel, ALT HLD, V/S, FLCH, and VNAV. The takeoff mode ends when the A/P is engaged or when both F/Ds are switched off.

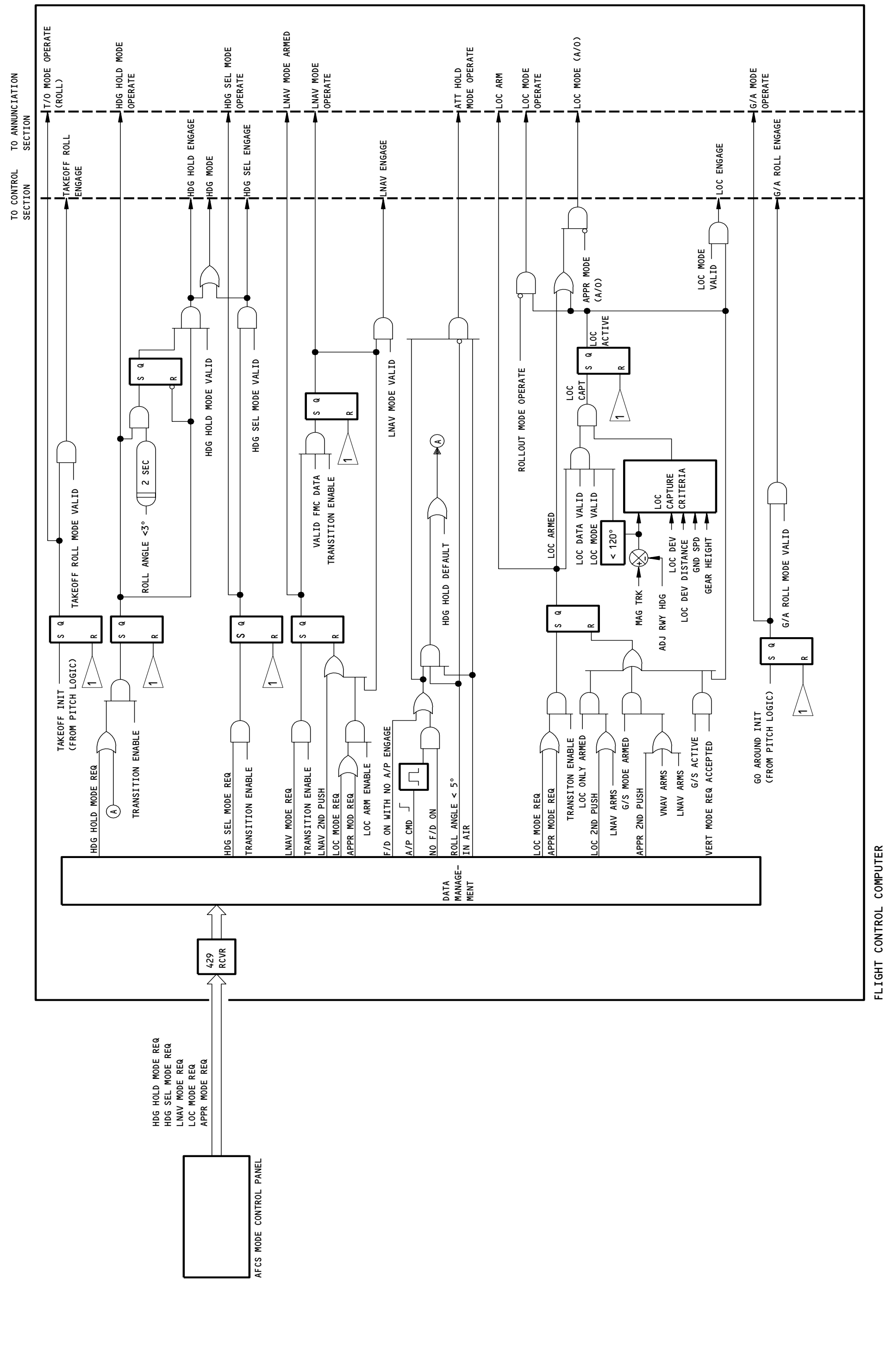
(5) Autopilot/Flight Director System Roll and Yaw Modes Logic (Fig. 12)

(a) Takeoff Mode Logic

1) When either Flight Director is switched ON, F/D engage logic enables the takeoff mode. The on-ground condition generates roll attitude washout logic which produces a roll F/D command of wings-level. At five feet radio altitude, the Heading Hold mode is automatically engaged. This changes the roll F/D commands from wings-level to track hold.



Takeoff Mode (F/D Only)
Figure 11



1 SYSTEM OFF OR ENTER ANOTHER MODE

Autopilot/Flight Director System Roll and Yaw Modes Logic
Figure 12

202792

(6) Autopilot/Flight Director System Roll and Yaw Control Signals Flow (Fig. 13)

(a) Takeoff Mode Control Signal Flow

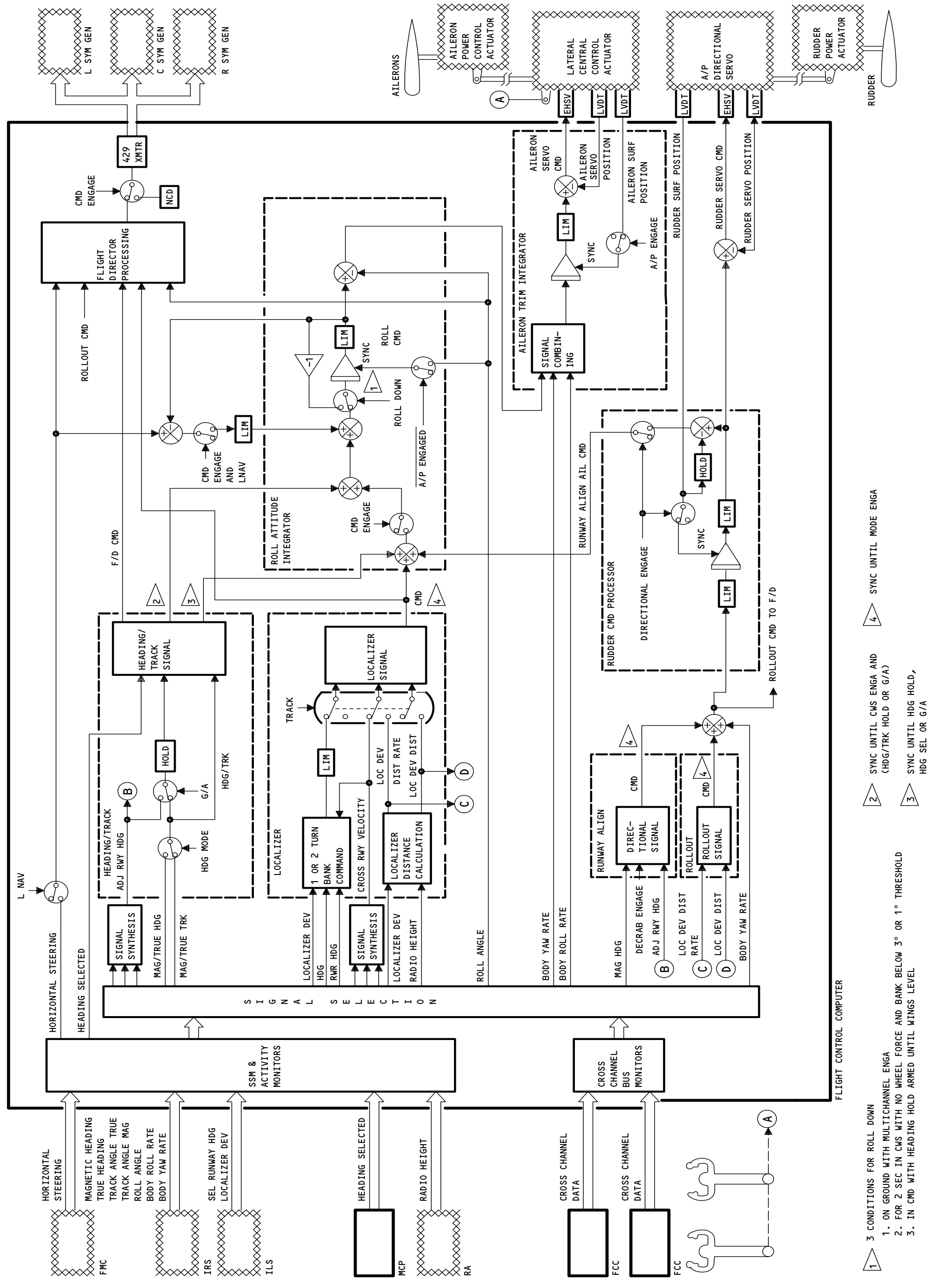
1) When the F/D is turned ON and the airplane is on the ground, the F/D command processor integrator is set to wings-level. The difference between roll attitude and the integrator output is roll attitude error. The roll error is combined with body roll rate and pitch attitude to produce the roll F/D bar command which is applied to the EADI. The wings-level command is provided until the in-air condition is sensed.

2) The in-air condition occurs when the radio altitude is five feet or greater. At this time, adjusted track is used as the input to the heading hold synchronizer and the turn direction command processor. However, heading hold logic disables the integrator in the heading hold synchronizer so that only adjusted track is used to produce the F/D heading command. The F/D heading command is applied to the roll F/D command processor when the roll attitude washout logic goes to zero at five feet radio altitude. The input is limited by the position of the bank angle (B/A) limit switch on the MCP.

(7) Output Signal Synchronization

(a) Aileron Servo Command

1) Aileron surface position from the LCCA is applied to the Aileron Trim Integrator (ATI). A synchronization loop established when the A/P is not engaged, synchronizes the roll servo command to ±65 degrees of wheel rotation. With the A/P engaged, the limiter is set at ±18 degrees in single channel operation and ±40 degrees in multi-channel operation. Any servo command in response to aileron position change is supplied to the EHSV. Since the ALCS is hydraulically armed, the servo actuator moves in response to the EHSV. The actuator piston LVDT output nulls the command. The EHSV centers but the servo actuator piston is off center and the detent pistons are retracted. When APENG occurs, detents clamp the crank at the off-center position.



Autopilot/Flight Director System
Roll and Yaw Control Signal Flow Schematic
Figure 13

(b) Flight Director Roll Command
1) Flight Director roll commands are computed continuously whether the A/P is engaged or not and whether the F/D is on or off.
2) The computed F/D roll command is transmitted to the EFIS. The F/D ON discrete is not present when F/D switches are OFF. Therefore, EFIS displays are inhibited.

(c) Rudder Servo Command
1) Rudder surface position from the DAS is applied to the rudder command processor. A synchronization loop is established when directional yaw steering (DYS) is not engaged. This synchronizes the rudder commands up to ±25 degrees. With DYS engaged, servo commands are applied to the EHSV. Servo actuator operation and LVDT output is similar to that described above for the LCCA. Output to the Runway Alignment Command Signal Processor (RACSP) is held at zero by synchronizing loop. When DYS is engaged, the RACSP receives only the rudder servo command.

(d) Computed Commands
1) The roll F/D command for aileron and the rudder command are individually synchronized in appropriate sections of the FCC. The source for F/D commands depends on the mode. Computed F/D commands are applied to the symbol generators and EADIs of the EFIS. During CMD ENG, a fixed bias is applied and the F/D command bars on the EADI are biased out of view.

Not Used
Figure 14

(8) Heading Hold and Heading Select Modes (Fig. 15)

(a) The MCP controls and displays associated with the heading modes are identified on Fig. 15.

(b) The heading hold mode can be engaged by pressing the HDG HOLD switch/light. If the bank angle (B/A) is greater than the heading capture criteria threshold, the AFDS commands a roll to wings-level and then commands to heading hold. If the B/A is within the heading capture criteria threshold, the AFDS commands to heading hold when the HDG switch/light is pressed.

(c) Heading select mode is used to change to a new heading. The HDG SEL knob and switch on the MCP control the selected heading and the heading select mode. The AFDS commands generated depend on whether the SEL switch is pressed before or after the SEL knob is turned to provide the selected heading.

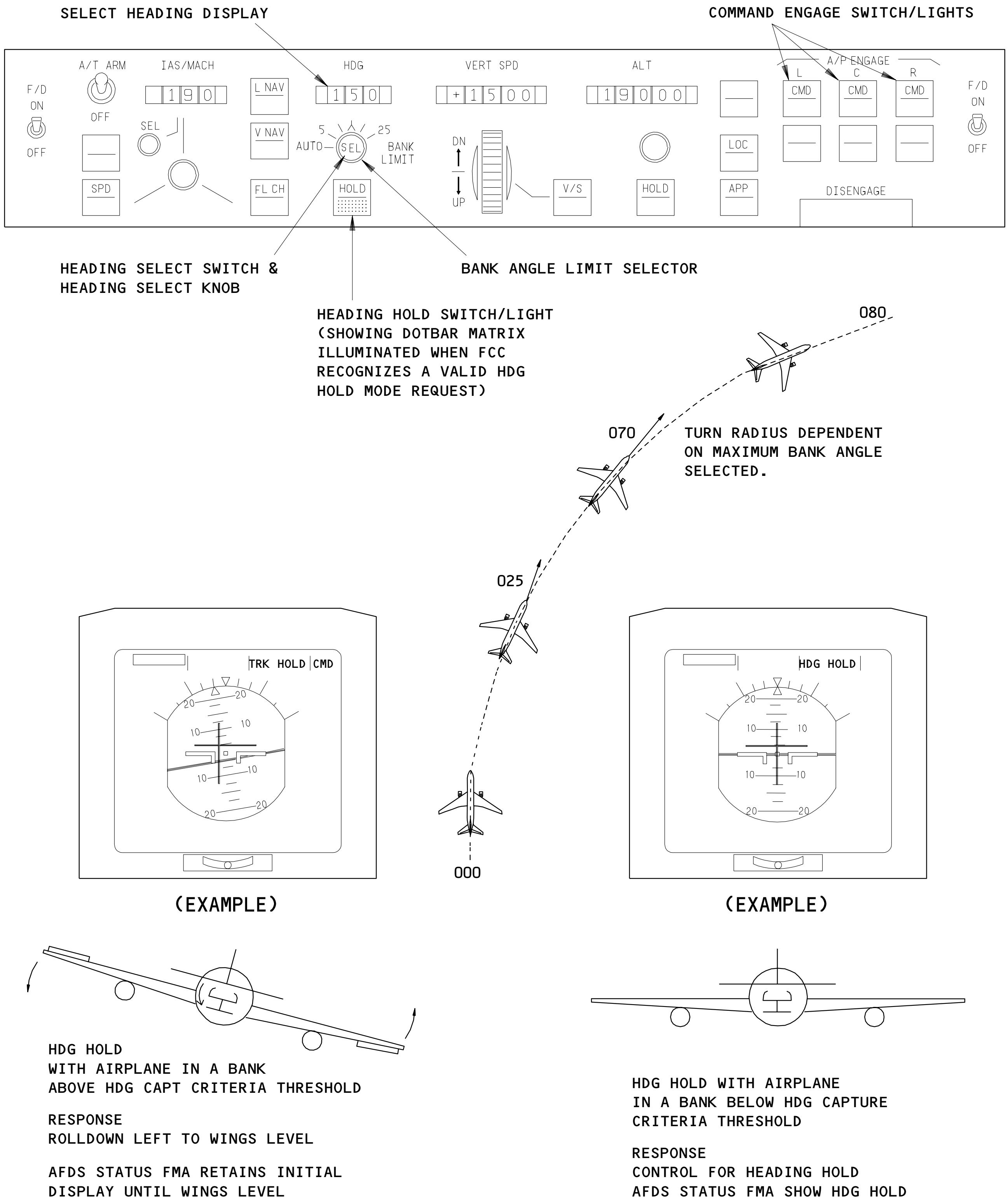
1) If the HDG SEL knob is rotated to the desired heading before the SEL switch is pressed, the AFDS commands a turn towards that heading in the direction of the smallest turn angle.

2) If the SEL switch is pressed first, the AFDS commands generated are always in the direction the HDG SEL knob is turned.

(d) Figure 15 shows an airplane at heading 000 (A) turning to a heading of 080 (D). After the SEL switch has been pressed, the airplane (B) rolls to a B/A proportional to the heading error but is limited by the B/A limit switch on the MCP. The F/D command bars on the EADI are centered indicating that correct roll command is being applied. At C, the heading error has reduced to 10 degrees right requiring a reduction in B/A to permit smooth rollout at 80 degrees. The EADI shows reduced roll attitude and roll left command. The heading of 080 at D is equal to the selected heading so the AFDS automatically engages into heading hold and maintains the selected heading.

(9) Heading Hold/Heading Select Mode Logic (Fig. 12)

(a) The heading hold mode is initiated by pressing the HDG HOLD switch/light. If adjusted heading is valid, the heading hold mode armed logic is generated. When the heading hold capture criteria logic is zero, indicating a B/A greater than the threshold, the roll attitude washout logic is generated. When the heading capture criteria logic goes high, a heading hold request is generated. One second later, heading hold engage is generated and the washout logic goes to zero. The heading hold mode logic is generated and this initiates the command heading/track control and the heading proportional command logic signals.



Heading Select/Heading Hold Mode
Figure 15

(b) Heading select mode enable logic is generated by pressing the HDG SEL switch. The heading select engage valid logic is generated when a selected heading is entered through the HDG SEL knob. These two logic signals produce the heading select mode engaged logic which then produces the heading/track and heading proportional logic.

(10) Heading Hold/Heading Select Signal Processing (Fig. 13)

(a) Heading/Track command processing computes the heading proportional command, the heading integral command, and the flight director heading command. It permits synchronization of heading computation software when the A/P is not engaged in a heading control mode. It selects the correct heading reference for the heading control mode in use and ensures that turn commands produce the correct turn direction for the mode in use. Gain programming is applied as required.

(b) Initial Conditions - Heading Hold Mode

1) When the airplane is on a given heading, magnetic heading is applied as roll attitude reference to both the autopilot and flight director turn direction controls. The heading command feedback from the heading command processor (the heading proportional command) is also magnetic heading. The output of both turn direction controls is zero and the heading integral command and the flight director command are both zero (i.e., no aileron command and F/D command bar is centered).

(c) Roll F/D Command - Heading Select Mode

1) When the HDG SEL button is pressed, the heading select mode is engaged and the selected heading from the MCP HDG SEL knob is compared with the magnetic heading. The difference is applied to both turn direction controls. Output from the F/D turn direction control is scaled to produce the F/D heading command. This is then modified by other inputs: roll attitude; roll attitude rate; and, if LNAV is engaged, horizontal steering commands. This produces the roll F/D command which drives the roll command bar on the EADI.

(d) Aileron Servo Command

1) Selected heading is integrated and limited by the B/A switch setting and the heading rate command filter. This output is summed with heading angle command and becomes the heading proportional command. The integrator output is also summed with magnetic heading and developed into the heading integral command. Localizer and runway alignment commands are added, when appropriate, and the roll rate integral command and delta roll attitude command are produced. The latter is differentiated and added to the former to produce the roll attitude command.

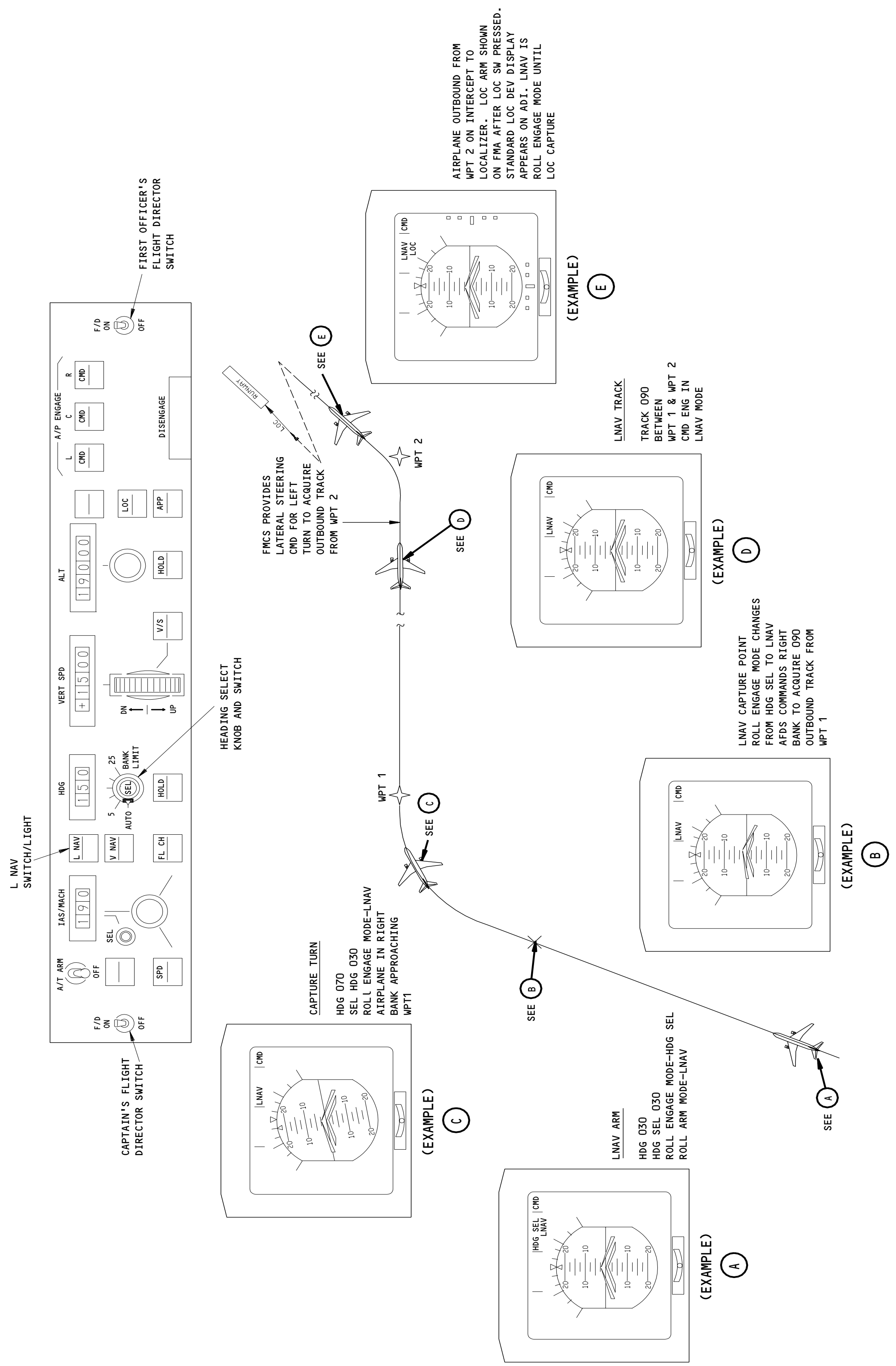
2) Roll attitude and roll attitude command are summed to produce roll attitude error. This error signal is summed with body roll rate, differentiated and summed with aileron servo position feedback to produce wheel rate. This is then applied to the aileron trim integrator which sets max and min values for surface deflection. After integration, wheel rate becomes the aileron servo command which is applied to the electrohydraulic servo valve (transfer valve). The amplitude of the aileron servo command determines the rate of response (roll rate) while the duration of the command determines the bank angle achieved.

(e) As the airplane turns toward the selected heading, the difference between magnetic and selected heading is reduced. Both the aileron servo command and the roll F/D command are reduced in magnitude. When the heading is within capture limits, the roll attitude washout logic is generated and the airplane rolls down to wings level. The heading hold mode is then engaged.

(11) Lateral Navigation (LNAV) Mode (Fig. 17)

(a) In the LNAV mode, the FMC provides lateral steering commands to the FCC to follow a selected flight plan. Applicable controls on the MCP for this mode are the LNAV switch/light and the HDG display.

Not Used
Figure 16



LNAV Mode
Figure 17

(b) LNAV ARM Mode (A)

1) The LNAV ARM mode is initiated during HDG SEL (roll engaged mode) by pressing the LNAV switch/light. The mode is annunciated on the EADI. The AFDS commands to hold selected heading until a valid LNAV steering command is received from the FMC at LNAV capture.

(c) LNAV Capture (B) and Capture Turn (C)

1) At LNAV capture, the FMC supplies valid lateral steering command (LNAVC). Detection of LNAVC changes the roll active mode to LNAV and the EADI roll engaged annunciations change to LNAV and LNAV ARM. The LNAVC commands a right turn to acquire flight plan track at WPT 1.

(d) LNAV Mode (D)

1) The FMC supplies LNAVC to maintain the FMC flight plan track from WPT 1 to WPT 2. Automatic compensation is applied to compensate for wind drift. There is no change in EADI indications. The FMC provides lateral steering commands at WPT 2 for a left turn to acquire the outboard track.

(e) LNAV and LOC ARM (F)

1) The LOC ARM submode is initiated by pressing the LOC switch/light on the MCP. The roll channel is armed for capture of selected ILS frequency. The EADI displays LOC as the roll armed mode but the AFDS responds to LNAVC until LOC capture. The EADI displays LOC and GS. Localizer mode will be discussed later.

(12) LNAV Mode Logic (Fig. 12)

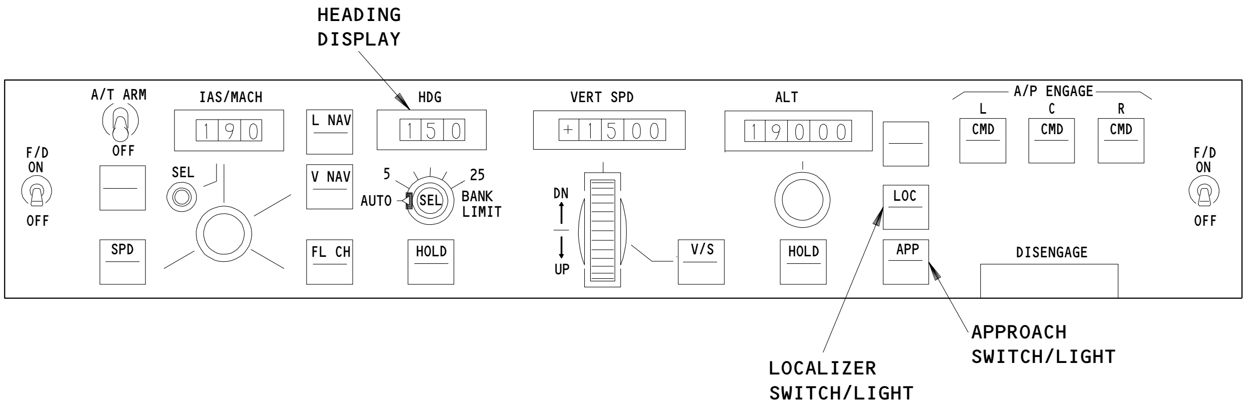
(a) When the LNAV switch/light is pressed and accepted, this request is processed by mode transition logic to produce the LAT NAV logic. When the FMC indicates a valid horizontal steering command (LNAV capture) the LAT NAV engage valid logic is produced. This develops the LNAV engage logic. If one or more channels are in CMD ENG, the LNAV command input logic (LNAVIN) is produced. The LNAV engage and LNAVIN logic signals are used in signal processing.

(13) LNAV Mode Control Signal Flow (Fig. 13)

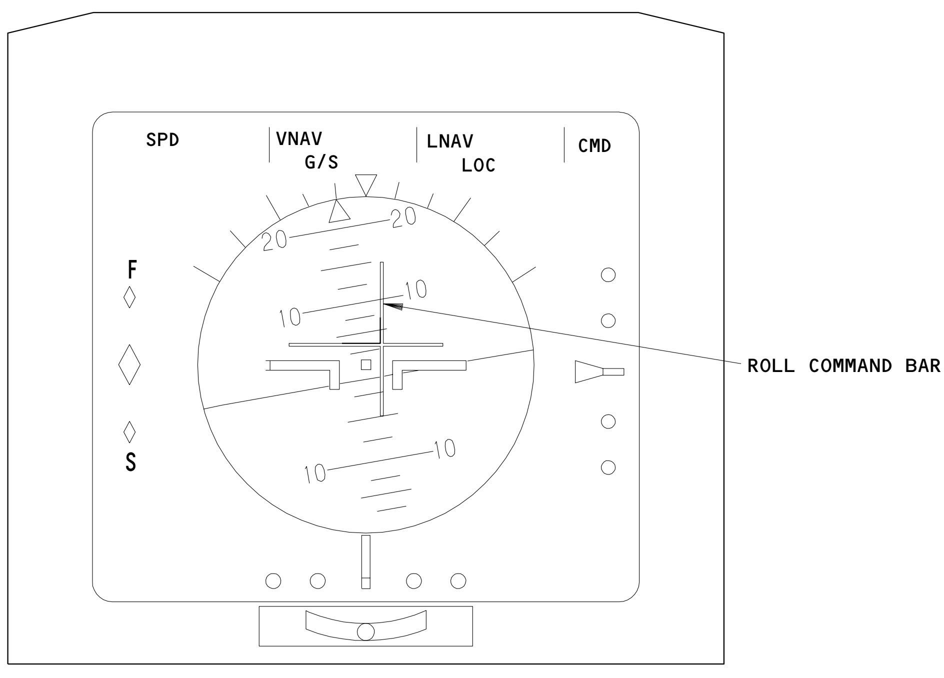
(a) The LNAV engage logic applies the horizontal steering command from the FMC to a summing point where it is summed with the F/D heading command in the development of the roll F/D commands. The LNAVIN logic applies the horizontal steering command to the roll attitude processing chain which generates the aileron servo command. Aileron servo position and surface position LVDTs provide feedback to the processor for nulling the servo command.

(14) Localizer (LOC) Mode (Fig. 18)

(a) The MCP controls and displays associated with the LOC mode are the HDG display, the LOC switch/light and the APP switch/light.



MODE CONTROL PANEL



EADI
(EXAMPLE)

Localizer Mode
Figure 18

(b) The AFDS acquires and maintains the front course LOC centerline during single or multichannel ILS approach. Localizer control laws generate AFDS commands for control of optimum flight path to the LOC centerline. At LOC capture, continuous tests are started to determine when transition from LOC capture to LOC track should occur. In LOC track, the AFDS commands for optimum tracking of the LOC centerline.

(c) The FMA on the EADI displays LOC in white as the roll armed mode before LOC capture and LOC in green as the roll engaged mode after LOC capture. At LOC capture, the heading display on the MCP changes from the last selected heading to the selected course (runway heading) on the ILS control panel.

(d) Single Channel LOC Approach

1) A single channel LOC approach is enabled by pressing the LOC switch. This mode is used when the pilot wishes to fly a different vertical path than that dictated by the glideslope beam, or where no G/S is available. The pitch axis may be controlled using any single channel vertical control mode.

2) The single channel LOC approach mode can be disengaged by: selecting heading hold, arming the approach mode and capturing G/S, sensing an on-ground condition, A/P disconnect, or switching the F/D off.

(e) Single Channel ILS Mode

1) Single channel ILS operation is not possible, since all channels are automatically armed when the APP switch/light is pressed.

(f) Multi-Channel ILS Approach (Autoland)

1) The multi-channel ILS mode is enabled by pressing the APP switch/light after one channel has been engaged in CMD. The other two channels engage in CMD after APP has been pressed and accepted. Multi-channel operation provides full LOC and G/S AUTOLAND control to CAT III B standards. The mode is disabled by A/P disconnect or by default to single channel A/P approach.

(15) Localizer Approach Geometry (Fig. 19)

(a) Controlling Factors

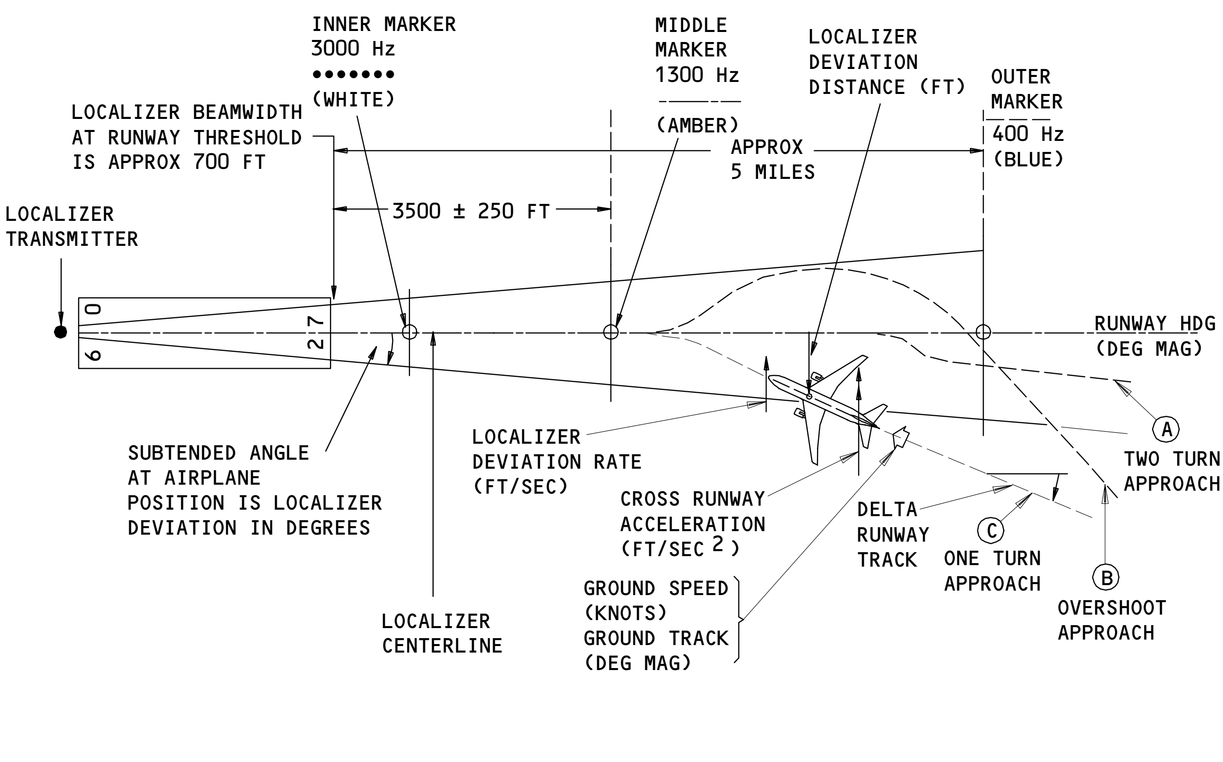
1) Localizer deviation distance (ft) is the horizontal distance from the localizer centerline to the airplane. Localizer deviation rate (ft/sec) is a function of the cross-localizer centerline velocity. Cross-runway acceleration is in ft/sec^2. The angle between ground track and runway heading is called delta runway track.

(b) Raw Data Inputs

1) Runway heading information comes from the ILS control panel. The Inertial Reference System (IRS) provides ground speed and ground track data.

(c) LOC Capture
1) Localizer capture occurs when LOC capture criteria are satisfied. Capture criteria threshold is a variable point depending on values of the controlling factors listed above. It is calculated so that a smooth capture is achieved with minimum control activity and without overshoot beyond beam threshold. Localizer capture is inhibited when delta runway track exceeds 120 degrees.

(d) Capture Flight Paths
1) If the delta runway track angle is so small that maintaining present track would cause intersecting the beam centerline too late or not at all, or if the angle is divergent up to 120 degrees, a two-turn capture (A) is commanded. A one-turn capture (C) is commanded if a sufficiently early beam center intersection is possible and a bank angle is computed to fly a circular flight path. One turn capture causes a beam center overshoot if the computed bank angle is greater than max bank angle allowed (B). The start of the capture turn on the flight path (C) is delayed until the minimum LOC capture bank angle is computed.



Localizer Approach Geometry
Figure 19

2) The capture turn is maintained at the commanded bank angle as long as possible, until computed rolldown rate to wings level is at the maximum permissible value. Control laws apply inertial smoothing to reduce effects of beam noise and propagation irregularities. Signal compensation is applied for gust filtering and crosswind shear effects.

(e) Localizer Tracking

1) Control laws are modified when LOCLIN (LOC linear) criteria are satisfied, when lateral position and deviation rate have primary control. The LOCLIN criteria is a function of cross runway acceleration and LOC processing signal levels.

(f) Control Law Objectives for LOC Operation

1) The objectives of localizer control laws are zero steady state error (eliminate LOC standoff error), max B/A commanded, 30 deg., max roll rate commanded, 7 deg/sec, maximize disturbance rejection, minimize control activity, and avoid unnecessary overshoots.

(16) Localizer Mode Logic (Fig. 12)

(a) LOC Mode Arm

1) The LOC mode arm logic is generated by pressing the LOC switch/light on the MCP, but is inhibited during the F/D disconnect pulse. This permits synchronization of the LOC signal and command processing so that the LOC integral and proportional roll commands are zero.

(b) LOC Mode Enable

1) The LOC enable logic is generated when LOC mode arm or APP arm and LOC deviation filter engage are a logic one. The LOC deviation filter engage is initially dependent on LOC in-range logic. Synchronization continues as in LOC mode arm. When runway correction enable is logic one, computation of filtered deviation distance is modified to be primarily controlled by cross runway acceleration.

(c) Two-Turn Capture (LOC2TC)

1) The following conditions determine the start of a two-turn capture. The threshold value of delta ground track (DGTRK) is the value below which LOC2TC becomes logic one (DPSIWN). It is the lesser of 20 degrees and a computed value which is a function of G/S attitude and filtered LOC deviation distance (DYLOCF). A divergent capture situation exists when the signs of DGTRK and DYLOCF are equal. When LOC2TC is logic one, LOC1TC is inhibited. Localizer capture (LOC CAPT) is generated by LOC 2 (LOC second phase).

(d) One-Turn Capture (LOC1TC)

1) When the computed B/A exceeds the minimum LOC capture turn B/A, LOC1TC logic is generated and LOC CAPT is generated. A latch maintains LOC CAPT and generates LOC CAPP (past value of LOC CAPT). Localizer signal and command synchronization is disabled to permit capture turns to be generated.

(e) LOC Linear
   1) After the airplane has become aligned to the LOC centerline, LOC LINEAR is generated and the LOC TRACK submode is enabled. Delta runway track must be less than 90 degrees but final control in developing LOC LINEAR is the output of the localizer deviation rate computation. This computation is sensitive to all critical variables for LOC centerline acquisition. The LOC LINEAR logic modifies the control laws for improved centerline tracking.
(f) LOC Engaged (LOC ENG)
   1) The LOC ENG logic is generated by LOC CAPT. At the transition to LOC LINEAR, LOC ENG momentarily goes to zero. This supresses transients to the LOC command processor during entry into LOC LINEAR and permits synchronizing for final capture.

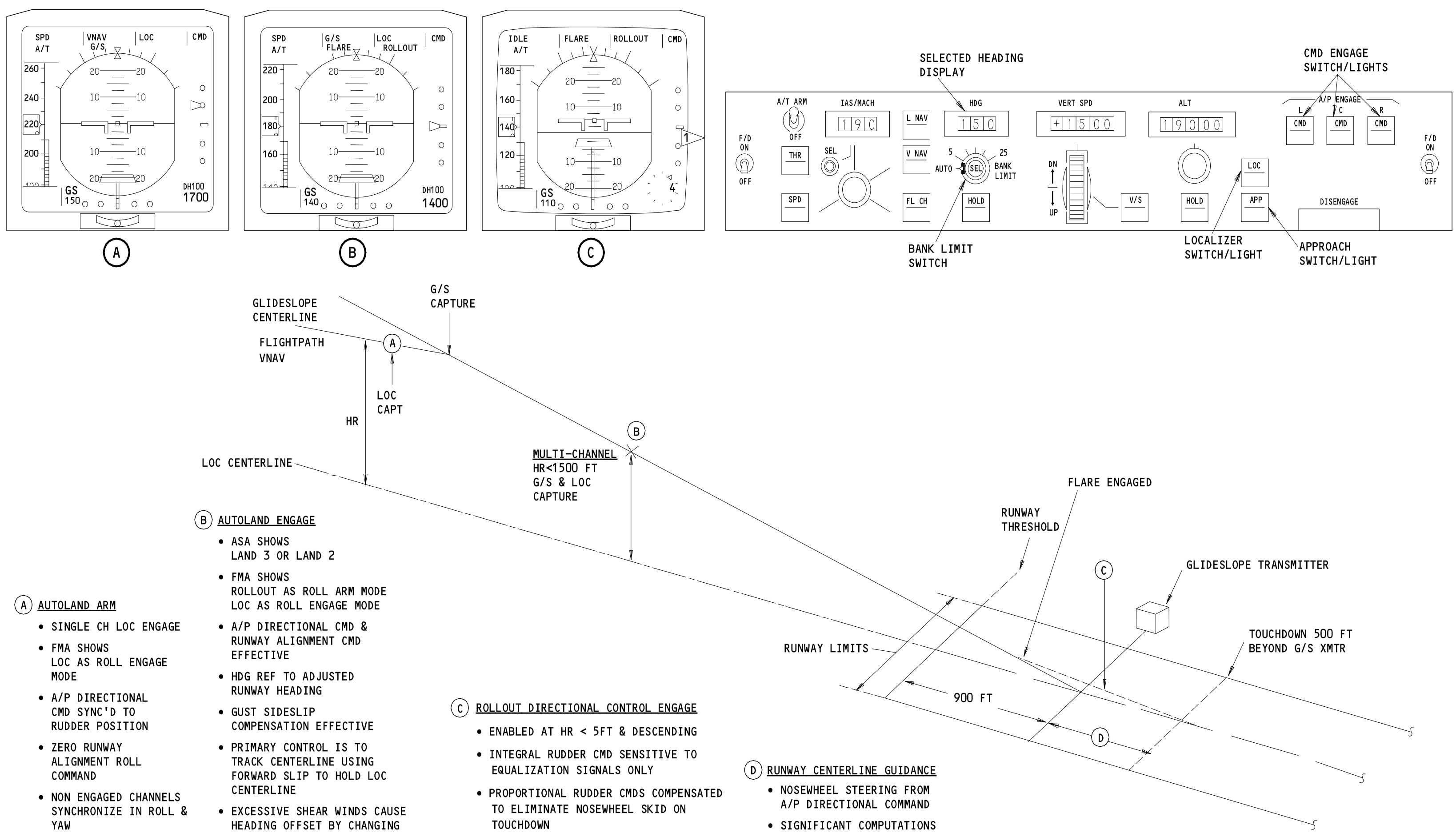
(17) Localizer Mode Control Signal Flow (Fig. 13)
(a) The localizer control laws produce commands to fly the necessary path to capture the beam center. The system initially controls ground track angle until an initial capture turn is commanded at the computed bank angle. Final capture of the beam centerline then starts using lateral position control. Overshoot and control activity is minimized by flying as much of the capture turn as possible at the computed bank angle. Compensation is provided for effects of turbulence, beam noise, and beam propagation irregularities.
(b) LOC Mode Arm
   1) The LOC command processor synchronizes to LOC displacement rate error (DYDOTE) and filtered LOC deviation distance (DYLOCF). The cross-runway integrator synchronizes to roll attitude (PHI). The DGTRK is multiplied by the sign of DYLOCF (±1) to generate the two turn bank angle command curve. The minimum bank angle is calculated as a function of complimentary filtered gear altitude. This sets the minimum value of the P2TURN limiter. The P2TURN signal is not used, however, if LOC2TC logic is zero. The P1TURN signal can only be computed when LOC ENG is logic one, but until LOC CAPT occurs, P1TURN is not used in processing.
(c) LOC One-Turn Capture
   1) The Sync loops are opened by LOC ENG and LOC CAPT. At LOC CAPP, the stored value of P1TURN (LOC Proportional roll command) is held through the signal latch. Subsequent changes of P1TURN have no effect. The P1TURN signal is applied through a ± 5.0 deg bank cmd/sec limiter and then converted into cross-runway acceleration. This develops cross runway velocity. This velocity is compared with LOC displacement rate (DYDOT) and LOC displacement rate error (DYDOTE) is produced. This error is converted to a proportional roll command and applied to a summing point. The other input to the summing point is zero, so the LOC proportional command is the only command generated at the initial capture turn.

(d) LOC Two-Turn Capture
1) When conditions cause LOC2TC to be logic 1, P2TURN is selected for processing. Since P2TURN has been continuously computed during LOC mode arm, LOC prop roll assumes the value of P2TURN at LOC CAPT and is maintained at that value during the turn towards the beam. System response downstream from this summing point is the same as during P1TURN operation. When LOC2TC conditions no longer exist due to increased DGTRK, P1TURN is selected for the second capture turn.

(e) LOC Linear Conditions
1) Final capture criteria is determined by the need to minimize overshoot and to avoid excessive rolldown rates. Continuous computation of lateral deviation and deviation rate occurs during the initial capture turn established by LOC proportional roll command. The LOCLIN logic is generated after LOC CAPT when LOC deviation rate is within the capture band in the LOC CMD processor filter frequency computation. The maximum rolldown rate command is P2MAX. When computations detect that rolldown rate equals P2MAX, the LOCLIN transition occurs and establishes filter values in the LOC CMD processor.
2) Localizer command integrator already has an output derived from cross runway acceleration command during the initial capture turn. This is now modified at the summing point and fed back to the integrator. This output is of opposite sign to the original input, so it causes a rundown of the integrator output. This establishes a maximum DYC at LOCLIN initiation.
3) The LOC displacement error (DYE) which reduces as DYLOCF feedback reduces, is converted into a proportional roll command and summed with LOC displacement rate error to modify PCLOC as a function of DYLOCF. The gain schedule generates a dependent LOC integral command which eliminates any standoff error.

(f) Final Capture of LOC Centerline
1) The final capture flight path follows an exponented law. When the computed rolldown command is less than P2MAX, a new value is generated to control the final rolldown command.

(18) Autoland Mode - Roll and Yaw (Fig. 20)
(a) The MCP controls and displays related to the Autoland Mode are identified on Fig. 20.



Ⓐ AUTOLAND ARM
- SINGLE CH LOC ENGAGE
- FMA SHOWS LOC AS ROLL ENGAGE MODE
- A/P DIRECTIONAL CMD SYNC'D TO RUDDER POSITION
- ZERO RUNWAY ALIGNMENT ROLL COMMAND
- NON ENGAGED CHANNELS SYNCHRONIZE IN ROLL & YAW
- ASA SHOWS AUTOLAND CAPABILITY LIMITATION
- MCP WINDOW SHOWS SELECTED RUNWAY HEADING, BUT GA EXIT HDG MAY BE PRESELECTED

Ⓑ AUTOLAND ENGAGE
- ASA SHOWS LAND 3 OR LAND 2
- FMA SHOWS ROLLOUT AS ROLL ARM MODE LOC AS ROLL ENGAGE MODE
- A/P DIRECTIONAL CMD & RUNWAY ALIGNMENT CMD EFFECTIVE
- HDG REF TO ADJUSTED RUNWAY HEADING
- GUST SIDESLIP COMPENSATION EFFECTIVE
- PRIMARY CONTROL IS TO TRACK CENTERLINE USING FORWARD SLIP TO HOLD LOC CENTERLINE
- EXCESSIVE SHEAR WINDS CAUSE HEADING OFFSET BY CHANGING HEADING REFERENCE
- RUDDER & AILERON EQUALIZATION SIGNALS EFFECTIVE
- CONTROL WITHIN RUNWAY LIMITS IN CROSSWINDS UP TO 25 KNOTS WITH BANK ANGLE LIMITED TO 2 DEG

Ⓒ ROLLOUT DIRECTIONAL CONTROL ENGAGE
- ENABLED AT HR < 5FT & DESCENDING
- INTEGRAL RUDDER CMD SENSITIVE TO EQUALIZATION SIGNALS ONLY
- PROPORTIONAL RUDDER CMDS COMPENSATED TO ELIMINATE NOSEWHEEL SKID ON TOUCHDOWN
- PRIMARY DATA FOR PROPORTIONAL RUDDER CMDS ARE DEVIATION DISTANCE & DEVIATION RATE
- RUNWAY ALIGNMENT COMMAND INEFFECTIVE
- FMA SHOWS ROLLOUT AS ROLL ENGAGE MODE

Ⓓ RUNWAY CENTERLINE GUIDANCE
- NOSEWHEEL STEERING FROM A/P DIRECTIONAL COMMAND
- SIGNIFICANT COMPUTATIONS PROGRAMMED BY GROUND SPEED TO OPTIMIZE CONTROL DOWN TO 25 FT/SEC (17 MPH)
- TERMINATES AT A/P DISCONNECT BEFORE RUNWAY TURNOFF
- NOSEWHEEL CMD TO RUDDER CMD RATIO IS APPROX 1:3

1 G/S POINTER REMOVED SINCE G/S DEV IS NCD AFTER PASSING G/S TRANSMITTER

Roll and Yaw Autoland Mode
Figure 20

586785

(b) The selected heading display shows the selected runway heading from the ILS control panel after LOC capture and maintains this display until the A/P is disconnected after landing or after the roll go-around mode is terminated. The B/A limit selector is usually left in the AUTO position during autoland. When pressed, the approach mode switch/light (APP) initiates the transition into the autoland configuration.

(c) Autoland Roll/Yaw Flight Path

1) Four conditions of rollout control are shown: rollout arm, rollout engage, rollout directional control engage, and runway centerline guidance.

(d) Rollout Arm (A)

1) Rollout arm is single channel in CMD with APP selected. The EADI displays LOC as the roll engaged mode. The A/P directional CMD is synchronized to rudder position. All non-engaged channels synchronize in roll and yaw.

(e) Rollout Engage (B)

1) All channels are CMDENG in roll, pitch and yaw. The EADI displays rollout white as the roll armed mode. The A/P directional CMD and runway alignment command are active. Heading reference is to adjusted runway heading. Primary control is to track centerline using forward slip to hold the LOC centerline.

(f) Rollout Directional Control Engage (C)

1) Rollout directional control engage is enabled when radio altitude is less than 5 feet and descending. The EADI displays rollout green as the roll engaged mode. The integral rudder command is sensitive to equalization signals only. Rudder proportional commands are compensated to eliminate nose wheel skid on touchdown and body yaw rate. Deviation distance and deviation rate serve as primary data for the rudder proportional commands.

(g) Runway Centerline Guidance (D)

1) Nose wheel steering is enabled after the nose wheel touches the runway. It maintains the airplane on the centerline of the runway. Significant computations are programmed by ground speed to optimize control down to 25 ft/sec (17 mph). Control terminates at A/P disconnect before runway turnoff.

(19) Autoland Mode Logic (Fig. 12)

(a) Upon receipt of the APP mode ARMED or OPERATIVE discrete, the APP switch dot-bar matrix is illuminated. Each CMD ENGAGE switch initiates a CMDENG request discrete which goes to the corresponding FCC. A valid request generates a REQUEST ACCEPTED discrete which illuminates the dot-bar matrix in the CMD switch/light. The APP switch generates a similar request when pressed. Each F/D switch generates an ON/OFF discrete but there is no visual feedback to the switches.

(b) Command Engage Sequence Detection Logic

1) Each FCC receives CMDENG REQUEST ACCEPTANCE logic from the other two FCCs via the cross-channel bus and records the order in which each become CMDENG. One of the discretes (L FIRST, C FIRST, R FIRST) is supplied to the first-in-command detection logic. This logic determines which FCC will supply FMA signals to the EFIS. It identifies LOCAL FIRST IN COMMAND and FOREIGN channels during multi-channel engage (MCHENG). On detection of a failure in the first-in-command channel, the fail sequence logic selects a different channel to be first-in-command.

(c) Autoland Processing Logic

1) The APP mode request generates the APPROACH ARM logic when the F/D disengage logic is zero. This is applied to the localizer logic as an alternate to the LOC mode arm logic (B on the illustration), and is also used in autoland logic.

2) When APP ARM, MCHENG, Runway alignment enable, and IN-AIR are all logic one, the generation of runway align engage logic is enabled. This logic goes high when roll attitude is less than 10 degrees, delta runway track is less than 2 degrees and the LOC LINEAR logic has been generated. Runway align enable logic enables the runway alignment command processor. This logic one is latched after 2 seconds to prevent loss of the signal due to out-of-tolerance inputs that would cause a zero logic from the preceeding AND gate. The runway align engage logic also produces the delta runway engage and delta runway path logic.

3) When radio height is less than five feet, Rollout logic is produced and runway alignment logic goes to zero. The delta runway engage and delta runway path logic remain logic ones.

(20) Autoland Mode Control Signal Flow (Fig. 13)

(a) Rollout Arm Conditions

1) Delta heading runway is synchronized to adjusted heading and is zero. Delta LOC deviation distance is synchronized to filtered LOC deviation distance and is also zero. The rudder servo command is synchronized to zero to maintain the rudder crossfeed command at zero so that the runway alignment roll command will be held at zero.

(b) Rollout Engage

1) At multi-channel engage, command equalization must occur to ensure that all FCCs deliver identical value commands to the flight control systems. All FCCs are synchronizing to rudder position.

2) The first channel in command engages into AUTOLAND CMD as soon as any other channel has a command request accepted. Any difference between servo actuator position and rudder position is the rudder equilization command (normally very small) and is routed to the max/min selectors. The max/min feature prevents excessive deflection from the rudder's present position.

3) The multi-channel engage logic one enables the generation of runway align engage when roll angle and delta ground track are less than 10 degrees and 2 degrees, respectively, and the LOC LINEAR logic is one.

(c) Rollout Engage

1) Rollout directional control engage logic is generated when radio height is five feet. This maintains delta runway engage and delta runway path at logic one while runway align engage goes to zero. Delta LOC deviation distance, which was synchronized to zero, is now increased, at a rate limited to 2 ft/sec, to the current value of filtered LOC deviation distance. This is converted into deg/sec and applied to the rudder integral command signal chain.

2) When rollout directional control engage becomes logic one, additional feedback is added. Body yaw rate is summed with pseudo body rate computed from ground speed and rudder position, and multiplied by a linear constant. This eliminates nose wheel skid on touchdown and thus ensures effective nose wheel steering as soon as the nose wheel touches the ground. As ground speed reduces, rudder control becomes less effective and nose wheel steering provides primary rollout direction control.

3) When ground speed reduces to below 25 ft/sec, inverse gain schedules cause increased command response as rollout speed reduces thus improving LOC centerline tracking.

(d) Rollout Command Processing

1) Runway alignment command in degrees of roll is summed with LOC proportional command. The result is applied to the roll flight director. This signal is also differentiated and added to the LOC integral command. This combined signal is the roll attitude command integrator input in the ROLLOUT engage mode.

2) Aileron or wheel position and LCCA servo position are summed to generate the roll equalization signal. This signal is limited to ±2 degrees and converted into deg/sec roll command. At MCHENG the equalization roll rate signal is added to the roll command signal. This signal modifies the response of an FCC coming online at MCHENG to compensate for electrical and mechanical hysteresis which results in small variations of LVDT outputs for the same surface position.

3) At MCHENG ±40 degrees becomes the reference for max/min select limiters. The aileron servo command is also limited to ±40 degrees. These larger control limits are required at lower speeds during approach when rapid airplane response is necessary.

(21) Roll Go-Around Mode (Fig. 21)

(a) Significant MCP Features

1) The MCP may show any of the following switch dot-bar matrices illuminated before entry into the go-around mode: HDG HOLD, LNAV, LOC, APP, CMD. The F/D switches may be ON or OFF. The HDG display shows selected runway heading before G/A and MCP selected heading during G/A.

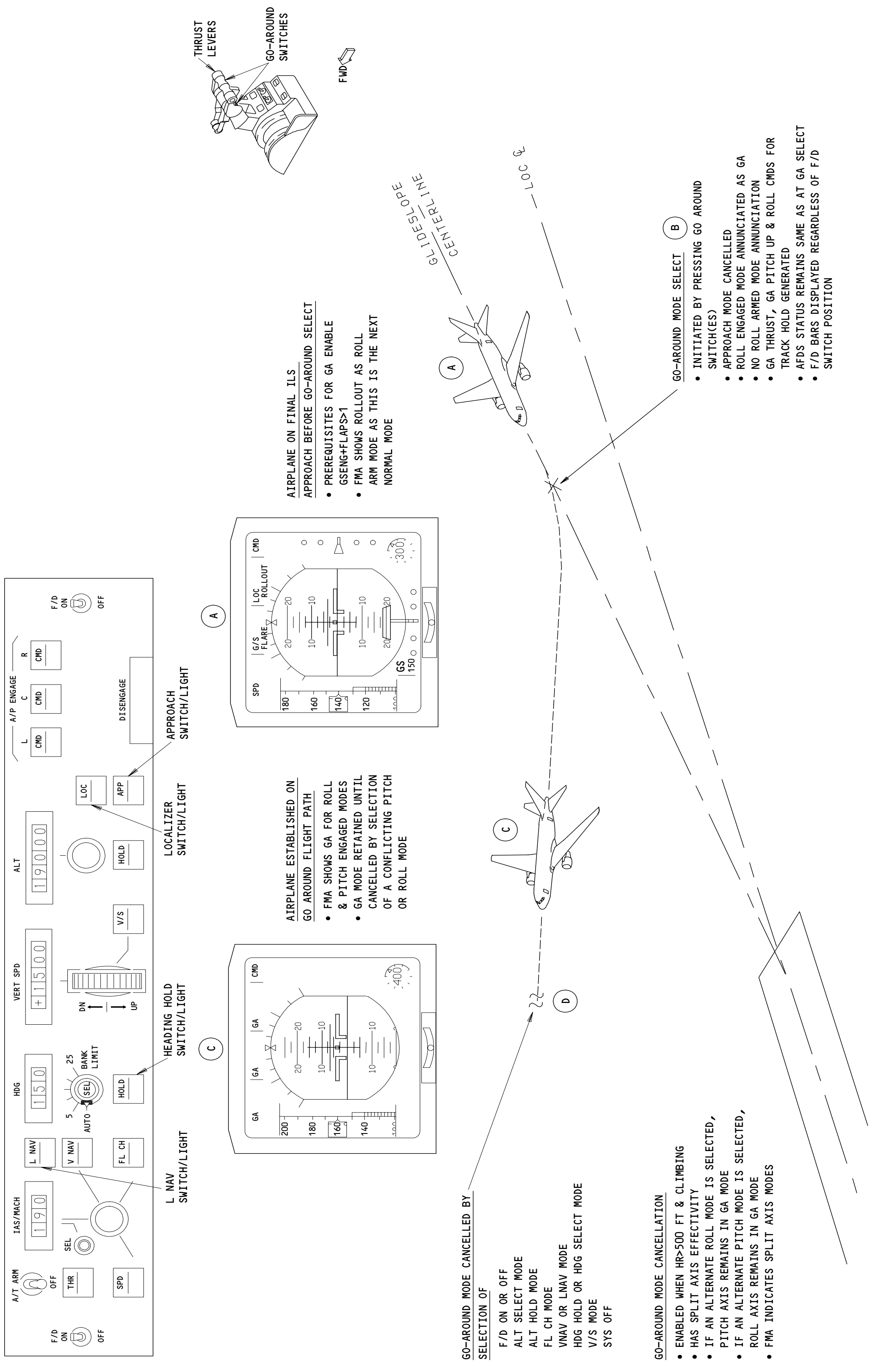
(b) Roll Go-Around Flight Path

1) The sequence of events if G/A is selected during an ILS approach is shown. Position A indicates pre-go-around conditions. The airplane is on final approach; G/S and flare are engaged; flaps less than 20 degrees. The EADI displays rollout white as the roll armed mode since that would normally be the next mode.

2) Position B shows the G/A select conditions. Go-around is initiated by pressing the G/A switch on the thrust levers. The ILS guidance and annunciation are discontinued; G/A is shown as the roll engaged mode (with no roll armed mode annunciation); and G/A thrust, pitch-up commands and roll commands are generated. The AFDS status remains the same as before G/A thrust. The flight director is displayed on the EADI during G/A, regardless of F/D switch position.

3) Position C indicates the G/A climbout. The EADI displays G/A green as the roll and pitch engaged modes. The G/A mode will be retained until cancelled by the selection of a conflicting pitch or roll mode.

4) Position D shows the G/A cancellation conditions. Cancellation is enabled when radio altitude is greater than 400 feet and the airplane is climbing. This produces a split axis condition. If an alternate roll mode is selected, the pitch axis remains in G/A mode and vice versa. The EADI FMA indicates split axis modes. Go-around can be cancelled by selection of F/D ON or OFF; altitude select or altitude hold; FLCH; VNAV or LNAV; HDG hold or select; V/S; or system OFF.



Roll-Go Around Mode
Figure 21

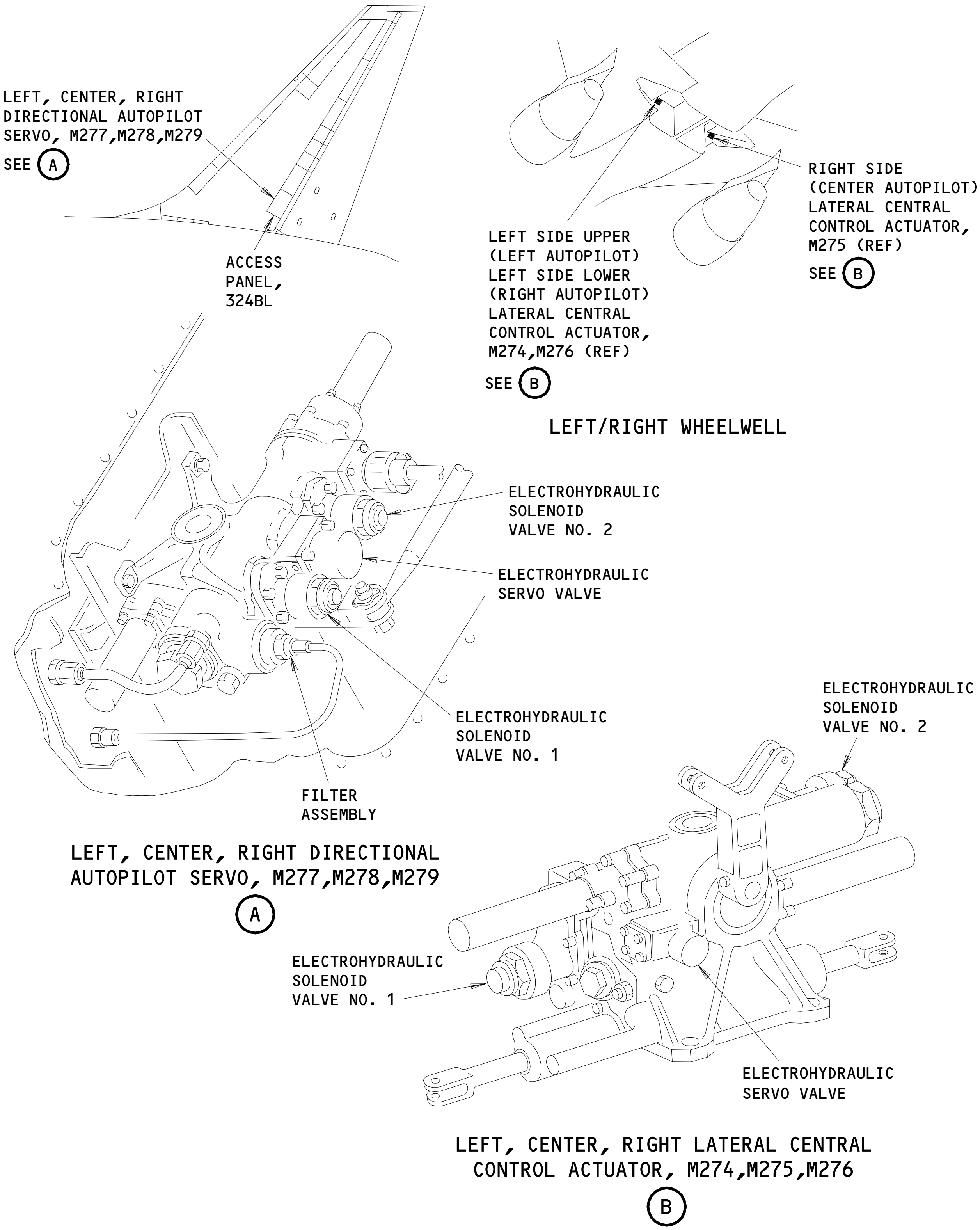
(22) Go-Around Mode Logic
   (a) Go-around engage logic is generated by pressing the go-around switch on the thrust levers. Go-around engage logic generates track mode select logic and go-around runway logic. After a one-frame delay, heading hold is generated. If the A/P is still CMD ENG, heading/track control and heading proportional command/track mode reference are generated.

(23) Go-Around Mode Control Signal Flow
   (a) At go-around engage, the heading reference is synchronized to the adjusted runway heading which is compensated for drift angle. The delta heading command from the A/P turn direction control is supplied to the roll F/D command processor. It is also used to generate the delta roll command. This is further processed to generate delta wheel command to the ALCS.
   (b) Track mode select logic selects adjusted track as input to the F/D turn direction control. The F/D heading command changes slightly to generate roll commands reflecting this change of reference from heading to track. The A/P turn direction control also senses this change in reference. The delta heading command then becomes runway heading minus adjusted track in degrees roll. This output is then processed and waits until heading track control logic is generated one frame later.
   (c) After heading track control is generated, the delta heading command from the A/P turn direction control generates the heading proportional command and roll commands to hold ground track.

AUTOPILOT/FLIGHT DIRECTOR ROLL AND YAW CHANNEL

COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
ACTUATOR - (FIM 27-11-00/101)				
LEFT SIDE LOWER (R AUTOPILOT) LATERAL CENTRAL CONTROL, M276				
LEFT SIDE UPPER (L AUTOPILOT) LATERAL CENTRAL CONTROL, M274				
RIGHT SIDE (C AUTOPILOT) LATERAL CENTRAL CONTROL, M275				
FILTER -		1	VERT STAB, 324BL EA DIRECTIONAL AUTOPILOT SERVO	22-13-02
SERVO - C DIRECTIONAL AUTOPILOT, M278		1	VERT STAB, 324BL	22-13-01
SERVO - L DIRECTIONAL AUTOPILOT, M277		1	VERT STAB, 324BL	22-13-01
SERVO - R DIRECTIONAL AUTOPILOT, M279		1	VERT STAB, 324BL	22-13-01
VALVE - LATERAL CENTRAL CONTROL ACTUATOR				
ELECTRO-HYDRAULIC SERVO		1	L/R WHEEL WELL EA LATERAL CENTRAL CONTROL ACTUATOR	22-13-03
ELECTRO-HYDRAULIC SOLENOID		2	L/R WHEEL WELL EA LATERAL CENTRAL CONTROL ACTUATOR	22-13-03
VALVE - DIRECTIONAL AUTOPILOT SERVO				
ELECTRO-HYDRAULIC SERVO		1	VERT STAB, 324BL EA DIRECTIONAL AUTOPILOT SERVO	22-13-02
ELECTRO-HYDRAULIC SOLENOID		2	VERT STAB, 324BL EA DIRECTIONAL AUTOPILOT SERVO	

Autopilot/Flight Director Roll and Yaw Channel - Component Index
Figure 101



Autopilot/Flight Director Roll and Yaw Channel - Component Location
Figure 102

DIRECTIONAL AUTOPILOT SERVO - REMOVAL/INSTALLATION

1. General
   A. Three Directional Autopilot Servos (DASs) are adjacent to each other at the bottom of the vertical stabilizer. They are immediately in front of the rudder hinge line. Each servo has an output rod connected to the rudder feel, centering, and trim mechanism.

TASK 22-13-01-004-001

2. Remove the Directional Autopilot Servo (Fig. 401)
   A. Equipment
      (1) Rig pin R4, P/N A20004-15, part of set A20004-XX (AMM 20-10-24/201).
   B. References
      (1) AMM 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
      (2) AMM 20-10-24/201, Rig Pins
      (3) AMM 24-22-00/201, Electrical Power - Control
      (4) AMM 29-11-00/201, Main (Left, Center, and Right) Hydraulic System
   C. Access
      (1) Location Zone
          Vertical Stabilizer Rear Spar to Trailing Edge

      (2) Access Panel
          324BL, Rudder L.E. Spar

   D. Prepare for Removal

      S 864-002
      (1) Supply electrical power (AMM 24-22-00/201).

      S 864-003
      (2) Remove pressure from the left, center and right hydraulic systems (AMM 29-11-00/201).

      S 864-004
      (3) Set the L, C, and R TAIL FLT CONTROL SHUTOFF switches on the P61 panel to OFF.

S 864-005
(4) Set the two STAB TRIM switches on the control stand panel, P10, to CUTOUT.

S 864-006
(5) Set the two F/D switches on the MCP to OFF.

S 864-007
(6) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
   (a) 11C12, STAB TRIM SHUTOFF L
   (b) 11C13, STAB TRIM SHUTOFF CENTER
   (c) 11E16, MODE CONT PNL L
   (d) 11E34, MODE CONT PNL R
   (e) 11H17, FLT CONT SHUTOFF TAIL L
   (f) 11H18, FLT CONT SHUTOFF TAIL CENTER
   (g) 11H27, FLT CONT SHUTOFF TAIL R

S 864-008
(7) Open these circuit breakers on the overhead circuit breaker panel, P11, for the applicable (left, center, or right) DAS and attach DO-NOT-CLOSE tags:
   (a) Left DAS:
      1) 11E17, FLT CONT CMPTR PWR L
      2) 11E18, FLT CONT CMPTR SERVO L
   (b) Center DAS:
      1) 11E20, FLT CONT CMPTR PWR C
      2) 11E21, FLT CONT CMPTR SERVO C
   (c) Right DAS:
      1) 11E35, FLT CONT CMPTR PWR R
      2) 11E36, FLT CONT CMPTR SERVO R

S 014-009
(8) Remove the access panel, 324BL, at the bottom of the vertical fin (left side) to access the servos (AMM 06-42-00/201).

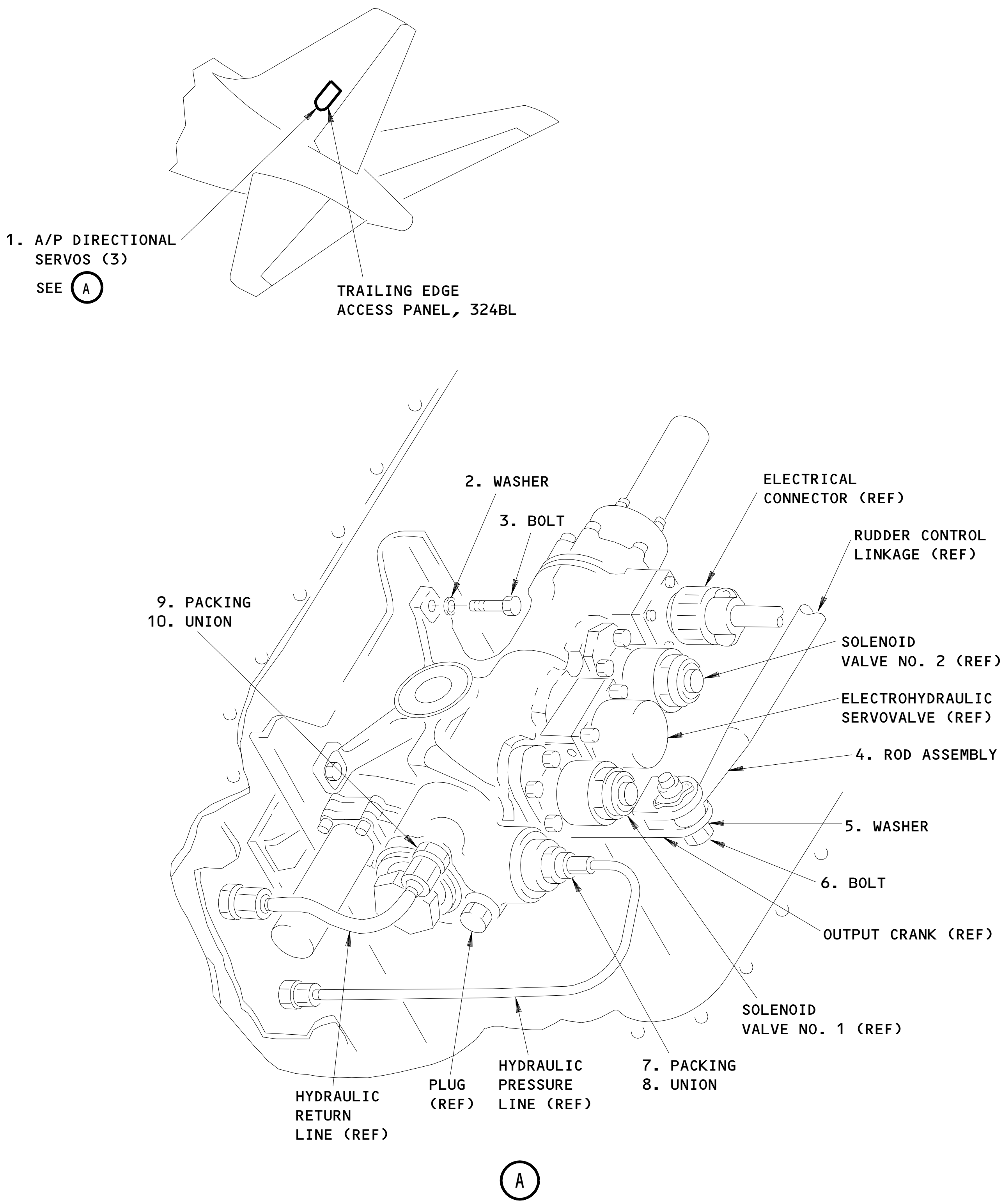
S 494-010
(9) Install the rig pin R4 in the Feel Centering and Trim mechanism.

E. Remove the Servo

S 034-011
(1) Disconnect the electrical connector from the DAS.

S 034-012
(2) Remove the bolt (item 6) and washer (item 5) to disconnect the rod assembly (item 4) from the servo.

S 034-013
(3) Disconnect the hydraulic lines from the servo.



Directional Autopilot Servo
Figure 401

S 394-014
(4) Seal the hydraulic lines and servo hydraulic ports with caps.

S 024-015
(5) Hold the servo in position and remove the bolts (item 3) and washers (item 2).

S 394-016
(6) Remove the servo from the vertical stabilizer.

TASK 22-13-01-404-017
3. Install the Directional Autopilot Servo
A. General
(1) Two adjustment procedures are given to adjust the Directional Autopilot Servo after installation.
(a) Adjustment with the Maintenance Control Display Panel (MCDP).
(b) Adjustment with the Phase Synchronous Voltmeter Model 101A.
B. Equipment
(1) Nose Gear Steering Valve Lockpin - A09003-1
(2) Rig pin R4, P/N A20004-15, part of set A20004-XX (AMM 20-10-24/201).
(3) Voltmeter, Phase Synchronous Model 101A - 101-AC6 or 101-AC7 kit, (Necessary when the MCDP is not used for adjustment) Electronic Aviation Systems, PO Box 9213, 10423 W. Douglas, Wichita, KS 67277
(4) Adapter Cable Assembly 1014002-11
C. Consumable Materials
(1) A00247 Sealant, BMS 5-95
D. Parts

AMM		NOMENCLATURE	AIPC		
FIG	ITEM		SUBJECT	FIG	ITEM
401	1	Servo	22-13-01	01	5
	2	Washer		01	8
	3	Bolt		01	7
	4	Rod Assembly	27-21-10	01	60
	5	Washer		01	20
	6	Bolt		01	10
	7	Packing	22-13-01	01	107
	8	Union		01	108
	9	Packing		01	115,135
	10	Union		01	125

E. References
  (1) AMM 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
  (2) AMM 09-11-00/201, Towing
  (3) AMM 20-10-21/601, Electrical Bonding
  (4) AMM 20-10-22/701, Metal Surfaces
  (5) AMM 20-10-24/201, Rig Pins
  (6) AMM 22-00-02/201, Autoflight BITE
  (7) AMM 24-22-00/201, Electrical Power - Control
  (8) AMM 29-11-00/201, Main (Left, Center, and Right) Hydraulic System
  (9) AMM 51-31-01/201, Seals and Sealing

F. Access
  (1) Location Zone
      Vertical Stabilizer, Rear spar to Trailing Edge

  (2) Access Panel
      324BL, Rudder L.E. Spar

G. Install the Servo

S 414-018
  (1) Install the rig pin R4 in the Feel Centering and Trim mechanism if it is not installed (AMM 20-10-24/201).

S 124-022
  (2) Clean the servo surfaces and the servo support surfaces where they attach to each other. Spot clean with rotary bonding brush or rotary abrasive disk (AMM 20-10-22/701).

S 424-024
  (3) Put the servo in position on the holes for the bolts.

S 424-019
  (4) Put the bolts (3) and washers (2) in the holes and tighten them by hand.

S 014-023
  (5) Remove the caps from the hydraulic lines.

S 414-025
(6) Connect the hydraulic lines and tighten.

S 434-026
(7) Tighten the bolts (item 3) that hold the servo.

S 764-027
(8) Use a bonding meter to make sure the maximum resistance between the servo body and the airframe is 3.5 milliohms (0.0035 ohms) (AMM 20-10-21/601).

S 394-028
(9) With the BMS 5-95 sealant put a fillet seal around the servo assembly where the servo touches the support surface (AMM 51-31-01/201).

S 624-020
(10) With the 5-95 sealant put a fillet seal on the bolt (item 3) heads (AMM 51-31-01/201).

S 434-029
(11) Connect the electrical connector to the servo.

S 094-030
(12) Remove the rig pin R4.

S 864-031
(13) Turn the Rudder Trim switch counterclockwise (CCW) to position the rudder at 3 to 5 units of left trim. This energizes the trim actuator.

S 984-075
(14) Manually turn the aft quadrant clockwise (CW) until you can install the rig pin R4.

NOTE: This removes backlash in the rig pin. Rig pin R4 must stay installed for the Directional Autopilot Servo Adjustment.

S 824-032
(15) If replacement of the control rod assembly (item 4) is not necessary, adjust the servo with the Maintenance Control Display Panel (MCDP) or the Model 101A Phase Synchronous Voltmeter.

H. Replacement of the Control Rod Assembly

S 494-033
(1) Make sure the rig pin R4 is installed in the Feel Centering and Trim mechanism.

S 034-034
(2) Remove the bolt (item 6) and washer (item 5) to disconnect the rod assembly (item 4) from the servo.

S 024-035
(3) Remove the bolt, nut, and washer to disconnect the rod assembly (4) from the Feel Centering and Trim Mechanism.

S 824-036
(4) Make sure that the nominal length of the replacement control rod assembly (4) is 20 5/32 inches from bolt-center to bolt-center.

S 424-037
(5) Install the rod end into the Feel Centering and Trim mechanism with the bolt, nut, and washer.

S 824-021
(6) Adjust the servo with the Maintenance Control Display Panel (MCDP) or the Model 101A Phase Synchronous Voltmeter.

I. Adjust the Servo with the Maintenance Control Display Panel (MCDP)

S 864-038
(1) Supply electrical power (AMM 24-22-00/201).

S 864-039
(2) Remove DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11E16, MODE CONT PNL L
(d) 11E17, FLT CONT CMPTR PWR L
(e) 11E18, FLT CONT CMPTR SERVO L
(f) 11E20, FLT CONT CMPTR PWR C
(g) 11E21, FLT CONT CMPTR SERVO C
(h) 11E34, MODE CONT PNL R
(i) 11E35, FLT CONT CMPTR PWR R
(j) 11E36, FLT CONT CMPTR SERVO R
(k) 11H17, FLT CONT SHUTOFF TAIL L
(l) 11H18, FLT CONT SHUTOFF TAIL CENTER
(m) 11H27, FLT CONT SHUTOFF TAIL R

S 744-042
(3) With the hydraulic pressure off and the rig pin R4 installed start MCDP Ground Test 66-XDCR OUTPUTS (AMM 22-00-02/201).

S 744-040
(4) Push the YES/ADV switch until the top line of the MCDP display shows 66 RUD SURF DEG.

S 824-043
(5) Adjust the rod assembly (item 4) until it is between 00.0 ±0.6 degrees with the control rod in its installed position on the servo.
(a) Control rod adjustment:

Coarse: One-half turn of the rod end bearing is equivalent to .26 degrees.

Fine: One-half turn of the rod barrel is equivalent to .05 degrees.

S 424-044
(6) Connect the rod assembly (item 4) to the servo output crank with the bolt (item 6) and washer (item 5).

S 714-076
(7) Do the Test the Servo Installation steps.

J. Adjust the Servo with the Model 101A Phase Synchronous Voltmeter

S 484-045
(1) Connect Phase Synchronous voltmeter to the power source specified on voltmeter placard.

S 864-046
(2) Set the ON/OFF (S1) switch to ON.

S 484-041
(3) Stop for 10 minutes to let the voltmeter prepare for operation.

S 864-047
(4) Set the CAL/RUN (S2) switch to the CAL CHK position.
(a) Make sure the IN-PHASE VOLTS are 9.990 to 10.010.

S 864-048
(5) Set the CAL/RUN (S2) switch to the RUN position.

S 034-049
(6) Disconnect the electrical connector from the servo.

S 484-050
(7) Connect the -11 cable assembly to the Autopilot Pitch Control Servo and the Phase Synchronous Voltmeter.

S 824-051
(8) With the rod assembly (item 4) connected to the servo, adjust the Autopilot Servo assembly rod until the Phase Synchronous voltmeter displays 0 ±.05 IN-PHASE VOLTS.

S 434-052
(9) Connect the rod assembly (item 4) to the servo output crank with the bolt (item 6) and washer (item 5).

S 434-077
(10) Tighten the rod assembly (4) jamnut.

S 084-053
(11) Disconnect the Phase Synchronous Voltmeter as follows:
(a) Set the ON/OFF switch to the OFF position.
(b) Disconnect the -11 cable assembly from the autopilot servo.
(c) Disconnect the voltmeter from the power source.

S 434-054
(12) Connect the airplane electrical connector to the autopilot servo.

S 714-055
(13) Do the Test Servo Installation steps

K. Test the Servo Installation

S 864-079
(1) Supply electrical power (AMM 24-22-00/201).

S 864-080
(2) Remove DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11E16, MODE CONT PNL L
(d) 11E17, FLT CONT CMPTR PWR L
(e) 11E18, FLT CONT CMPTR SERVO L
(f) 11E20, FLT CONT CMPTR PWR C
(g) 11E21, FLT CONT CMPTR SERVO C
(h) 11E34, MODE CONT PNL R
(i) 11E35, FLT CONT CMPTR PWR R
(j) 11E36, FLT CONT CMPTR SERVO R
(k) 11H17, FLT CONT SHUTOFF TAIL L
(l) 11H18, FLT CONT SHUTOFF TAIL CENTER
(m) 11H27, FLT CONT SHUTOFF TAIL R

S 084-056
(3) Remove the rig pin R4.

S 864-057

WARNING: LOCK THE NOSE GEAR STEERING WHEN ANY MOVEMENT OF THE RUDDER IS SCHEDULED. THIS WILL PREVENT INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(4) Move the towing lever on the nose gear metering valve module to the towing position.

S 434-058
(5) Install the nose gear steering valve lockpin (AMM 09-11-00/201).

S 864-059
(6) Set the L, C, and R TAIL FLT CONTROL SHUTOFF switches on the P61 panel to ON.

S 864-072
(7) Set the two STAB TRIM switches on the P10 panel to NORMAL.

S 864-078

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(8) Pressurize the left, center, and right hydraulic systems (AMM 29-11-00/201).

S 944-060
(9) Manually operate the rudder system three times.

S 744-073
(10) Start MCDP Ground Test 66-XDCR OUTPUTS, (AMM 22-00-02/201).

S 744-061
(11) Push the YES/ADV switch until the MCDP top line message shows 66 RUD SURF DEG.

S 864-062
(12) With the rudder pedals released make sure that the MCDP bottom line values are 0 ±0.2 degree.

S 984-063
(13) Push the left or right rudder pedal through full travel.

S 754-064
(14) Make sure the value on the bottom line of the MCDP is more than ±10 degrees for L, C, and R servos.

S 714-065
(15) Do MCDP Ground Test 09-SERVO RUD (AMM 22-00-02/201).
(a) Make sure no fault messages show during the test.

S 214-066
(16) Examine the installation for hydraulic leaks and correct as necessary.

L. Put the Airplane Back to Its Usual Condition

S 414-067
(1) Close the access panel 324BL.

S 864-068

WARNING: STAY AWAY FROM THE NOSE GEAR WHEELS WHEN YOU REMOVE THE LOCKPIN. WHEELS CAN MOVE TO THE CENTER POSITION QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(2) Make sure the nose gear wheels are in the center position.

S 084-069
(3) Remove the nose gear steering valve lockpin.

S 864-074
(4) Set the MCDP to off.

S 864-070
(5) Remove the hydraulic power if it is not necessary (AMM 29-11-00/201).

S 864-071
(6) Remove the electrical power if it is not necessary (AMM 24-22-00/201).

DAS ELECTROHYDRAULIC SERVOVALVE AND SOLENOID VALVES - REMOVAL/INSTALLATION

1. General
   A. The three Directional Autopilot Servos (DASs) are located at the base of the vertical stabilizer, immediately forward of the rudder hinge line. Each servo unit has one electrohydraulic servovalve (EHSV) and two solenoid valves. The servos can be accessed through panel 324BL on the lower left side of the vertical stabilizer (Ref 06-42-00).

TASK 22-13-02-004-085

2. Remove the DAS Electrohydraulic Servovalve or Solenoid Valve (Fig. 401)
   A. References
      (1) 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
      (2) 24-22-00/201, Electrical Power - Control
      (3) 29-11-00/201, (Left, Center, and Right) Main Hydraulic Systems
   B. Access
      (1) Location Zone
          324 Vertical Stabilizer, Rear Spar to Trailing Edge

      (2) Access Panel
          324BL Rudder L.E. Spar

   C. Prepare for Removal

      S 864-044
      (1) Supply electrical power (Ref 24-22-00).

      S 864-045
      (2) Remove the pressure from the applicable hydraulic system (L, C, or R) (Ref 29-11-00).
          (a) Left DAS - left hydraulic system
          (b) Center DAS - center hydraulic system
          (c) Right DAS - right hydraulic system

      S 864-046
      (3) Set the L, C, and R TAIL FLT CONTROL SHUTOFF switches on the right side panel, P61 to OFF.

      S 864-047
      (4) Set the two STAB TRIM switches on control stand panel, P10, to CUTOUT.

      S 864-048
      (5) Set the two F/D switches on the MCP to OFF.

S 864-049

(6) Open these circuit breakers on the overhead circuit breaker panel, P11, DO-NOT-CLOSE tags:
   (a) 11C12, STAB TRIM SHUTOFF L
   (b) 11C13, STAB TRIM SHUTOFF CENTER
   (c) 11E16, MODE CONT PNL L
   (d) 11E34, MODE CONT PNL R
   (e) 11H17, FLT CONT SHUTOFF TAIL L
   (f) 11H18, FLT CONT SHUTOFF TAIL CENTER
   (g) 11H27, FLT CONT SHUTOFF TAIL R

S 864-050

(7) Open these circuit breakers on the P11 panel for the applicable (left, center, or right) DAS and attach DO-NOT-CLOSE tags:
   (a) Left DAS
      1) 11E17, FLT CONT CMPTR PWR L
      2) 11E18, FLT CONT CMPTR SERVO L
   (b) Center DAS
      1) 11E20, FLT CONT CMPTR PWR C
      2) 11E21, FLT CONT CMPTR SERVO C
   (c) Right DAS
      1) 11E35, FLT CONT CMPTR PWR R
      2) 11E36, FLT CONT CMPTR SERVO R

S 864-051

(8) Remove the access panel 324BL at the base of the vertical fin (left side) for access to the rudder control servos (Ref 06-42-00).

D. Remove the Valve

S 144-052

CAUTION: DO NOT LET CONTAMINATION GET INTO THE DIRECTIONAL AUTOPILOT SERVO AND VALVES. THE SERVO AND VALVES ARE VERY SENSITIVE ELECTROHYDRAULIC DEVICES. CONTAMINATION IN THE SERVO AND VALVES CAN CAUSE DAMAGE TO THE EQUIPMENT.

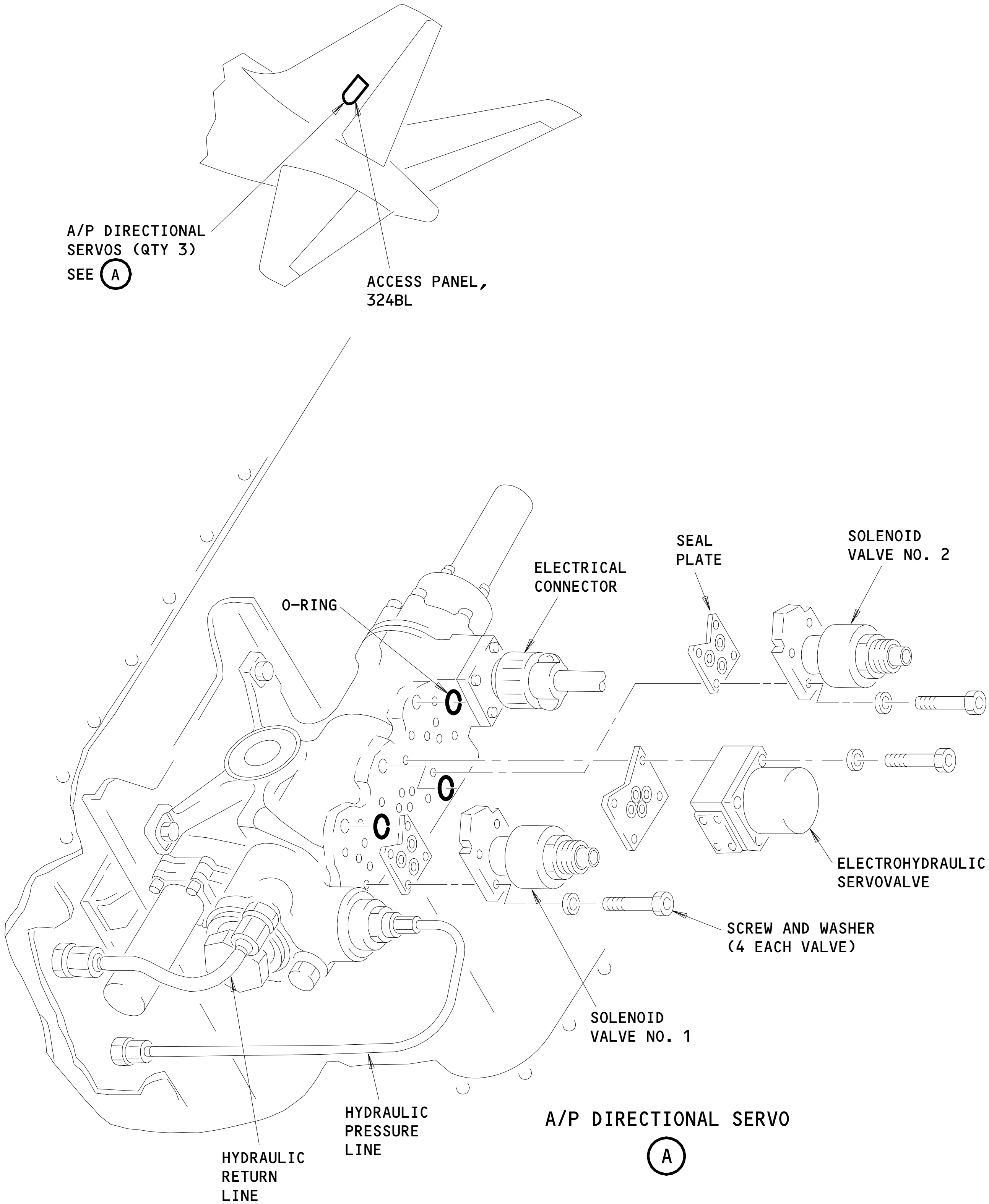
(1) Carefully clean the external areas of the valve and around the valve mounting surface.

S 034-053

(2) Remove the lockwire.

S 024-054

(3) Remove the mounting screws, the valve with O-ring, and the seal plate from the servo assembly.



DAS Electrohydraulic Servovalve and Solenoid Valves - Installation
Figure 401

S 434-055
(4) Put covers on the hydraulic and electrical servo openings to make sure contamination does not get into them.

TASK 22-13-02-404-056

3. <u>Install the DAS Electrohydraulic Servovalve or Solenoid Valve</u> (Fig. 401)

A. Equipment
   (1) Nose Gear Steering Valve Lockpin - A09003-1

B. Consumable Materials
   (1) D00148 Hydraulic Fluid, Fire Resistant, BMS 3-11
   (2) D00209 Lubricant - MCS 352B

C. References
   (1) 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
   (2) 09-11-00/201, Towing
   (3) 20-10-21/601, Electrical Bonding
   (4) 22-00-02/201, Autoflight BITE
   (5) 24-22-00/201, Electrical Power - Control
   (6) 29-11-00/201, (Left, Center, and Right) Main Hydraulic System

D. Access
   (1) Location Zone
       324 Vertical Stabilizer, Rear Spar to Trailing Edge

   (2) Access Panel
       324BL Rudder L.E. Spar

E. Install the Valve

S 034-057
(1) Remove the covers from the openings.

S 144-058
(2) Clean the mating surfaces of the valve and the servo assembly with clean hydraulic fluid. Do not let the hydraulic fluid to go into the electrical connector.

S 644-059
(3) Apply a small layer of MCS 352B lubricant to the new seal plate and the O-ring.

S 434-060
(4) Install the new O-ring on the valve.

S 424-061
(5) Install the seal plate and the valve with the screws and washers.
(a) Tighten the screws to 60 pound-inches.

S 764-062
(6) Use a bonding meter to make sure the resistance between the valve assembly and the servo body is 3.5 milliohms (0.0035 ohms) or less (Ref 20-10-21).

S 434-064
(7) Put a lockwire on the mounting screws.

F. Test the DAS Electrohydraulic Servovalve and Solenoid Valves

S 494-065

CAUTION: LOCK THE NOSE GEAR STEERING WHEN RUDDER MOVEMENT IS SCHEDULED. THIS MUST BE DONE TO PREVENT INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Move the towing lever on the nose gear metering valve module to the towing position.

S 494-066
(2) Install the nose gear steering valve lockpin (Ref 09-11-00).

S 864-067
(3) Supply electrical power (Ref 24-22-00).

S 094-068
(4) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11E16, MODE CONT PNL L
(d) 11E17, FLT CONT CMPTR PWR L
(e) 11E18, FLT CONT CMPTR SERVO L
(f) 11E20, FLT CONT CMPTR PWR C
(g) 11E21, FLT CONT CMPTR SERVO C
(h) 11E35, FLT CONT CMPTR PWR R
(i) 11E36, FLT CONT CMPTR SERVO R
(j) 11H17, FLT CONT SHUTOFF TAIL L
(k) 11H18, FLT CONT SHUTOFF TAIL CENTER
(l) 11H27, FLT CONT SHUTOFF TAIL R

S 864-069
(5) Set the L, C, and R TAIL FLT CONTROL SHUTOFF switches on the right side panel, P61, to ON.

S 864-070
(6) Set the STAB TRIM switches on the P10 panel to NORMAL.

S 864-071

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(7) Pressurize the left, center, and right hydraulic systems (Ref 29-11-00).

S 984-072
(8) Manually operate the rudder system three times.

S 714-073
(9) Do MCDP Ground Test 09 - SERVO RUD (Ref 22-00-02).
   (a) Make sure no failure messages show during the test.

S 864-074
(10) Set the L, C, and R TAIL FLT CONTROL SHUTOFF switches to OFF.

S 864-075
(11) Set the two STAB TRIM switches to CUTOUT.

S 864-076
(12) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
   (a) 11C12, STAB TRIM SHUTOFF L
   (b) 11C13, STAB TRIM SHUTOFF CENTER
   (c) 11H17, FLT CONT SHUTOFF TAIL L
   (d) 11H18, FLT CONT SHUTOFF TAIL CENTER
   (e) 11H27, FLT CONT SHUTOFF TAIL R

S 214-077
(13) Examine the valves for hydraulic leakage and correct as necessary.

G. Put the Airplane Back to Its Usual Condition

S 414-078
(1) Close the access panel 324BL (Ref 06-42-00).

S 864-086
(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11H17, FLT CONT SHUTOFF TAIL L
(d) 11H18, FLT CONT SHUTOFF TAIL CENTER
(e) 11H27, FLT CONT SHUTOFF TAIL R

S 864-079
(3) Set the L, C, and R TAIL FLT CONTROL SHUTOFF switches to ON.

S 864-080
(4) Set the L and C STAB TRIM switches to NORMAL.

S 494-081

WARNING: STAY AWAY FROM THE NOSE GEAR WHEELS WHEN THE LOCKPIN IS REMOVED. THE NOSE WHEELS CAN MOVE QUICKLY TO THE CENTER POSITION. INJURY TO PERSONS OR DAMAGE TO EQUIPMENT COULD OCCUR.

(5) Make sure the nose gear wheels are in the center position and remove the nose gear steering valve lockpin (Ref 09-11-00).

S 864-082
(6) Set the MCDP to off.

S 864-083
(7) Remove hydraulic power if it is not necessary (Ref 29-11-00).

S 864-084
(8) Remove electrical power if it is not necessary (Ref 24-22-00).

LCCA ELECTROHYDRAULIC SERVOVALVE AND SOLENOID VALVES - REMOVAL/INSTALLATION

1. General
   A. Two lateral central control actuators (LCCAs) are in the left wing root and one is in the right wing root. The LCCAs can be accessed through the main landing gear wheel wells.

TASK 22-13-03-004-035

2. Remove the LCCA Electrohydraulic Servovalve or Solenoid Valve (Fig. 401)
   A. References
      (1) 24-22-00/201, Electrical Power - Control
      (2) 29-11-00/201, Main (Left, Center, and Right) Hydraulic Systems
   B. Access
      (1) Location Zones
          732/742 Main Landing Gear Body Door

      (2) Access Panels
          732/742 Main Gear and Wheel Well Components

   C. Prepare for the Removal

      S 864-036
      (1) Supply electrical power (Ref 24-22-00).

      S 864-037
      (2) Remove pressure from the applicable hydraulic system (Ref 29-11-00).
          (a) Left Side Upper LCCA (Left Autopilot) - left hydraulic system
          (b) Right Side LCCA (Center Autopilot) - center hydraulic system
          (c) Left Side Lower LCCA (Right Autopilot) - right hydraulic system

      S 864-038
      (3) Set the L, C, and R WING FLT CONTROL SHUTOFF switches on the right side panel, P61, to OFF.

      S 864-039
      (4) Set the two F/D switches on the MCP to OFF.

      S 864-040
      (5) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11E16, MODE CONT PNL L
          (b) 11E34, MODE CONT PNL R
          (c) 11H15, FLT CONT SHUTOFF WING L
          (d) 11H16, FLT CONT SHUTOFF WING CENTER
          (e) 11H26, FLT CONT SHUTOFF WING R

S 864-041

(6) Open these circuit breakers on the overhead circuit breaker panel, P11, for the applicable (left, center, or right autopilot) LCCA and attach DO-NOT-CLOSE tags:
 (a) Left Side Upper LCCA (Left Autopilot)
  1) 11E17, FLT CONT CMPTR PWR L
  2) 11E18, FLT CONT CMPTR SERVO L
 (b) Right Side LCCA (Center Autopilot)
  1) 11E20, FLT CONT CMPTR PWR C
  2) 11E21, FLT CONT CMPTR SERVO C
 (c) Left Side Lower LCCA (Right Autopilot)
  1) 11E35, FLT CONT CMPTR PWR R
  2) 11E36, FLT CONT CMPTR SERVO R

D. Remove the Valve

S 144-042

CAUTION: DO NOT LET CONTAMINATION GET INTO THE LCCA AND VALVES. THE LCCA AND VALVES ARE VERY SENSITIVE ELECTROHYDRAULIC DEVICES. CONTAMINATION IN THE LCCA AND VALVES CAN CAUSE DAMAGE TO EQUIPMENT OR FAILURE.

(1) Carefully clean the external areas of the valve and around the valve mounting surface.

S 034-043

(2) Remove the lockwire.

S 024-044

(3) Remove the mounting screws, washers, valve, O-ring, and seal plate from the LCCA.

S 434-045

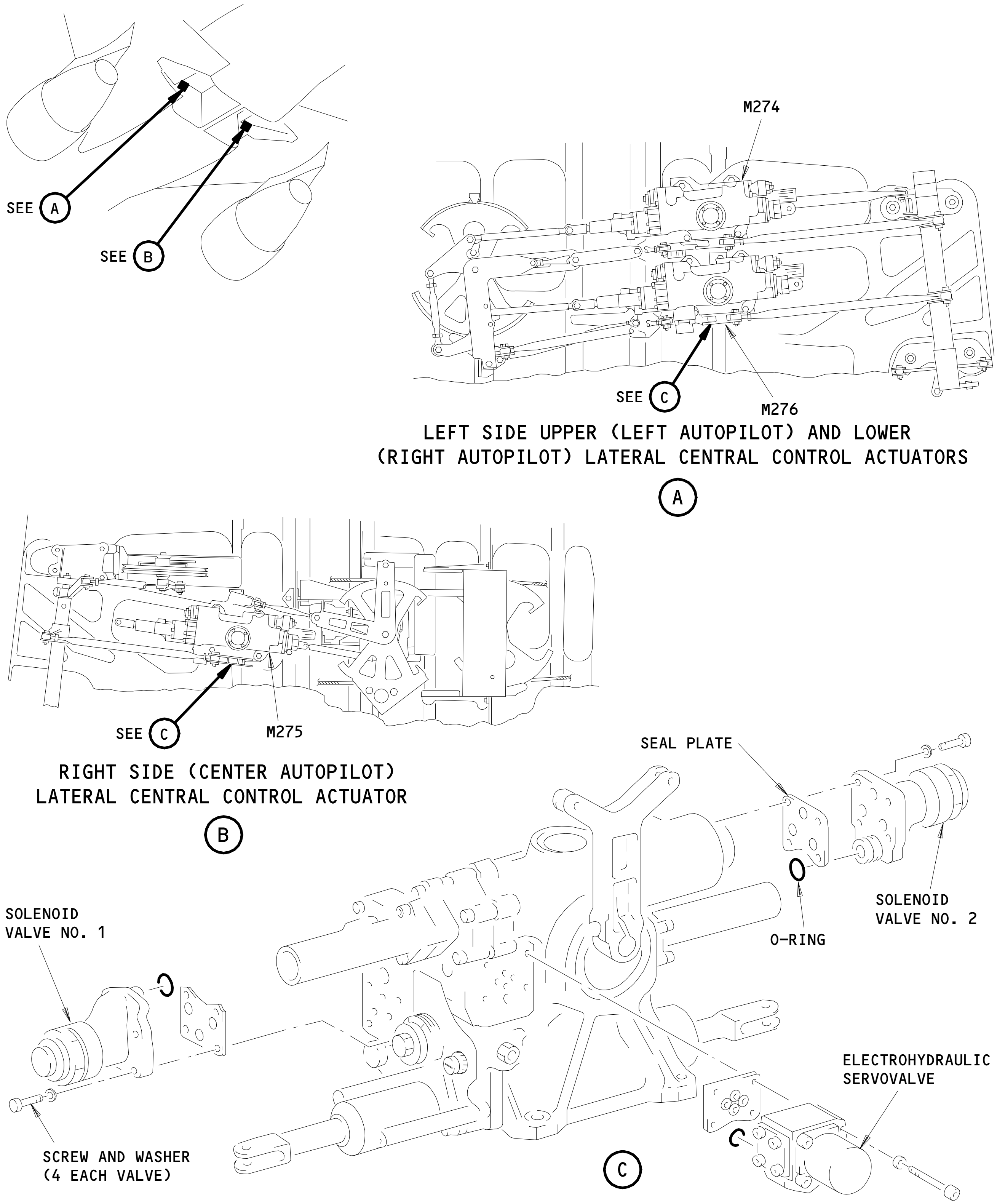
(4) Put covers on the hydraulic and electrical openings of the LCCA to make sure contamination does not get into them.

TASK 22-13-03-404-046

3. Install the LCCA Electrohydraulic Servovalve or Solenoid Valve (Fig. 401)

A. Consumable Materials
 (1) D00148 Hydraulic Fluid - fire resistant, BMS 3-11
 (2) D00209 Lubricant MCS 352B

B. References
 (1) 20-10-21/601, Electrical Bonding
 (2) 22-00-02/201, Autoflight BITE
 (3) 24-22-00/201, Electrical Power - Control



LCCA Electrohydraulic Servovalve and Solenoid Valves - Installation
Figure 401

(4) 29-11-00/201, Main (Left, Center, and Right) Hydraulic Systems

C. Access

(1) Location Zones

732/742 Main Landing Gear Body Door

(2) Access Panels

732/742 Main Gear and Wheel Well Components

D. Install the Valve

S 034-047

(1) Remove the covers from the openings.

S 114-048

(2) Clean the mating surfaces of the valve and the LCCA with clean hydraulic fluid. Do not let hydraulic fluid get into the electrical connector.

S 644-049

(3) Apply a small layer of MCS 352B lubricant to the new seal plate and the O-ring.

S 434-050

(4) Install the new O-ring on the valve.

S 424-051

(5) Install the seal plate and the valve with the screws and the washers.

(a) Tighten the screws to 60 inch pounds.

S 764-052

(6) Use a bonding meter to make sure the resistance between the valve assembly and the LCCA body is 5.0 milliohms (0.0050 ohm) or less (Ref 20-10-21).

S 434-054

(7) Put a lockwire on the mounting screws.

E. LCCA Electrohydraulic Servovalve and Solenoid Valves Test

S 864-055

(1) Supply electrical power (Ref 24-22-00).

S 864-056

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:

(a) 11E16, MODE CONT PNL L

(b) 11E17, FLT CONT CMPTR PWR L
(c) 11E18, FLT CONT CMPTR SERVO L
(d) 11E20, FLT CONT CMPTR PWR C
(e) 11E21, FLT CONT CMPTR SERVO C
(f) 11E34, MODE CONT PNL R
(g) 11E35, FLT CONT CMPTR PWR R
(h) 11E36, FLT CONT CMPTR SERVO R
(i) 11H15, FLT CONT SHUTOFF WING L
(j) 11H16, FLT CONT SHUTOFF WING CENTER
(k) 11H26, FLT CONT SHUTOFF WING R

S 864-057
(3) Set the L, C, and R WING FLT CONTROL SHUTOFF switches on the right side panel, P61, to ON.

S 864-058

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY OR DAMAGE CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(4) Pressurize the left, center, or right hydraulic system for the applicable LCCA (Ref 29-11-00).

S 984-059
(5) Manually operate the aileron three or more times.

S 714-060
(6) Do the MCDP Ground test 07 - SERVO AIL (Ref 22-00-02).
(a) Make sure no failure messages show during the test.

S 864-061
(7) Set the L, C, or R WING FLT CONTROL SHUTOFF switches to OFF.

S 864-062
(8) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11H15, FLT CONT SHUTOFF WING L
(b) 11H16, FLT CONT SHUTOFF WING CENTER
(c) 11H26, FLT CONT SHUTOFF WING R

S 214-063
(9) Examine the valves for hydraulic leakage and correct if it is necessary.

F. Put the Airplane Back to Its Usual Condition

S 864-064

(1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
   (a) 11H15, FLT CONT SHUTOFF WING L
   (b) 11H16, FLT CONT SHUTOFF WING CENTER
   (c) 11H26, FLT CONT SHUTOFF WING R

S 864-065

(2) Set the L, C, and R WING FLT CONTROL SHUTOFF switches to ON.

S 864-066

(3) Set the MCDP to off.

S 864-067

(4) Remove hydraulic power if it is not necessary (Ref 29-11-00).

S 864-068

(5) Remove electrical power if it is not necessary (Ref 24-22-00).

DAS FILTER - REMOVAL/INSTALLATION

1. General
   A. Three Directional Autopilot Servos (DASs) are side by side at the base of the vertical stabilizer, forward of the rudder hinge line. Each servo has an output rod connected to the rudder feel, centering and trim mechanism. You can get access to the servos through the removable panel 324BL on the lower left side of the vertical stabilizer (Ref 06-42-00). Each servo has an LRU hydraulic filter element.
   B. This procedure gives removal and installation steps for the two-piece filter cover that uses a high pressure filter screen and a filter adapter.

TASK 22-13-05-004-001

2. Remove the DAS Filter (Fig. 401)
   A. Equipment
      (1) Nose Gear Steering Valve Lockpin - A09003-1
   B. References
      (1) 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
      (2) 09-11-00/201, Towing
      (3) 22-00-02/201, Autoflight BITE
      (4) 24-22-00/201, Electrical Power, Control
      (5) 29-11-00/201, Main Left, Center, and Right Hydraulic Systems
   C. Access
      (1) Location Zone
         324 Vertical Stabilizer - Rear Spar to Trailing Edge

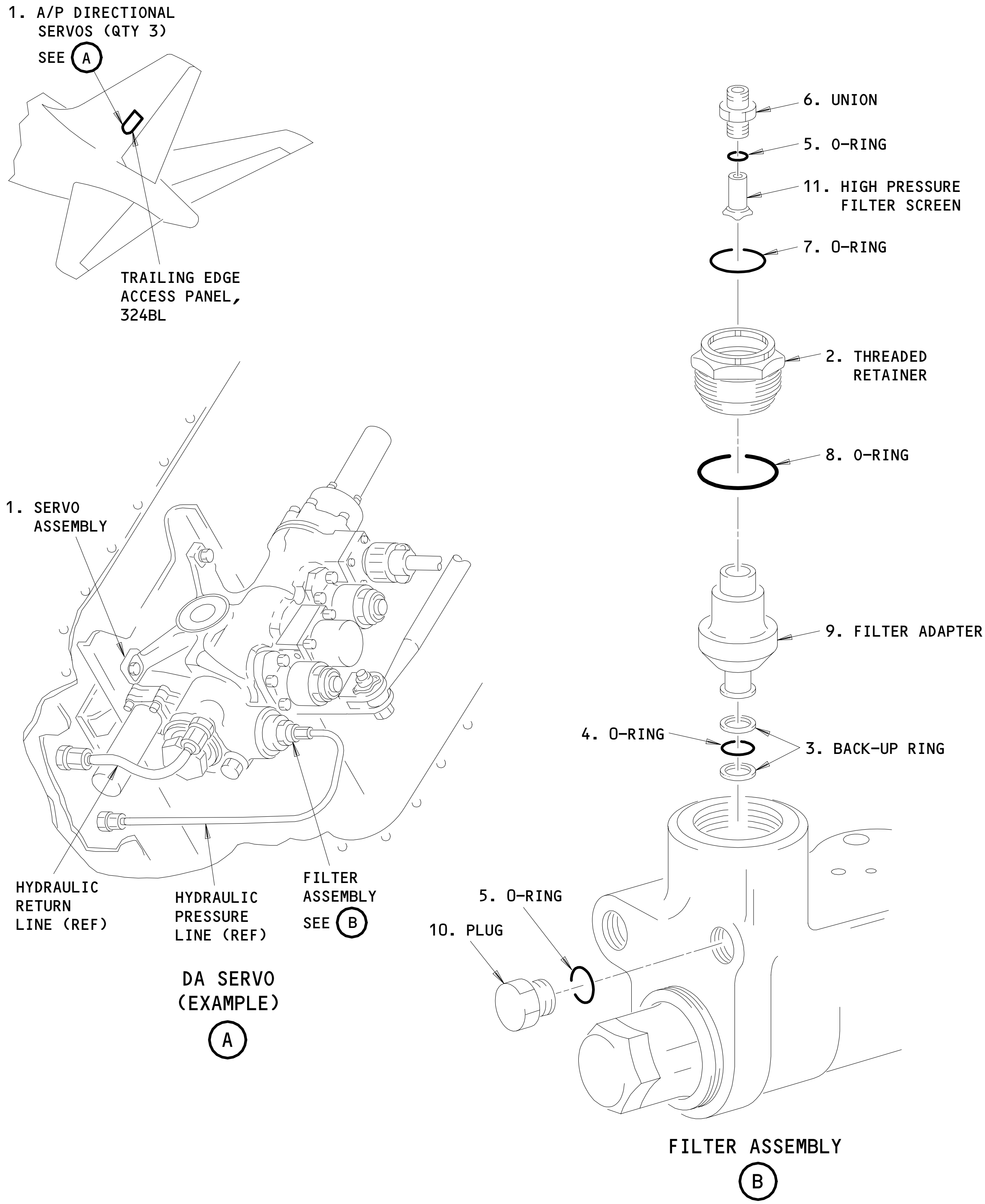
      (2) Access Panel
         324BL Vertical Stabilizer Rear Spar to Trailing Edge

   D. Prepare for Removal

      S 864-003

      WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

      (1) Remove the pressure from the applicable hydraulic system (Ref 29-11-00).



DAS Filter Installation
Figure 401

S 864-004

(2) Supply electrical power (Ref 24-22-00).
 (a) The left hydraulic system supplies power to the left Directional Autopilot Servo.
 (b) The center hydraulic system supplies power to the center Directional Autopilot Servo.
 (c) The right hydraulic system supplies power to the right Directional Autopilot Servo.

S 864-006

(3) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches, on the right side panel, P61, in the OFF position.

S 864-007

(4) Put the two STAB TRIM switches, on the control stand panel, P10, in the CUTOUT position.

S 864-008

(5) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
 (a) 11C12, STAB TRIM SHUTOFF L
 (b) 11C13, STAB TRIM SHUTOFF CENTER
 (c) 11H17, FLT CONT SHUTOFF TAIL L
 (d) 11H18, FLT CONT SHUTOFF TAIL CENTER
 (e) 11H27, FLT CONT SHUTOFF TAIL R

S 014-009

(6) Open the access panel, 324BL (Ref 06-42-00).

E. Remove Filter

S 164-012

CAUTION: DO NOT LET CONTAMINATION GET INTO THE DIRECTIONAL AUTOPILOT SERVO AND VALVES. THE DIRECTIONAL AUTOPILOT SERVO AND VALVES ARE VERY SENSITIVE ELECTROHYDRAULIC DEVICES. CONTAMINATION IN THE SYSTEM CAN CAUSE DAMAGE TO THE EQUIPMENT.

(1) Clean the external areas of the filter cover and the servo.

S 034-022

(2) Remove the lockwire.

S 034-023

(3) Remove the filter cover threaded retainer (2) and the filter adapter (9) from the servo assembly.

S 434-024

(4) Put a cover on the opening in the servo to prevent contamination of the hydraulic passages.

S 034-025
(5) Remove the O-rings (7 and 8) from the threaded retainer (2).

S 034-026
(6) Remove the two back-up rings (3) and the O-ring (4) from the filter adapter (9).

S 034-027
(7) Remove the union (6) from the filter adapter (9).

S 034-028
(8) Remove the filter screen (11) from the filter adapter (9).

S 034-029
(9) Remove the O-ring (5) from the union (6).

TASK 22-13-05-404-002
3. Install the DAS Filter (Fig. 401)
A. Equipment
(1) Nose Gear Steering Valve Lockpin - A09003-1
B. Consumable Materials
(1) D00290 Lubricant - Hydraulic System O-Rings, Back-Up Rings, Fittings, MCS 352B (AMM 20-30-04/201)
C. Parts

MM			IPC		
FIG	ITEM	NOMENCLATURE	SUBJECT	FIG	ITEM
401	1	Servo Assembly	22-13-01	01	5
	2	Retainer, Filter Cover Threaded			83
	3	Back-Up Ring			105
	4	O-Ring			100
	5	O-Ring			107
	6	Union			108
	7	O-Ring			82
	8	O-Ring			80
	9	Filter Adapter			78
	10	Plug			109
	11	High Pressure Filter Screen			106

D. References
(1) 06-42-00/201, Empennage (Major Zone 300) Access Doors and Panels
(2) 09-11-00/201, Towing
(3) 22-00-02/201, Autoflight BITE
(4) 24-22-00/201, Electrical Power, Control
(5) 29-11-00/201, Main Left, Center, and Right Hydraulic Systems

E. Access
(1) Location Zone
324 Vertical Stabilizer - Rear Spar to Trailing Edge

(2) Access Panel
324BL Vertical Stabilizer Rear Spar to Trailing Edge

F. Install Filter

NOTE: Apply MCS 352B lubricant to all O-rings and back-up rings as the parts are installed and assembled.

S 434-044
(1) Install the O-ring (5) onto the union (6).

S 434-045
(2) Install the filter screen (11) into the filter adapter (9).

S 344-093
(3) Install the union (6) into the filter adapter (9).

CAUTION: TIGHTEN THE UNION (6) ON A SURFACE WHICH WILL NOT CAUSE DAMAGE TO THE FILTER ADAPTER (9).

(a) Tighten the union to 110 ±10 pound-inches.

S 434-046
(4) Install the two back-up rings (3) and the O-ring (4) on the filter adapter (9).

S 434-047
(5) Install the O-rings (7 and 8) into the threaded retainer (2).

S 434-048
(6) Install the filter adapter (9) into the autopilot assembly (1) filter "boss" cavity.

S 434-049
(7) Install the threaded retainer (2) into the autopilot assembly (1).
(a) Tighten the threaded retainer to 20 ±2 pound-feet.

G. Directional Autopilot Servo Test

S 864-050
(1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11H17, FLT CONTROL SHUTOFF TAIL L
(d) 11H18, FLT CONT SHUTOFF TAIL CENTER
(e) 11H27, FLT CONT SHUTOFF TAIL R

S 864-051

(2) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches in the ON position.

S 864-052

(3) Put the two STAB TRIM switches in the NORM position.

S 864-098

WARNING: LOCK OUT THE NOSE GEAR STEERING BEFORE YOU DO ANY RUDDER MOVEMENT. IF THE NOSE GEAR STEERING IS NOT LOCKED OUT, INJURY TO PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR.

(4) Move the towing lever on the nose gear metering valve module to the towing position and install the nose gear steering valve lockpin (Ref 09-11-00).

S 864-054

(5) Supply pressure to the L, C, and R hydraulic systems (Ref 29-11-00).

S 864-055

(6) Manually operate the rudder system several times.

S 744-056

(7) Do the MCDP ground test 09-SERVO RUD.
   (a) Make sure that no fault messages are shown after the test is engaged (Ref 22-00-02).

S 794-057

(8) Examine the installation for leaks. Correct as necessary.

NOTE: Pressurize the servo for a minumum of 2 minutes before a leakage check is done.

S 434-076

(9) Attach the filter cover threaded retainer (2) and the plug (10) to the servo assembly (1) with a lockwire.

H. Put the Airplane Back to Its Usual Condition

S 414-058

(1) Close the access panel, 324BL.

S 864-060

(2) Set the MCDP to off.

S 864-059

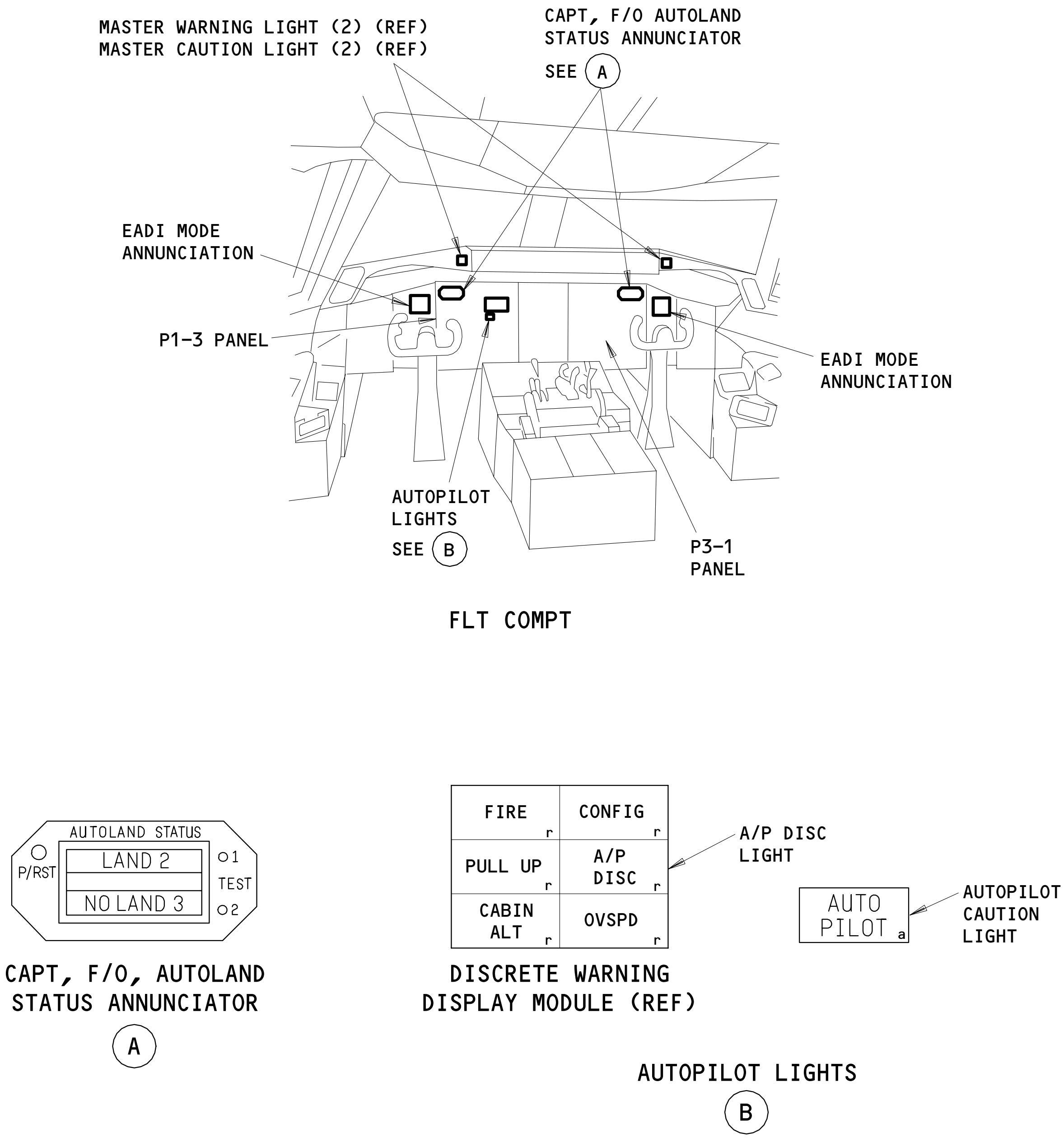
(3) Remove power from the L, C, and R hydraulic systems if it is not necessary (Ref 29-11-00).

S 864-061

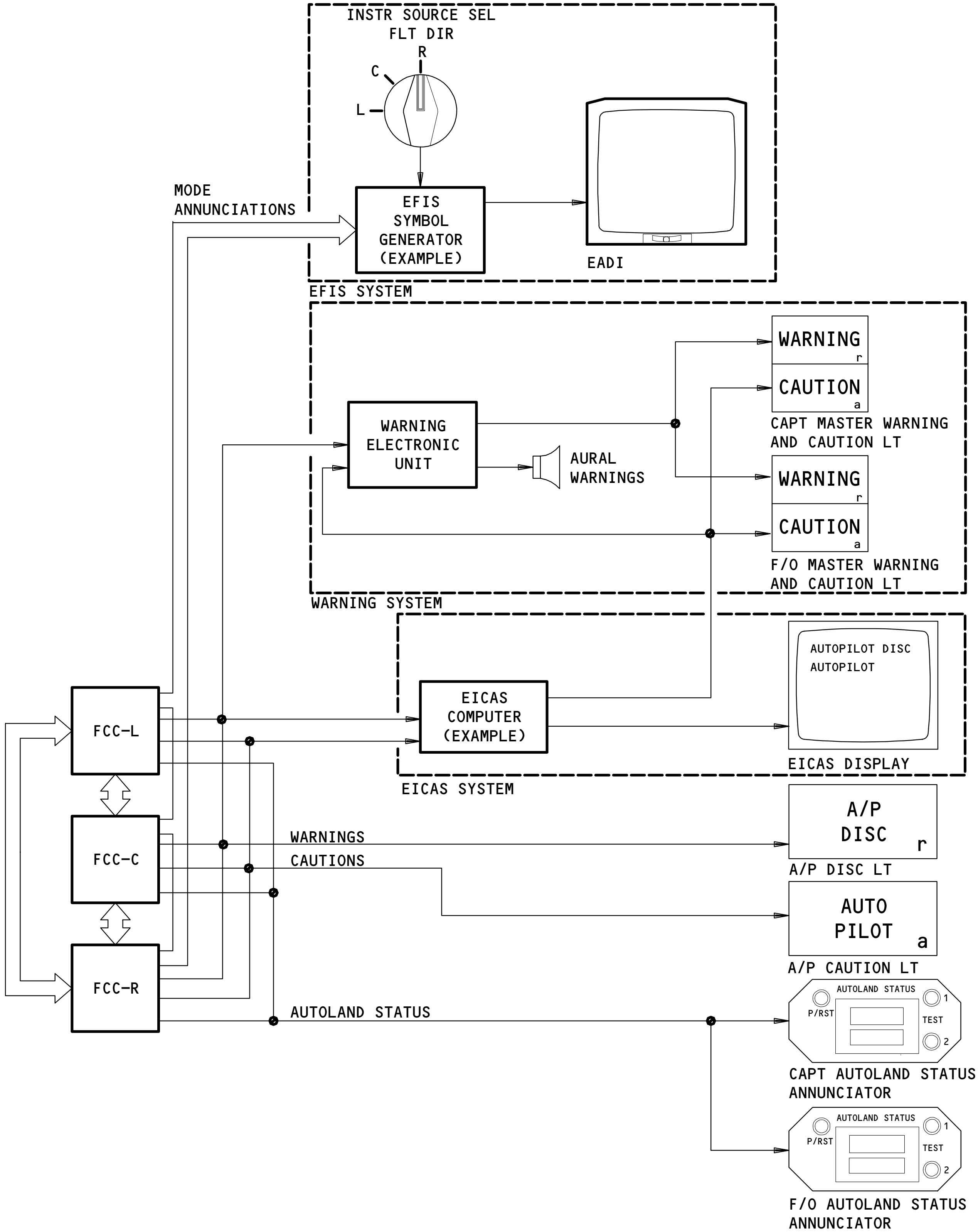
(4) Remove electrical power if it is not necessary (Ref 24-22-00).

AUTOPILOT/FLIGHT DIRECTOR WARNING AND ANNUNCIATION - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The warning and annunciation functions of the Flight Control Computers (FCCs) provide Autopilot/Flight Director System (AFDS) status and alerts the crew to system failures. The following displays are provided.
      (1) Autopilot (A/P) caution and warning lights located on the pilots' center instrument panel (P1).
      (2) Two master caution and warning lights located on the pilots' glareshield (P7) (Ref 31-51-00).
      (3) Two Electronic Attitude Director Indicators (EADI) one on the P1 panel and one on the P3 panel. The EADIs (Ref 34-22-00) display the following annunciations:
         (a) Flight mode annunciation
         (b) Flight mode failure annunciation
         (c) Autopilot CMD engage status annunciation
         (d) Flight Director (FD) engage status annunciation
      (4) Two Autoland Status Annunciators (ASA) on the pilots' main instrument panel (P1/P3).
   B. AFDS Warning and Annunciation (Fig. 2)
      (1) EADI Annunciation
         (a) The operational (armed and engaged) flight modes are annunciated on the Electronic Attitude Director Indicators (EADIs). The displays include autopilot and flight director engage status. The armed and engaged modes for pitch and roll are also displayed. Status and mode failures add a yellow line through the annunciation and the F/D bars are biased from view. The EADIs are controlled by the EFIS symbol generators. The symbol generators receive pitch, roll, and status mode information from the FCCS.
         (b) All three FCCs share CMD mode information with each other over the interchannel data buses. With a FCC engaged in the CMD mode, both EADIs display the CMD mode annunciation.
      (2) Autoland Status Annunciators
         (a) Autoland status is provided by two Autoland Status Annunciators (ASAs) connected in parallel. They are controlled by the three Flight Control Computers (FCCs). The ASAs display the loss of autoland capability in all phases of flight, and display the autoland status during the approach mode.
         (b) All three FCCs share autoland status information with each other over the interchannel data buses. The shared information allows each FCC to know the autoland status of the other two FCCs.



AFDS Warning and Caution Annunciation - Component Location
Figure 1



AFDS Warning and Annunciation
Figure 2

(3) Autopilot Caution Light

(a) The amber AUTOPILOT caution light is connected in parallel with the three FCCs. The light indicates a problem requiring crew attention. A separate output on the FCC's turns on both master CAUTION lights, an aural tone, and the EICAS computer displays an AUTOPILOT message (Ref 31-51-00).

(4) Autopilot Disconnect Warn Light

(a) The red A/P DISC (autopilot) disconnect warning is provided by a dual lamp indicator connected in parallel to the three FCCs. The lamp indicates disengagement of an FCC when no other FCCs are engaged. One lamp is controlled by normal power, the other by standby power (Ref 33-16-00). In ground test mode, only the right lamp (powered by standby power) illuminates when disengagement of an FCC occurs. Separate outputs from the FCCs turn on both master WARNING lights, an aural warning and the EICAS computer displays an AUTOPILOT DISC message (Ref 31-51-00).

2. <u>Component Details</u>

A. Autopilot/Flight Director Flight Mode Annunciation (Fig. 3)

(1) Status

(a) The autopilot engage status (CMD) is displayed in the top right corner of the EADI. The flight director engaged status (FD) is also annunciated.

(2) Roll Modes

(a) The armed and engaged roll modes are indicated below the status display.

(3) Pitch Modes

(a) The armed and engaged pitch modes are displayed in the second column from the left at the top of the EADI. Overspeed or minimum speed limit displays override normal annunciation.

(4) Fail Display

(a) Status or mode failures place a yellow line through the status or mode annunciation. The F/D command bars are also biased from view.

B. Autoland Status Annunciator (Fig. 4)

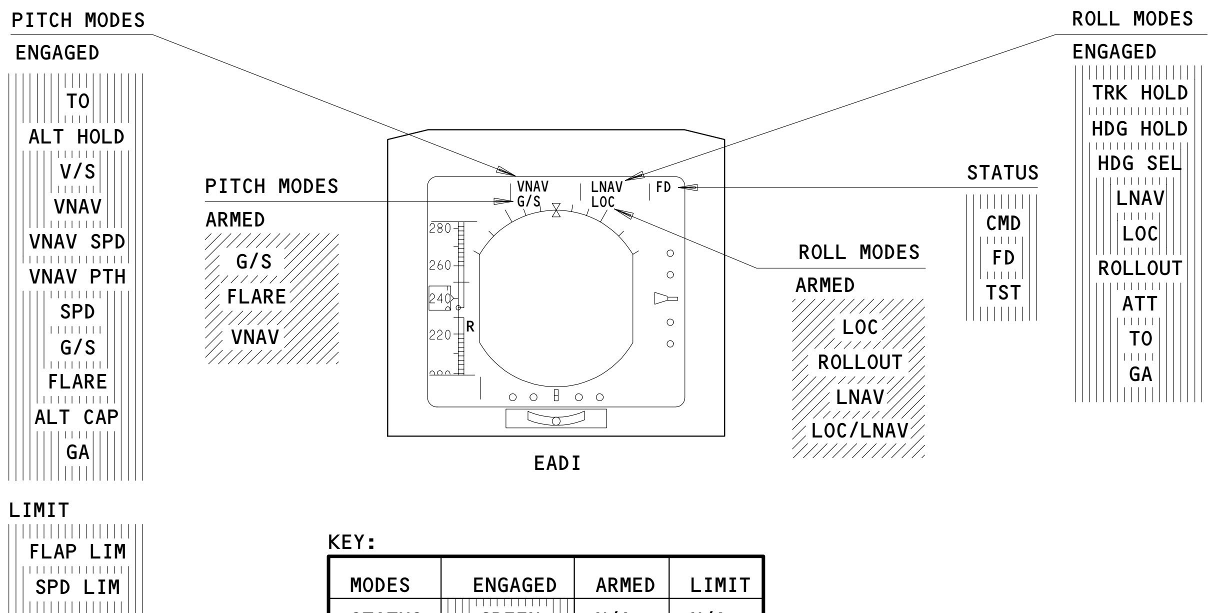
(1) Displays

(a) The Autoland Status Annunciator (ASA) consists of two mechanical displays driven by magnetic coils. The coils use 28 vdc to operate the display. Each display has three faces. One face is blank; the other two faces are designated as A and B. Figure 4 shows the displays and associated color.

(2) Switches

(a) Three momentary pushbutton switches are on the ASA panel. Pressing the reset (P/RST) switch clears the bottom display, which indicates the failure mode. Two Press-To-Test (PTT) switches are on the right side of the annunciator. Test 1 shows both A faces. Test 2 shows both B faces.

C. Autopilot Warning and Caution Lights and Annunciations (Fig. 5)



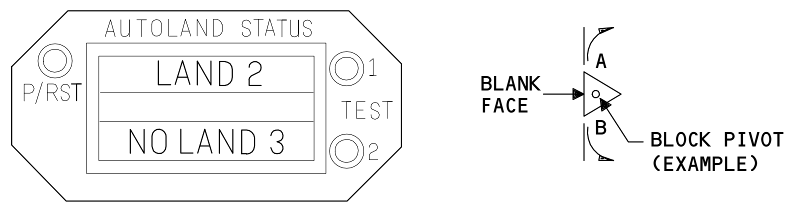
KEY:

MODES	ENGAGED	ARMED	LIMIT
STATUS	GREEN	N/A	N/A
PITCH	GREEN	WHITE	GREEN
ROLL	GREEN	WHITE	N/A

Autopilot/Flight Director System Mode Annunciation
Figure 3

PTT 1 SW PRESSED CAUSES ROTATION OF BOTH BLOCKS TO SHOW FACE A

PTT 2 SW PRESSED CAUSES ROTATION OF BOTH BLOCKS TO SHOW FACE B



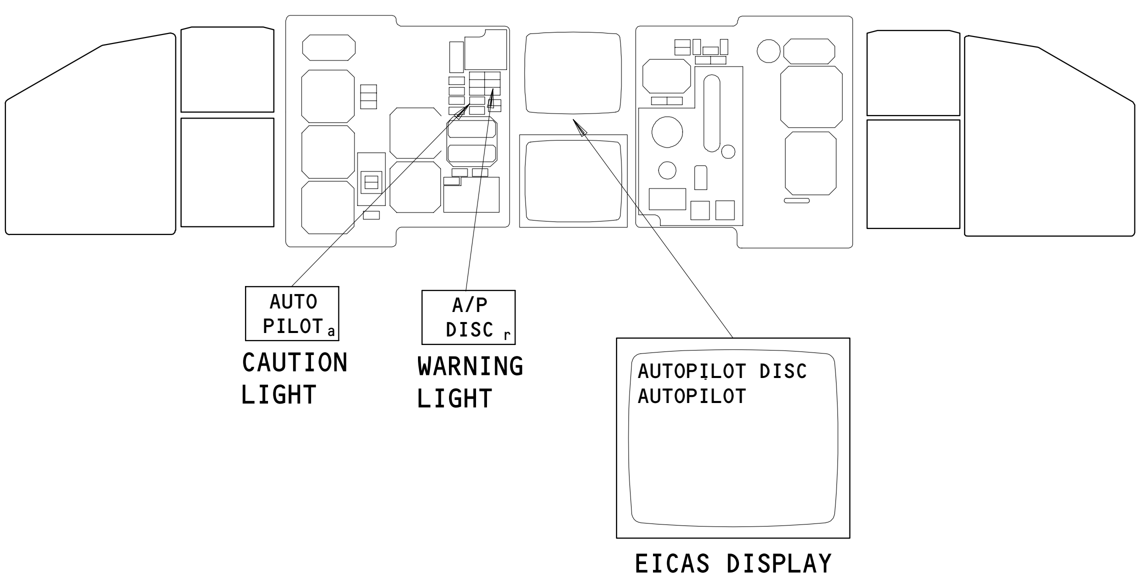
SYSTEM STATUS	DISPLAY
ON GROUND	READS OUT DEGRADATION OF AUTOLAND STATUS (NON-LATCHING)
IN ROUTE	READS OUT DEGRADATION OF AUTOLAND STATUS (LATCHING UNTIL RESET OR APPROACH)
AT APPROACH SELECT	AUTOMATIC RESET PULSE READS OUT CURRENT AUTOLAND SYSTEM STATUS (LATCHING UNTIL RESET)
APPROACH	READS OUT CURRENT AUTOLAND SYSTEM STATUS ABOVE ALERT HEIGHT (LATCHING UNTIL RESET)

SYSTEM CONFIGURATION	DISPLAY
FOLLOWING FIRST FAIL (FAIL PASSIVE), BEFORE MULTICHANNEL ENGAGEMENT	BLANK (BLACK) NO LAND 3 (AMBER) A
AFTER 2ND FAILURE, BEFORE MULTICHANNEL ENGAGEMENT	BLANK (BLACK) NO AUTOLAND (AMBER) B
FAIL OPERATIONAL, AFTER MULTICHANNEL ENGAGEMENT	LAND 3 (GREEN) A BLANK (BLACK)
FOLLOWING FIRST FAIL (FAIL PASSIVE), AFTER MULTICHANNEL ENGAGEMENT	LAND 2 (GREEN B /WHITE) NO LAND 3 (AMBER) A
AFTER PILOT PRESSES P/RST SW DISPLAY CHANGES	LAND 2 (GREEN B /WHITE) BLANK (BLACK)
SINGLE CHANNEL AUTOPILOT OPERTION (2ND FAILURE) AFTER MULTICHANNEL ENGAGEMENT	BLANK (BLACK) NO AUTOLAND (AMBER) B

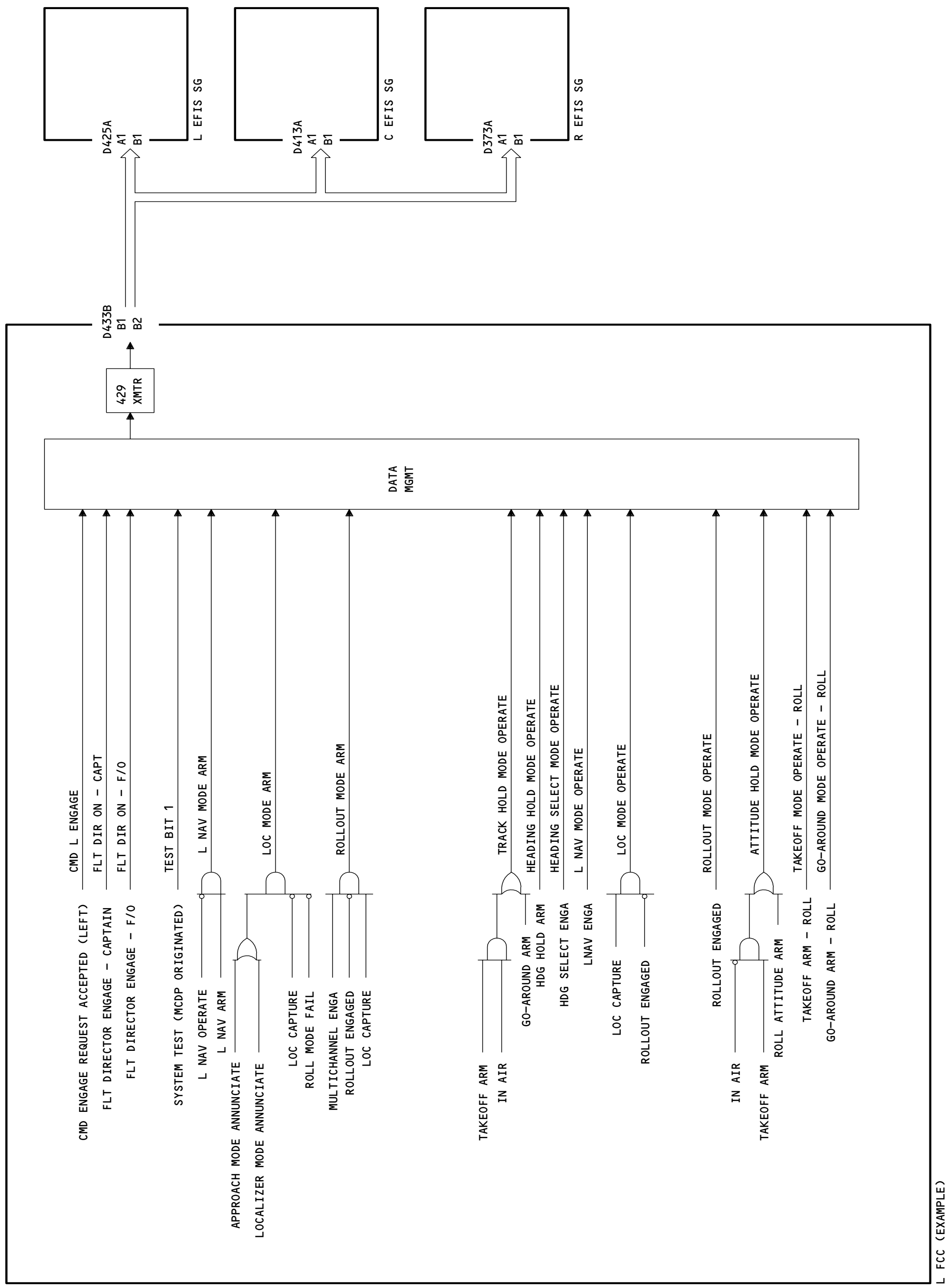
Autoland Status Annunciator
Figure 4

(1) Warning Light
(a) The red A/P DISC light is a dual-lamp indicator controlled by the FCCs. One lamp uses normal 28 vdc power. The other lamp uses 28 vdc standby power. In ground test mode, only the right lamp (powered by standby power) illuminates when disengagement of an FCC occurs. The A/P DISC light is part of the Discrete Warning Display Module (M779) (Ref 33-16-00).
(2) Caution Light
(a) The amber AUTO PILOT caution light is a dual-lamp indictor controlled by a single output from the FCCs. The lamps use 28 vdc power (Ref 33-16-00).
(3) Warning and Caution Annunciations
(a) The EICAS computer receives a ground from the FCCs when an autopilot disengage or autopilot input sensor signal failure occurs. This ground causes the EICAS computer to display a red (level A) or amber (level B) autopilot message (Ref 31-41-00).

3. Operation
A. Functional Description
(1) EADI AFDS Status and Roll Mode Annunciation Signals (Fig. 6)
(a) Status Annunciation
1) Autopilot status messages are sent from the FCC that has an autopilot or flight director channel engaged. Four status signals are supplied by the FCC. These are as follows:
a) CMD ENGAGE
b) FLT DIR ON - CAPT



Autopilot Warning and Caution Annunciations
Figure 5



EADI AFDS Status and Roll Mode Annunciation Signals
Figure 6

586403

c) FLT DIR ON - F/O
d) TEST BIT 1 - when the Maintenance Control and Display Panel (MCDP) is conducting a ground test (Ref 22-41-00)

(b) Roll Mode Annunciation

1) The FCC roll mode logic determines the following armed and engaged modes:

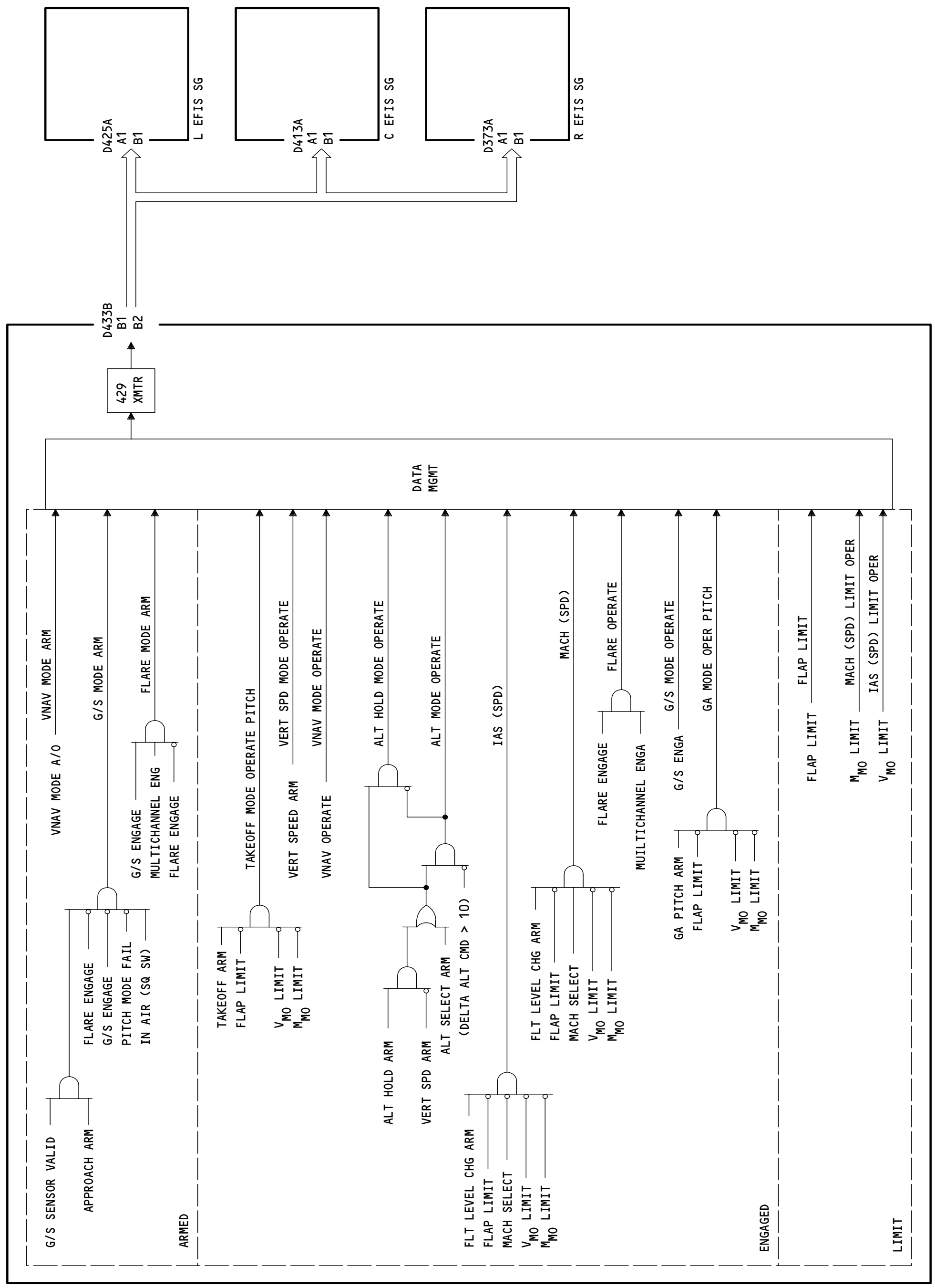
a) LNAV MODE ARM - LNAV armed but not engaged
b) LOC MODE ARM - Approach or localizer mode armed, no localizer capture, and no roll mode fail
c) ROLLOUT MODE ARM - Multichannel engaged, localizer captured, and rollout not engaged
d) LOC/LNAV MODE ARM - LOC and LNAV modes armed.
e) TRACK HOLD MODE OPERATE - Takeoff arm and in air, or go around mode arm
f) HEADING HOLD MODE OPERATE - Heading hold selected
g) HEADING SELECT MODE OPERATE - Heading select engaged
h) LNAV MODE OPERATE - LNAV engaged
i) LOC MODE OPERATE - Localizer capture and rollout disengaged
j) ROLLOUT MODE OPERATE - Rollout engaged
k) ATTITUDE HOLD MODE OPERATE - Takeoff arm and on ground, or roll attitude arm
l) TAKEOFF MODE OPERATE (ROLL) - Takeoff arm (roll)
m) GO AROUND MODE OPERATE (ROLL) - Go around arm (roll)

(2) EADI Pitch Mode Annunciation Signals - Armed, Engaged, and Limit (Fig. 7)

(a) Armed and Engaged Modes

1) The FCC pitch mode logic determines the following armed and engaged modes:

a) GS MODE ARM - The glideslope (GS) sensor must be valid and approach mode armed. Also, the airplane must be in air with the flare mode disengaged, GS disengaged, and no pitch mode fail
b) FLARE MODE ARM - GS engaged, multichannel engaged, and the flare mode disengaged
c) VNAV MODE ARM - VNAV armed but not engaged
d) TAKEOFF MODE OPERATE (PITCH) - Takeoff armed (limit modes override this display)
e) VERT SPD MODE OPERATE - Vertical speed armed
f) VNAV MODE OPERATE - VNAV engaged
g) ALT HOLD MODE OPERATE - Valid only when ALT MODE OPERATE is not valid. Also, altitude hold must be armed with no vertical speed arm, or altitude select arm must be valid
h) SPD - Flight level change armed and Mach or IAS selected on MCP (limit modes override this display)



EADI Pitch Mode Annunciation Signals - Armed, Engaged and Limit
Figure 7

i) GS MODE OPERATE - Glide slope engaged
j) FLARE OPERATE - Multichannel engaged and flare engaged
k) ALT MODE OPERATE - Valid only when the delta altitude command is greater than 10. Also, altitude hold must be armed with no vertical speed arm, or altitude select arm must be valid
l) GA MODE OPERATE (PITCH) - Go around pitch arm (limit modes override this display)

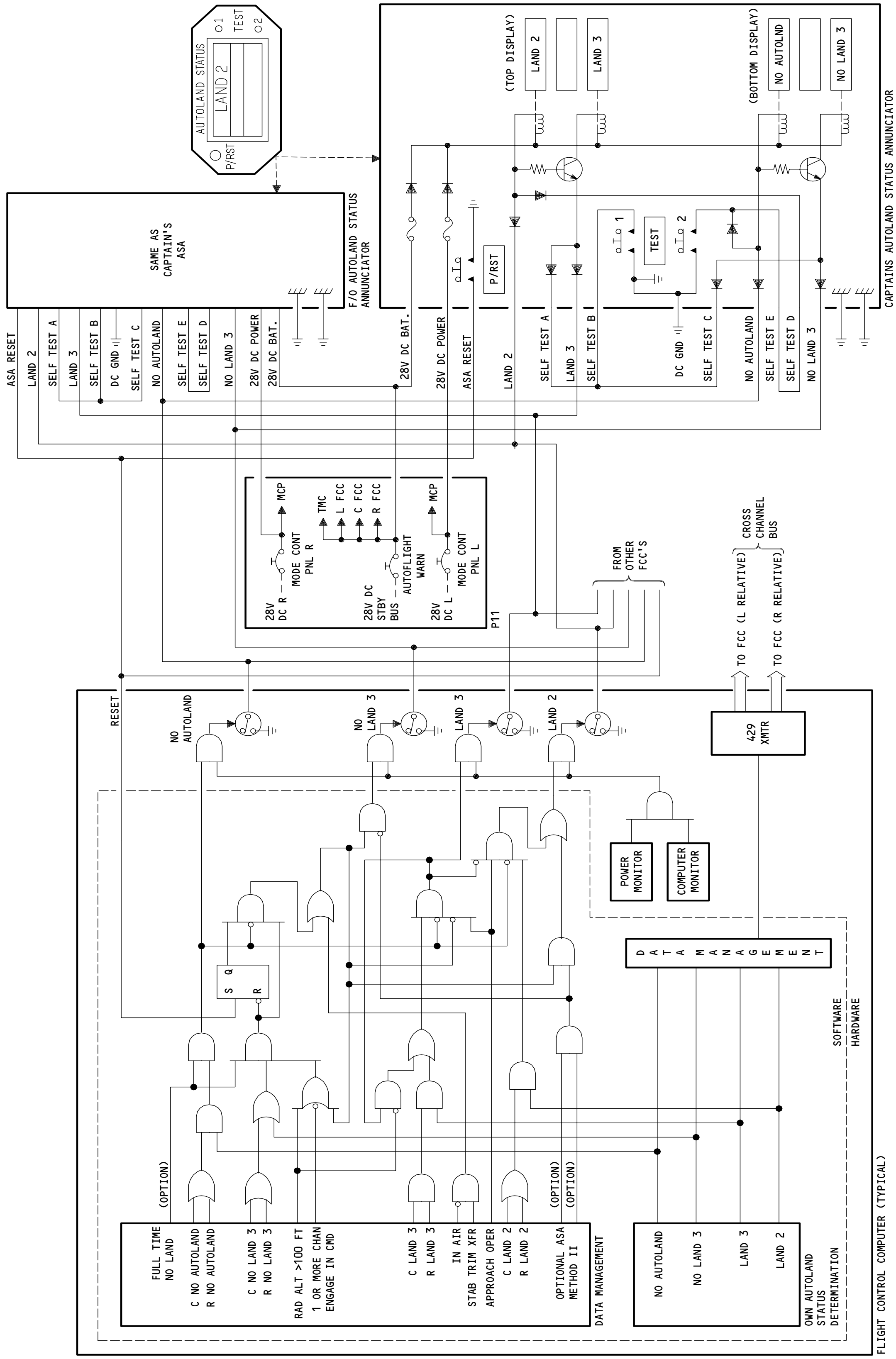
(b) Limit Modes
1) When the FCC is controlling speed through use of the elevator, overspeed limits override the engaged mode. The normal annunciation is inhibited and the operating limit mode is displayed. The following limit modes are displayed:
a) FLAP LIMIT
b) SPD LIMIT OPER - Maximum operating airspeed or Mach limit

(3) Autoland Status Annunciator (Fig. 8)
(a) General
1) Each FCC independently determines its own autoland status. Through cross-channel data transfer, the FCCs determine the proper display. Only one display output is possible unless a NO LAND 3 condition occurs. The NO LAND 3 and the LAND 2 displays will then be shown during multi-channel approach. In decreasing order of autoland capability, the displays are LAND 3, NO LAND 3, LAND 2, and NO AUTOLND.

(b) LAND 3
1) The LAND 3 display is enabled only when the approach mode is operational. The display requires that all three FCCs have determined LAND 3 is valid. LAND 3 is valid when all FCCs, A/P servos, and outer loop sensor (Air Data Computer, Radio Altimeter and Instrument Landing System) are normal. The display latches at 200 ft. of radio altitude.

(c) NO LAND 3
1) The NO LAND 3 display is enabled during the entire flight. The NO LAND 3 is displayed when one or more of the FCCs determine a NO LAND 3 status exists. A NO LAND 3 status exists if a single FCC, servo, or outer loop sensor failure occurs. This display is inhibited below 200 ft. unless already displayed, or no FCC is in the CMD mode. Transfer of the automatic stabilizer trim function from the first channel in command to another channel causes NO LAND 3 to be displayed after touchdown. When the P/RST switch is pressed, the reset flip-flop is set. This clears the display if the fail status is corrected.

(d) LAND 2
1) The LAND 2 display is enabled only when the approach mode is operational. The display requires an FCCs own determination of LAND 2, plus one other FCC with a LAND 2 status. LAND 2 is valid when the remaining FCCs, A/P servos, and outer loop sensors are normal.

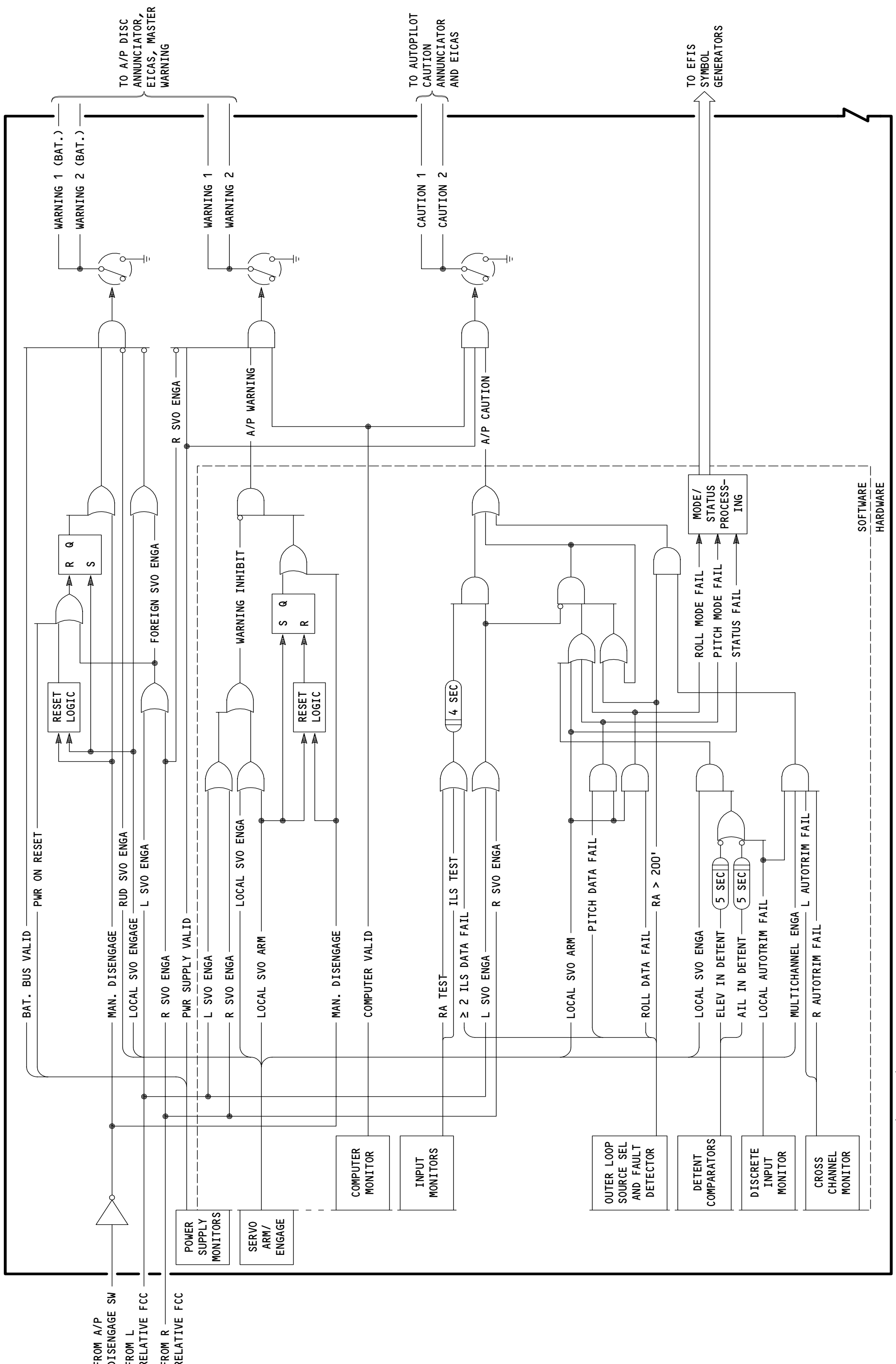


AFCS Autoland Status Annunciator Signal Development and Control Schematic
Figure 8

(e) NO AUTOLND
1) The NO AUTOLND display is enabled during the entire flight. To receive a NO AUTOLND display, at least two FCCs must determine that a NO AUTOLND status exists. A NO AUTOLND status exists if two FCCs, servos, or like outer loop sensors fail. When the P/RST switch is pressed, the reset flip-flop sets. This clears the display if the fail status is corrected.

(4) Autoland Status Annunciator Drive
(a) FCC Outputs
1) Software digital output signals in the FCC are converted to analog and supplied to hardware drivers. The drivers are inhibited during a power loss or computer failure, to prevent a false annunciation. The outputs of the three FCCs are connected in parallel to the ASAs.
(b) Display Drive
1) The top and bottom displays are driven when a ground is provided by the FCC. The ground completes the 28 vdc circuit to energize the magnetic coils and operate the mechanical display. Standby 28 vdc power connects in parallel with normal 28 vdc power. This provides a back-up source of power. A ground to the B side of the display overrides a ground to the A side.
(c) Switches
1) Diodes isolate the primary control from the test switches and allow testing the displays. Operation of the test switches checks power and display operation. Pressing TEST 1 displays both A faces. Pressing TEST 2 displays both B faces. The P/RST switches are paralleled and allow resetting the software logic.

(5) Autopilot/Flight Director System Caution and Warning Annunciation (Fig. 9)
(a) Autopilot Caution
1) Any AFDS failure requiring crew attention illuminates the AUTOPILOT caution light. A master CAUTION light is also illuminated, the EICAS computer displays an AUTOPILOT message, and an aural tone sounds (Ref 31-51-00). A power or computer failure inhibits the caution output to prevent a false annunciation. The following conditions cause a CAUTION SIGNAL prior to multi-channel engage only:
a) Failure of Air Data Computer, Radio Altimeter, and Instrument Landing System (outer loop) data or interfaces for 4 seconds when the pitch or roll servos are engaged.
b) An elevator or aileron servo cammout monitor was tripped.
c) An automatic stabilizer trim failure when the servos are engaged
d) Autopilot in flare mode with only one channel engaged in CMD mode.
e) Local inner loop data invalid in the F/D mode.



Autopilot/Flight Director System Caution and Warning Schematic
Figure 9

329726

f) Altitude hold or override mode cannot be engaged within 4 seconds.
g) Air Data Computer inner loop signals invalid for 4 seconds (airplane in cruise).
h) Self test failures occur in the ILS receivers / MMR or RA receivers.

2) During multi-channel engage, the CAUTION SIGNAL is inhibited except for the following conditions:
a) FCC program pin status invalid (Ref 22-11-00).
b) Simultaneous test on two or more ILS receivers (ILS test requests are normally inhibited during multi-channel engage)
c) Simultaneous ILS data failure (two or more ILS receivers) due to No Computed Data (NCD) in the data word SSM for more than four seconds
d) Automatic stabilizer trim failure in all three FCCs (airplane above alert height)

(b) EADI Failure Annunciation

1) The Electronic Flight Instrument System (EFIS) EADI indicates an autopilot mode failure with a yellow line through or removal of the engaged mode annunciation. An autopilot mode failure also biases the F/D command bars from view. A F/D mode failure is indicated when the F/D command bars are biased from view. An A/P or F/D mode failure is annunciated when the conditions under the caution section occur.

2) When the FCC(s) detects a pitch and/or roll mode failure, the FMA fault bit in the appropriate mode's data word is set high. The data word is sent to the EFIS symbol generators (S.G.) which draw a line through the failed mode annunciation. If the FCC(s) set the sign status matrix (SSM) bits in the status, pitch, and/or roll data word to NCD, the EFIS S.G.s remove the appropriate mode annunciation and bias the F/D command bars from view. A data bus failure between the FCC(s) and S.G.(s) also causes removal of the mode annunciation. When a FCC is in the F/D mode, NCD in the SSM of the data word causes the S.G.(s) to bias the F/D command bars from view.

(6) Autopilot Warning Annunciation

(a) General

1) Disengagement of the only engaged FCC causes the following:
a) The red A/P DISC warning light illuminates. (NOTE: In ground test mode, only the right half of the light illuminates.)
b) The red master WARNING light illuminates.
c) The EICAS computer displays an AUTOPILOT DISC message.
d) A siren alert sounds.

2) Two WARN outputs are generated for each of these warning functions. One WARN output uses software logic and is inhibited by a loss of normal power or a computer failure. The second WARN output uses hardware logic that is powered by 28 vac standby power. This ensures a valid warning if normal power is lost.

(b) Signal Generation

1) Latches are set when the FCC is engaged in CMD or the servos are armed. If the FCC subsequently disengages, the warning is issued if no other FCC is engaged. All outputs are generated anytime either autopilot disengage switch is operated.

(c) Reset

1) When the autopilot is disengaged with the A/P disengage switch, pressing the switch again resets the disengage warnings. Reset of an autopilot disengage warning due to any other means of disconnect is achieved by pressing the control wheel disengage switch once.

B. Autopilot/Flight Director System Signal Source Fault Detection (SSFD) (Fig. 10)

(1) Signal Source and Fault Detection provides independent selection and monitoring of sensor input data in each FCC. Signal Selection is provided to maintain servo tracking during multi-channel operation and when dual flight directors are selected. Monitoring sensor data allows detection of a failed sensor and permits exclusion of the invalid data from autopilot servo and flight director command computations. The SSFD also provides failure flags for ASA, Caution, and mode fail annunciations. There are three types of SSFD algorithms utilized for triplex, discrete, and dual/single source sensors.

(a) Median Select SSFD Process computes the mid-value of triplex (3) sensor inputs.

(b) Discrete SSFD Process perform a majority vote of three discretes.

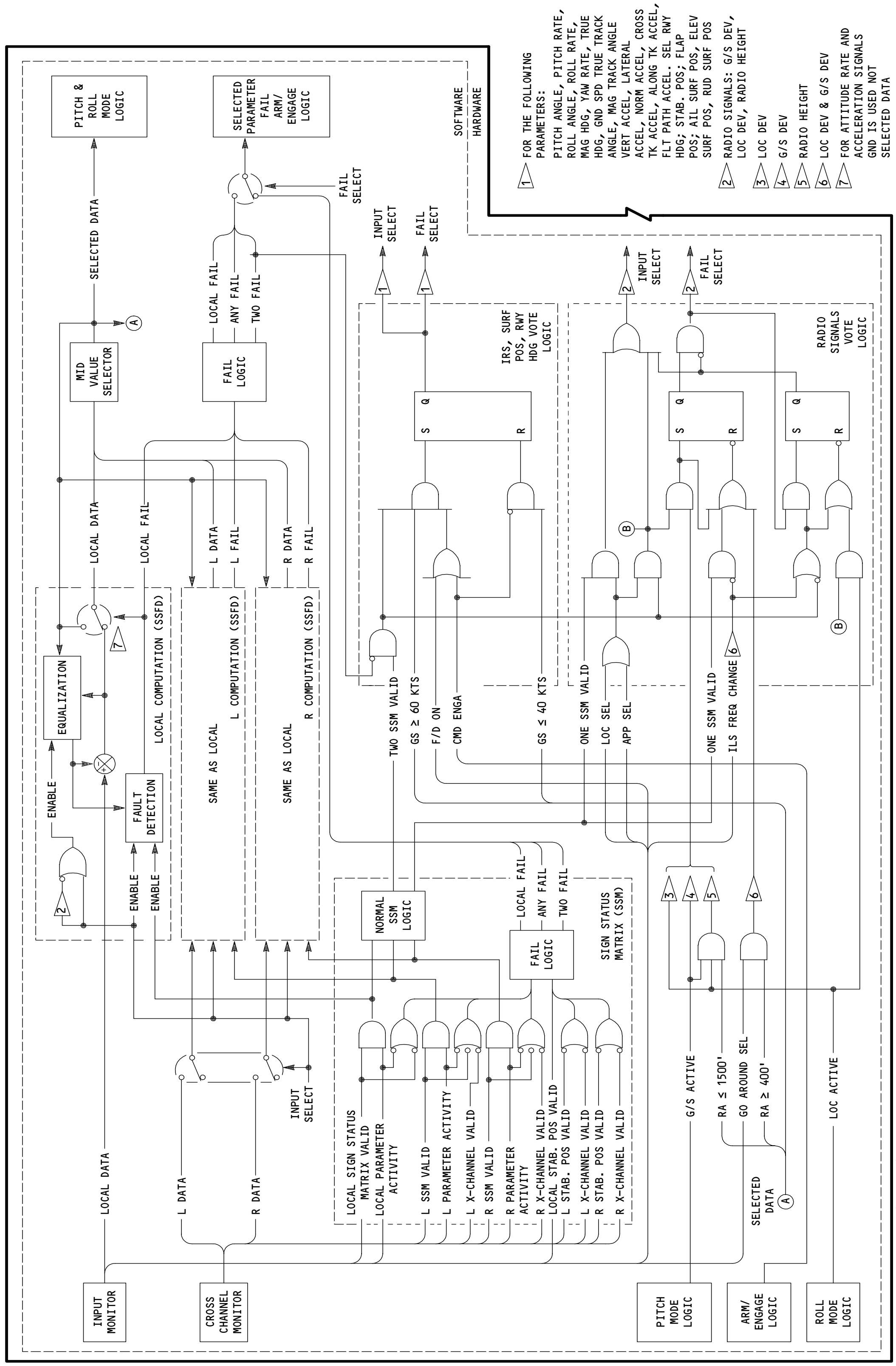
(c) Force Select SSFD process selects one data source for use by all FCCs. All ADC and ADC source IRU data (inertial altitude and inertial vertical speed) are force selected.

C. Autopilot/Flight Director System MCDP Monitor and Test (Fig. 11)

(1) The MCDP provides a manual test of the ASAs. The ASA test verifies the proper operation of the front panel test and reset switches.

(2) The FCC provides the following system interface tests:

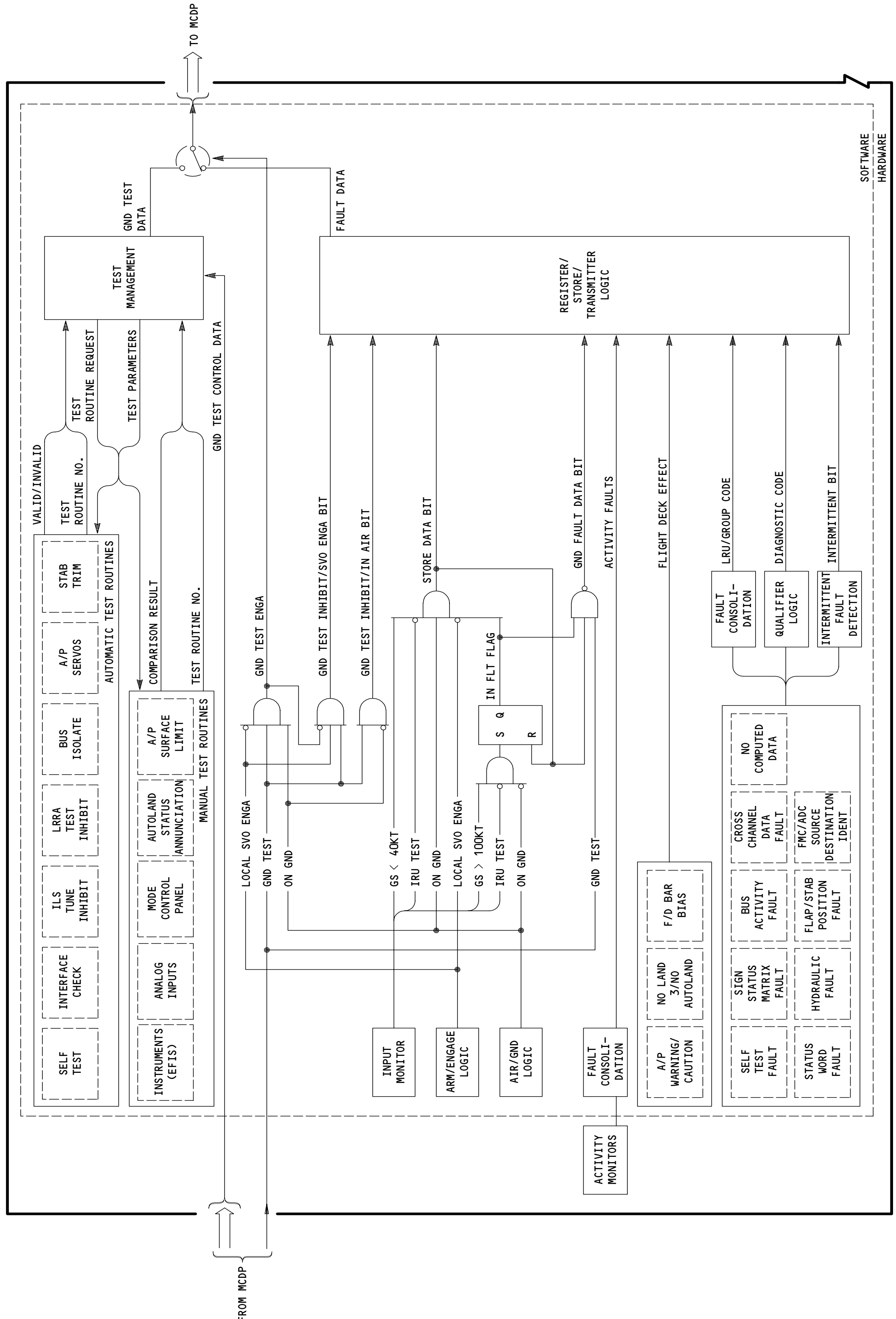
(a) ARINC 429 inputs from system units are checked in two ways. First, the FCC checks the sign status matrix to verify the validity of the incoming data. The buffer memory locations on I/O data path assembly are also periodically checked to determine if the data words have been updated. If the words have not been updated, then a fault is assumed and the data words at the tested memory locations are not used.



FLIGHT CONTROL COMPUTER (TYPICAL)

Autopilot/Flight Director System Signal Source Fault Detection (SSFD) Schematic
Figure 10

329727



Autopilot/Flight Director System MCDP Monitor and Test Schematic
Figure 11

329728

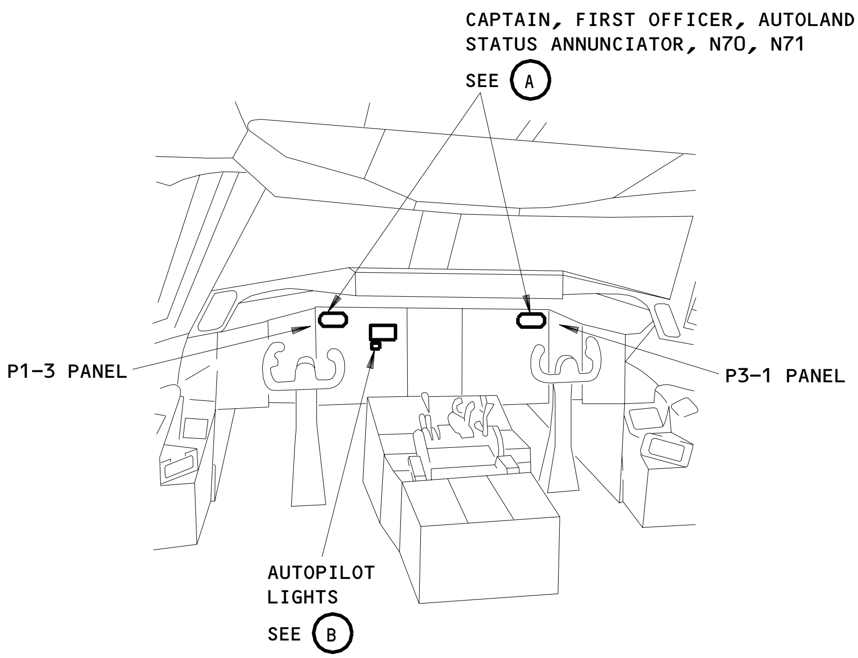
(b) Discrete input and output interfaces are monitored by software which detects shorted wires. These monitors are used for the caution/warning engage and arm solenoid power, trim outputs, engage and arm commands and inputs as well as the autoland status annunciator inputs and outputs. The interface monitor circuits are tested, as appropriate, at power up or prior to engagement of autopilot/autoland.

AUTOPILOT/FLIGHT DIRECTOR WARNING AND ANNUNCIATION

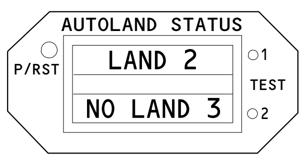
COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
ANNUNCIATOR - CAPT AUTOLAND STATUS, N70	--	1	FLIGHT COMPARTMENT, P1	22-14-01
ANNUNCIATOR - F/O AUTOLAND STATUS, N71	--	1	FLIGHT COMPARTMENT, P3	22-14-01
LIGHT - A/P DISC	--	1	FLIGHT COMPARTMENT, P1, DISCRETE WARNING DISPLAY MODULE, M779 (REF)	*
LIGHT - AUTOPILOT CAUTION, L19	--	1	FLIGHT COMPARTMENT, P1	*
MODULE - (FIM 33-16-00/101) DISCRETE WARNING DISPLAY				

* SEE THE WDM EQUIPMENT LIST

Autopilot/Flight Director Warning and Annunciation - Component Index
Figure 101

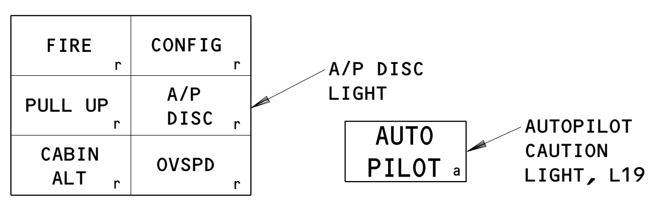


FLIGHT COMPARTMENT



CAPTAIN'S, FIRST OFFICER'S, AUTOLAND
STATUS ANNUNCIATOR, N70, N71

A



DISCRETE WARNING
DISPLAY MODULE, M779 (REF)

AUTOPILOT LIGHTS

B

Autopilot/Flight Director Warning and Annunciation - Component Location
Figure 102

AUTOLAND STATUS ANNUNCIATOR - REMOVAL/INSTALLATION

1. General
   A. Two Autoland Status Annunciators (ASAs) are found in the flight compartment. The left (Captain's) ASA is on the P1-3 panel. The right (First Officer's) ASA is on the P3-1 panel. Each ASA is attached to the panel with a clamp. Two connectors are on the rear of each ASA.

TASK 22-14-01-004-001

2. Remove the Autoland Status Annunciator
   A. Access
      (1) Location Zones
          211/212 Flight Compartment

   B. Prepare for Removal

      S 864-002
      (1) For the left ASA, open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11A17, AUTOFLIGHT WARN
          (b) 11E16, MODE CONT PNL L

      S 864-003
      (2) For the right ASA, open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11A17, AUTOFLIGHT WARN
          (b) 11E34, MODE CONT PNL R
   C. Remove the Autoland Status Annunciator

      S 034-004
      (1) Loosen the two screws that attach the clamp to the ASA. The screws are found in the top right and the bottom left corners.

      S 024-005

      CAUTION: CAREFULLY REMOVE THE ASA FROM THE PANEL TO PREVENT FORCE ON THE ELECTRICAL CABLES. DAMAGE TO ELECTRICAL CABLES COULD OCCUR.

      (2) Pull the ASA out of the panel.

S 034-025

CAUTION: CROSS CONNECTION POSSIBILITY WHEN WORKING WITH THIS COMPONENT. CLEARLY IDENTIFY CONNECTIONS UPON DISCONNECTION AND FUNCTION CHECK UPON RECONNECTION.

(3) Disconnect the electrical connectors.

TASK 22-14-01-404-007

3. Install the Autoland Status Annunciator

A. References

(1) 22-00-02/201, Autoflight BITE

(2) 24-22-00/201, Electrical Power - Control

B. Access

(1) Location Zones

211/212 Flight Compartment

C. Install the Autoland Status Annunciator

S 434-026

CAUTION: CROSS CONNECTION POSSIBILITY WHEN WORKING WITH THIS COMPONENT. POSITIVELY IDENTIFY CONNECTIONS PRIOR TO RECONNECTION.

(1) Connect the electrical connectors.

S 424-009

CAUTION: CAREFULLY PUT THE ASA INTO THE PANEL TO PREVENT FORCE ON THE ELECTRICAL CABLES.

(2) Put the ASA into the panel opening.

S 434-010

(3) Tighten the two screws, in the top right and the bottom left corners, that attach the clamp to the ASA.

D. Test the Autoland Status Annunciator.

S 864-011

(1) Supply electrical power (24-22-00).

S 864-012

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
   (a) 11A17, AUTOFLIGHT WARN
   (b) 11E16, MODE CONT PNL L
   (c) 11E34, MODE CONT PNL R

S 714-013

(3) Push the TEST 1 switch on the applicable ASA.
   (a) Make sure the top display shows LAND 3 and the bottom display shows NO LAND 3.

S 714-014

(4) Push the TEST 2 switch on the applicable ASA.
   (a) Make sure the top display shows LAND 2 and the bottom display shows NO AUTOLAND.

S 714-015

(5) Do MCDP ground test 06-ASA (Ref 22-00-02).
   (a) Make sure no failures occur during the test.

E. Put the Airplane Back to Its Usual Condition.

S 864-016

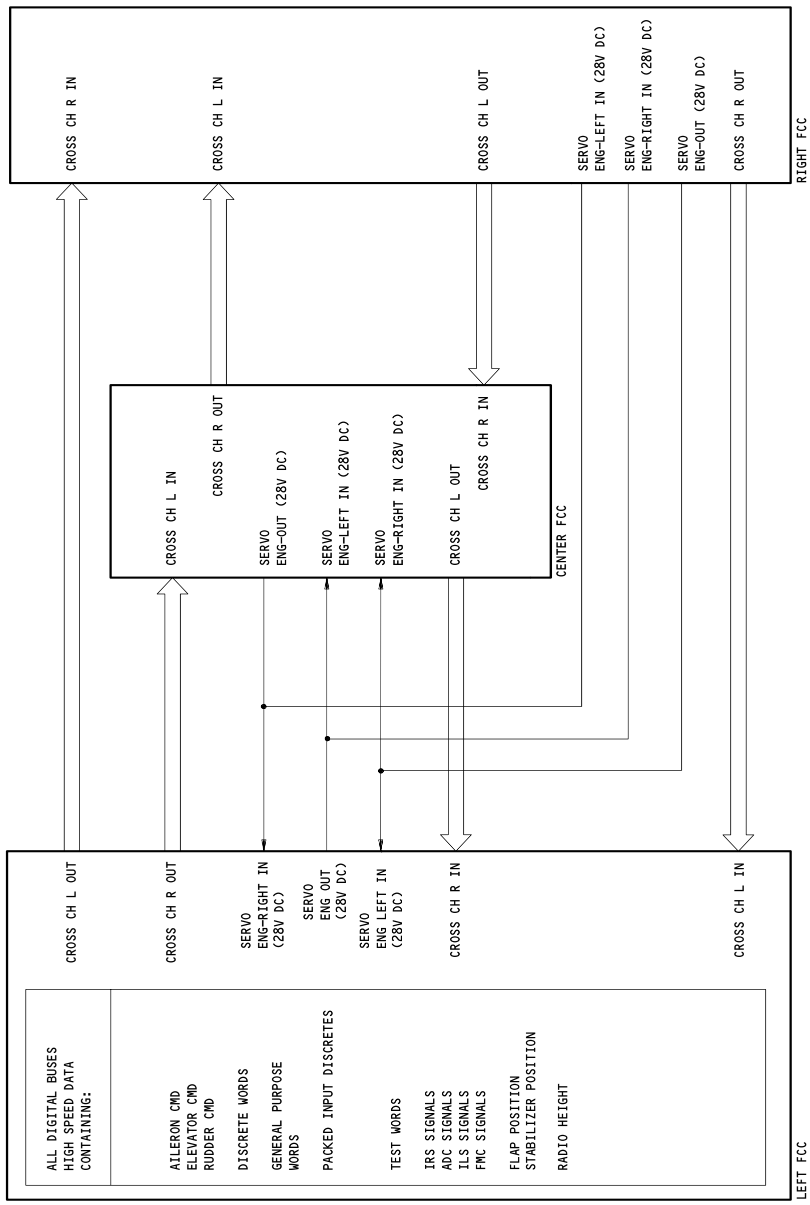
(1) Set the MCDP to off.

S 864-017

(2) Remove electrical power if it is not necessary (Ref 24-22-00).

AUTOPILOT/FLIGHT DIRECTOR INTERCHANNEL DATA - DESCRIPTION AND OPERATION

1. General
   A. The three Flight Control Computers (FCCs) communicate with each other via interchannel digital data buses and analog discretes. There are two input and two output interchannel high speed ARINC 429 buses connected to each FCC. There are two input and one output discrete data lines for analog discretes. Sensor signals, engage, and autoland status are shared to allow signal voting, fault detection, and determination of proper annunciations based on the status of all FCCs.
2. Operation
   A. Functional Description
      (1) Autopilot/Flight Director Interchannel Data (Fig. 1)
         (a) General
            1) All three FCCs communicate with each other via the four ARINC 429 high speed interchannel data buses and the three analog discretes data lines. The buses operate at 100 Kilobits Per Sec (100KBPS). The FCCs share sensor signals on the interchannel digital data buses. This allows signal voting and fault detection through comparison. The three FCCs share the interchannel discrete data lines containing engage and autoland status. This allows each FCC to determine proper annunciations based on the status of all FCCs.
         (b) Digital Buses
            1) All data is transmitted as non-standard words. The ARINC word is reformated so that data in the most significant bit is right shifted one position. This makes bit 29 a SSM bit. The right shifting makes bit 28 the most significant bit. This causes the full scale input range to double since the maximum input range is .5. Therefore, test equipment for a standard 429 bus cannot be used without manual conversion and interpretation of the data.
            2) Sensor inputs, surface commands, input discretes, and status are carried on the buses as follows:
               a) Aileron, elevator, and rudder surface positions in Binary Numerical Representation (BNR) format.



Autopilot/Flight Director Interchannel Data Schematic
Figure 1

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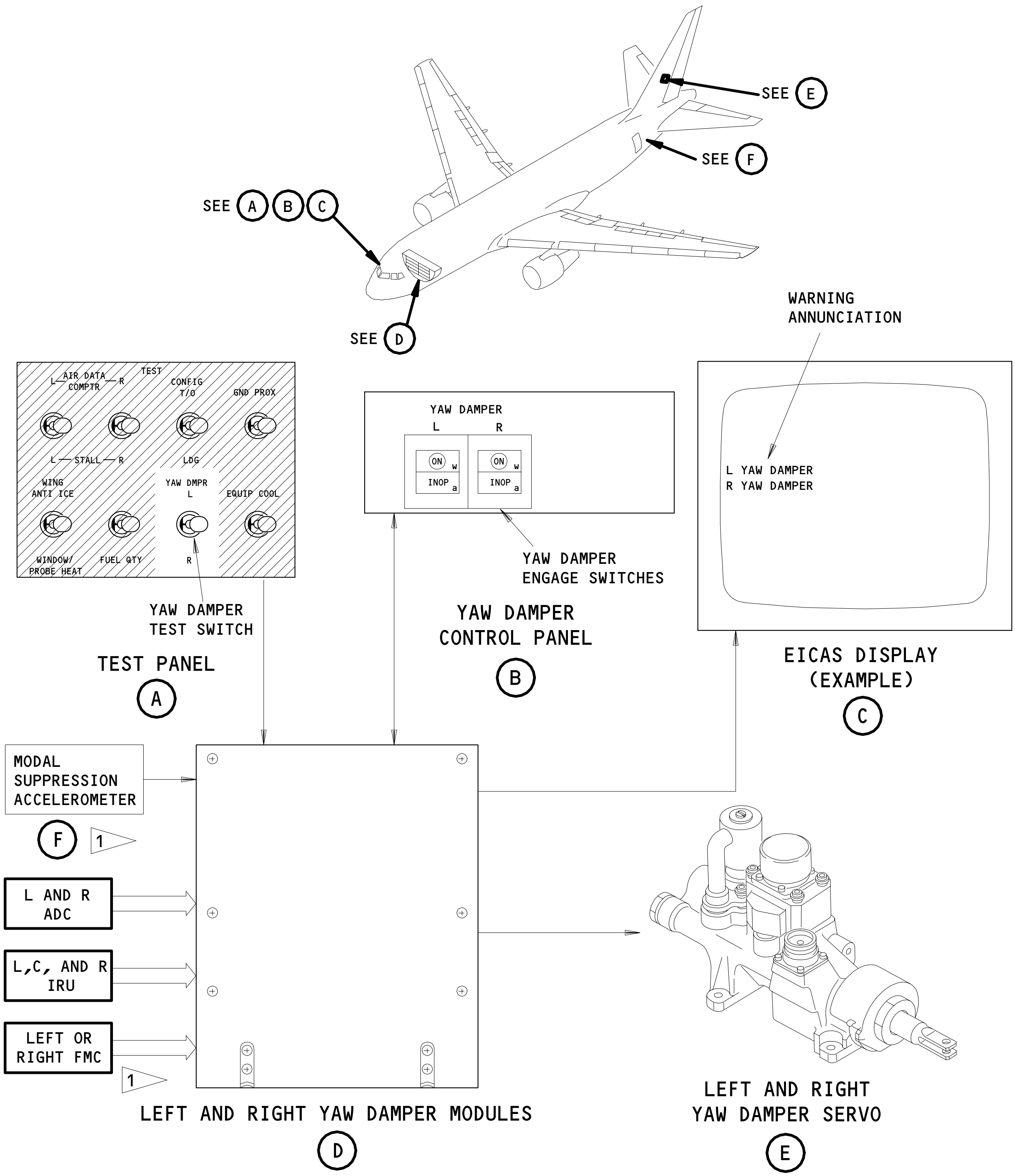
b) Four discrete words contain past value, autoland, armed, and active mode status.
c) Three general purpose words are BNR spares.
d) Two packed input discrete words. One word contains go-around and slats extended mode status. The second word is reserved for strapping pins.
e) Two test words are generated by each FCC. They verify the validity of the intercommunication paths.
f) Nineteen Inertial Reference Unit (IRU) signals contain vertical and lateral navigation data in BNR format.
g) Eight Air Data Computer (ADC) signals contain altitude, angle of attack, mach, airspeed, and impact pressure in BNR format.
h) Three Instrument Landing System (ILS) signals contain selected runway heading, localizer, and glideslope deviation in BNR format.
i) Three Flight Management Computer (FMC) signals contain vertical and lateral navigation in BNR format.
j) Flap position, stabilizer position, and radio height data in BNR format.
k) Heading and altitude reference data in BNR format for power-up synchronization.

(c) Analog Discretes
1) Pitch and roll servo engage discretes are generated by hardware from center, left, and right FCCs. The FCC's hardware logic use this as part of the information to establish autopilot disconnect warnings.

YAW DAMPER SYSTEM - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. An inherent characteristic of all swept back wing aircraft is a yawing motion that induces sideslip and roll. The Yaw Damper System (YDS) uses system inputs from the following to dampen undesirable sideslip and rolling motions:
      (1) Air Data Computers
      (2) Inertial Reference Units
      (3) 767-300 AIRPLANES;
          Flight Management Computers.
      (4) 767-300 AIRPLANES;
          Modal Suppression Accelerometer
   B. The YDS commands drive yaw damper servos which control the rudder mechanical linkage and power control actuators. Rudder displacement commands are proportional to yaw rate. Turn coordination is also provided by the YDS. Roll attitude inputs are used to compute turn coordination commands.
   C. Control Panel
      (1) The yaw damper panel on overhead panel P5 provides control and indication of yaw damper engagement status. The panel indicates a yaw damper ON or inoperative (INOP) state. The YAW DMPR test switch on the P61 panel initiates the system operational test.
   D. Yaw Damper Modules (2)
      (1) The yaw damper modules receive sensor inputs from the following systems to compute rudder commands appropriate to flight conditions:
         (a) Air Data Computers (ADCs)
         (b) Inertial Reference Units (IRUs)
         (c) 767-300 AIRPLANES;
             Flight Management Computers (FMCs)
         (d) 767-300 AIRPLANES;
             Modal Suppression Accelerometer
      (2) These commands drive the yaw damper servos which control the rudder through actuators. The modules monitor system operation and indicate system faults through automatic and manually initiated system testing. Yaw damper fault indications consist of the following:
         (a) Yaw damper panel INOP light
         (b) EICAS display unit annunciation
         (c) Fault message display
   E. Yaw Damper Servos
      (1) The yaw damper servos use electrical command inputs from the yaw damper modules. The electrical input commands control hydraulic flow to actuator pistons, which provide a mechanical output. This output is connected in series with manual and autopilot rudder commands supplied to rudder power control actuators, which actually move the rudder.



1 767-300 AIRPLANES

Yaw Damper System
Figure 1

(2) The maximum rudder authority for each yaw damper servo is 6 degrees (3 degrees in each direction). If both servos are engaged, the maximum authority is 12 degrees. The yaw damper servo command is zero while the airplane is on ground, except during preflight testing. The yaw damper command is provided to the servos via an easy-on/easy-off function to reduce transients during ground/air transitions. If a system fault is detected, the servos are disengaged.

(3) 767-300 AIRPLANES;
The modal suppression accelerometers provide the yaw damper modules with lateral acceleration data. Each yaw damper module has its own modal suppression accelerometer.

F. Yaw Damper Component Locations (Fig. 2)

(1) Flight Deck

(a) The pilots overhead panel (P5) contains the yaw damper controls and indicators. Each channel (left and right) has a separate yaw damper ON switch/light and INOP indicator. The YAW DMPR test switch is on the P61 panel.

(2) Main Electrical/Electronic Equipment Center

(a) The yaw damper modules are in the main electrical/electronic equipment center. The right module is installed on the E2-1 shelf. The left module is installed on the E1-1 shelf.

(3) Vertical Stabilizer

(a) The two yaw damper electrohydraulic servos are connected to the mechanical rudder linkage. The mechanical linkage is the interface between the yaw damper servos and the power control actuators, which move the rudders.

(4) 767-300 AIRPLANES;
Modal Suppression Accelerometer

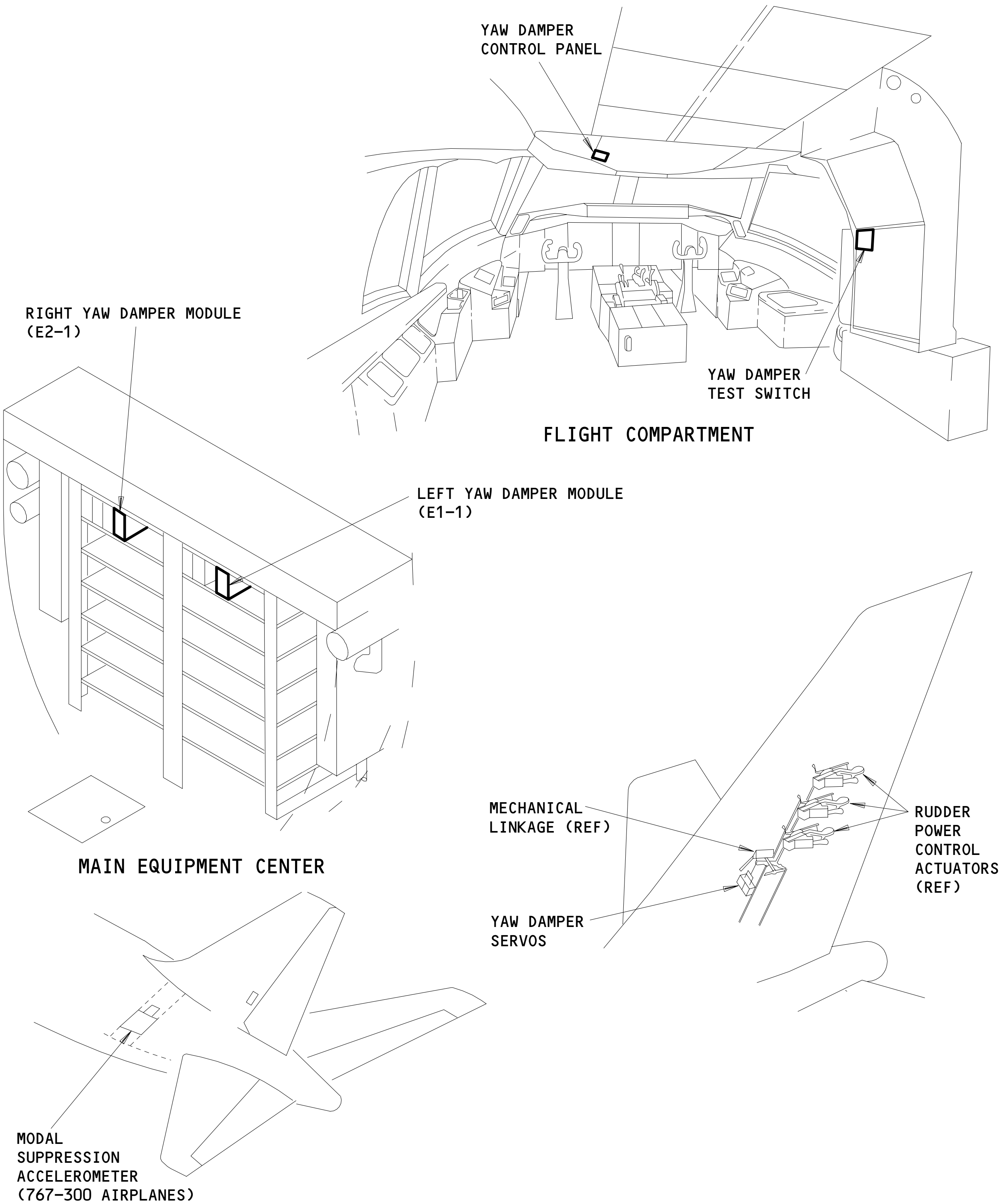
(a) The two modal suppression accelerometers are located in the area aft of the bulk cargo compartment and can be accessed through the aft cargo door (822).

G. Yaw Damper Block Diagram (Fig. 3)

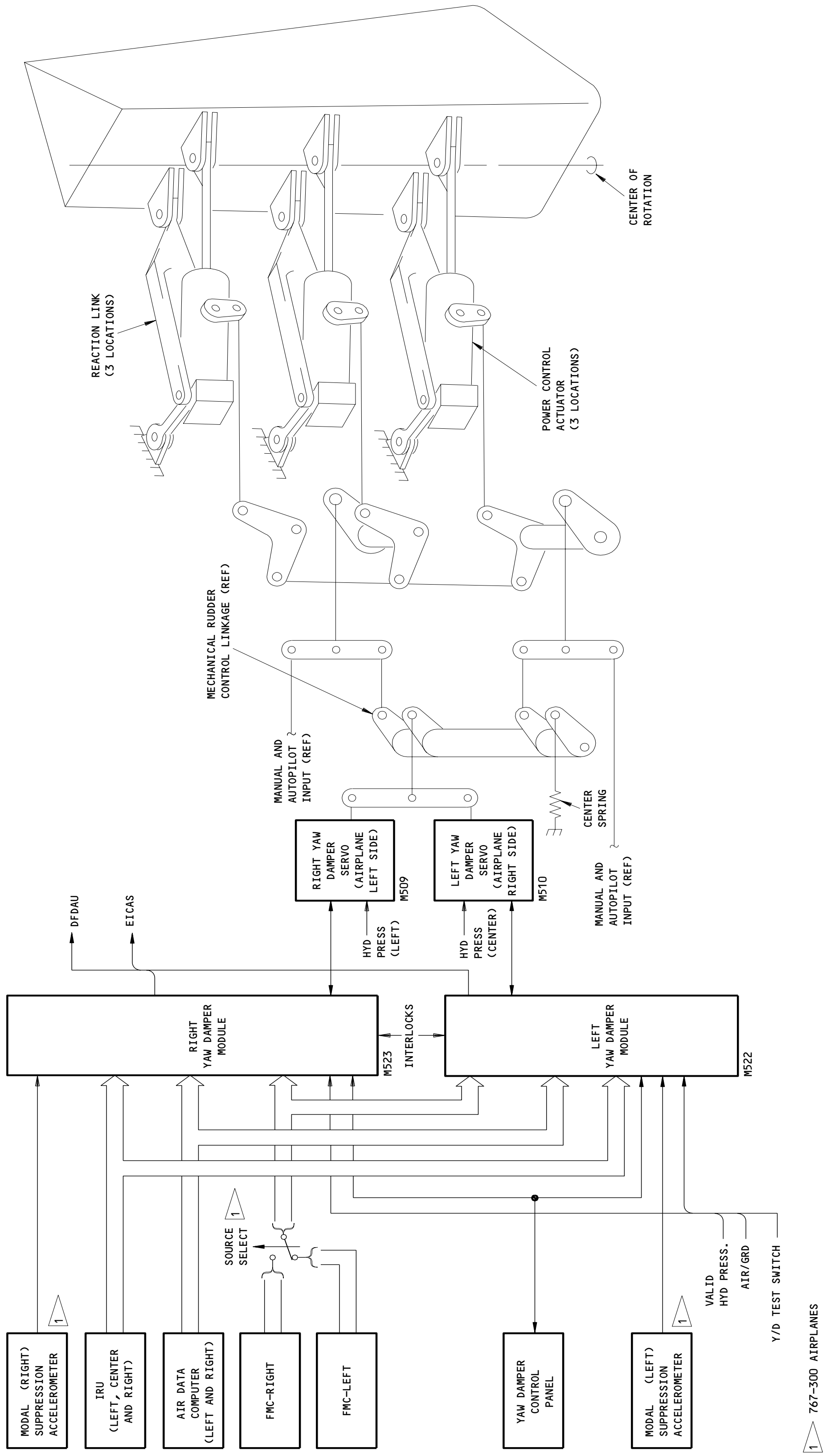
(1) System Inputs

(a) The YDS uses the following inputs to determine existing conditions, rudder commands, and system operation.

1) The left, right, and center IRUs each provide roll attitude, roll rate, yaw rate, lateral acceleration, and ground speed. The left and right ADCs each provide true airspeed (TAS), impact pressure (IAS) and angle of attack (AOA).



Yaw Damper - Component Location
Figure 2



Yaw Damper Block Diagram
Figure 3

2) 767-300 AIRPLANES;
The left or right FMC provides the gross weight.
3) The left and center hydraulic pressure switches sense normal hydraulic pressure to the upper and lower yaw damper servos, respectively. The air/ground relay provides an analog discrete signal indicating the airplane is airborne or on the ground.
4) Airplane type code is provided by program pins selectively grounded through airplane wiring. The yaw damper channel (left or right) code is also provided by program pins selectively grounded through airplane wiring.
5) 767-300 AIRPLANES;
The modal suppression accelerometers provide lateral acceleration data to their corresponding yaw damper modules.

(2) Yaw Damper Modules
(a) The yaw damper modules contain ARINC 429 receivers (data bus input) plus software signal check and selection programmed routines. As multi-input parameters are received, a median value selector provides single valid input parameters for microprocessor calculations.
(b) The yaw damper modules provide mode control for various operational conditions encountered over the full flight regime. Rudder commands computed by the modules assist in turn coordination and damping of undesirable yaw. Internal circuits in the modules conduct tests and exercise monitoring circuits. These provide failure indications and YDS disengagement if a fault is detected.
(c) The yaw damper modules perform signal conditioning for analog rudder command outputs to the servos. Servo Linear Variable Differential Transducer (LVDT) feedback monitored by the modules determines rudder position. The two yaw damper modules are connected by an interlock to determine if both modules are installed. If one module is not installed, the interlock is open circuited. This causes the installed module to illuminate an INOP annunciator (for the absent module) on the yaw damper panel.

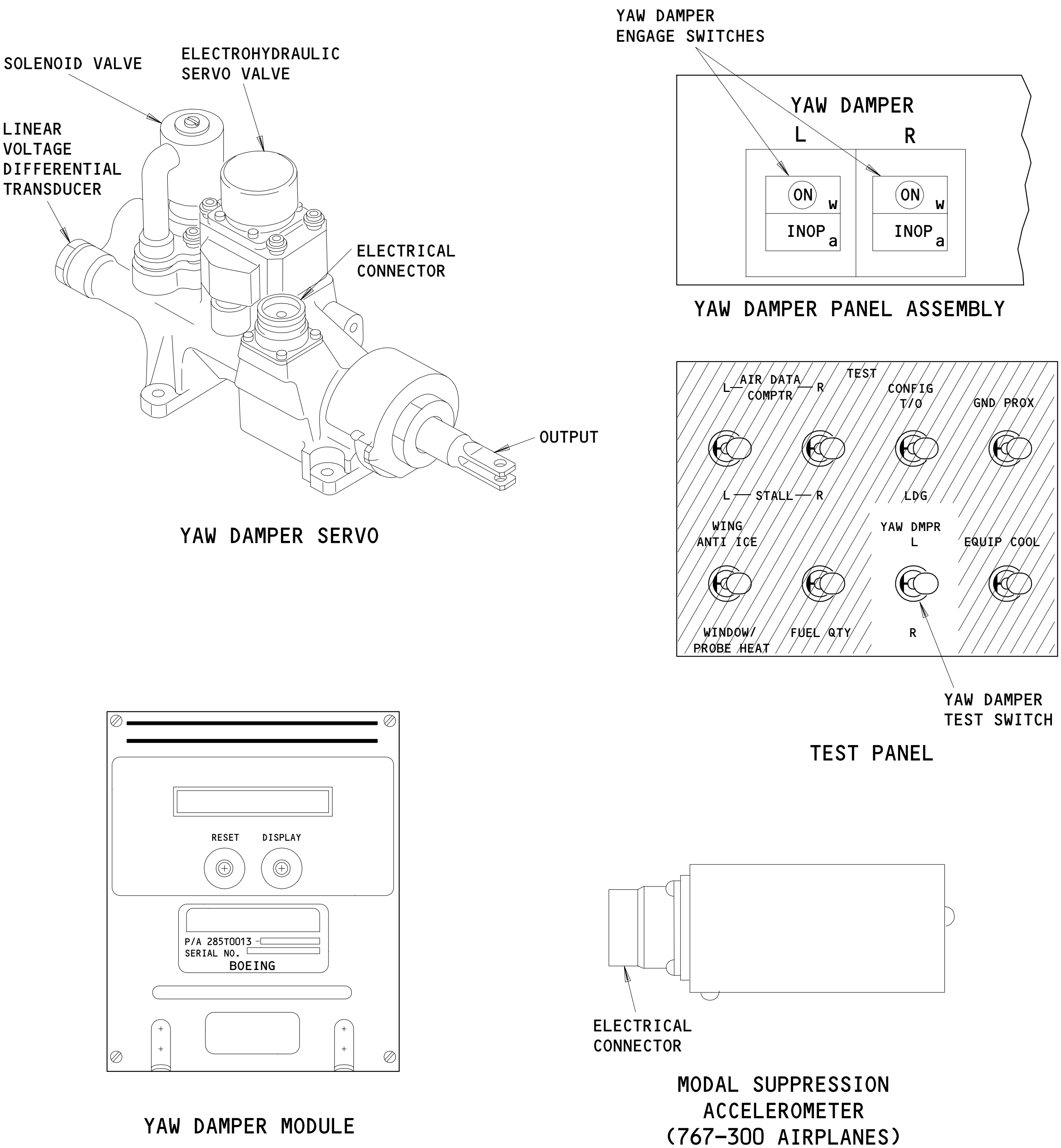
(3) Yaw Damper Servos
(a) Two yaw damper servos are connected in series to the rudder actuating linkage. The yaw damper servos provide the mechanical output to the rudder power control actuator input linkage. The servos receive electrical commands from the yaw damper modules and convert these signals to mechanical commands via an electrohydraulic servo valve (EHSV). The EHSV responds to electrical commands and delivers hydraulic flow proportional to the input signals. Hydraulic fluid then drives the servo actuator piston. The actuator piston drives the LVDT, which provides position feedback to the yaw damper module.

(4) System Annunciation
(a) The yaw damper ON and INOP annunciation is on the yaw damper panel. System failure status is also provided to the EICAS computer for display on the EICAS display unit (Ref 31-41-00/001).
(5) 767-300 AIRPLANES;
Modal Suppression Accelerometer
(a) The modal suppression accelerometers measure the lateral acceleration of the aft of the airplane. This information is provided to the corresponding yaw damper modules as an analog signal.
(6) Digital Flight Data Acquisition Unit (DFDAU) Interface.
(a) The left and right yaw damper modules provide the left and right yaw damper engagement status to the DFDAU (AMM 31-31-00/001).

H. Configuration
(1) Yaw Damper Module

EFFECTIVITY	-122	-124	-125	-126
SAS 050-051		BASIC	SB 22-62	
SAS 052-149				BASIC
SAS 150-156	BASIC		SB 22-62	
SAS 157,162-167		BASIC	SB 22-62	
SAS 158-161,168-999				BASIC
MTH 275-276	BASIC		SB 22-62	
MTH 277-278		BASIC	SB 22-62	
MTH 279-999				BASIC

2. Component Details (Fig. 4)
A. Yaw Damper Panel Assembly
(1) The YDS controls and indicators are on the overhead panel (P5). The ON lamps use 0-5 vac. The INOP indicator lamps use 28 vdc provided by the master dim and test system (Ref 33-16-00, Master Dim and Test). Case cooling is forced ambient cooling air. Two electrical connectors (one for each yaw damper module) are on the back side of the panel.
(2) Each yaw damper module has its own set of controls and indicators on the panel.
(a) The ON switch/light controls engagement of the YDS by controlling 28 vdc to the servo solenoid valve on the yaw damper servos. When the switch/light is first pressed, the ON legend is illuminated. Pressing the switch/light a second time turns off the YDS.



Yaw Damper System Components
Figure 4

(b) The INOP indicator illuminates under the following conditions:

NOTE: Yaw damper INOP is delayed for 15 seconds if it is activated by either YDM faults or after removal of hydraulic pressure (for annunciating servo loop faults only).

1) When the yaw damper module associated with the INOP is not installed (the opposite module must be installed and functioning properly)
2) When the yaw damping function is lost
3) During the pre-flight test (the YAW DMPR test switch initiates a pre-flight YDS test, provided the airplane is on ground)
4) During the power-up test

B. Yaw Damper Module

(1) Each yaw damper module consists of two digital computing channels (control and monitor) and a servo amplifier. The module consists of the following functional components:

(a) Fault Display - provides the logic to display fault messages on the front panel of the module.
(b) Monitor CPU - provides the digital computing function for the yaw damper, duplicating the computing of the control channel.
(c) Monitor Receiver - provides three high-speed and two low-speed ARINC 429 receiver interfaces for the control and monitor microprocessors.
(d) Analog - provides the output to the yaw damper servos and analog to digital and digital to analog signal conversion for both channels.
(e) Input/Output - provides input and output discrete signal conditioning for the control and monitor microprocessors.
(f) Control CPU - provides the digital computing function for the yaw damper.
(g) EMI Filter - provides EMI filtering for signal inputs, signal outputs and electrical power inputs for the control and monitor channels.

(2) The servo amplifier provides rudder command signals to the yaw damper electrohydraulic servo-valve. Feedback from the Linear Variable Differential Transducer (LVDT) position sensor is received by the actuator position demodulator assembly to close the servo loop. The assembly also provides servo loop monitoring which operates the automatic solenoid disengagement relay when a failure affecting operation is detected.

(3) One connector with three inserts is on the rear panel of each module. Two inserts contain the YDS signal pins and low voltage power pins. One insert contains the ground reference pins. Cooling is provided by air from the rack plenum being forced from the bottom to the top of the modules. A hinged front provides access to the interior of the module.

(4) The 28v dc engage servo solenoid power is routed from the control panel, through the module auto-disengage relay contacts, to the servo. Module operating power (±5v dc and ±15v dc) and the 26v ac LVDT demodulator reference is provided by the Control System Electronics Unit (CSEU) power supply modules (Ref 27-09-00/001).

(5) The fault detection logic in the modules isolates faults to the LRU level within the YDS and interfacing LRUs. The faults are divided into FAULTS NOW and LAST LEG FAULTS. FAULTS NOW corresponds to the fault monitors currently indicating a fault. LAST LEG FAULTS are the faults that occurred prior to the last air/ground transition.

(6) The following are the messages displayed by the yaw damper module:

(a) NO FAULTS - There are no FAULTS NOW and no LAST LEG FAULTS.

(b) NO FAULT NOW - There are no FAULTS NOW but there are LAST LEG FAULTS.

(c) NO LEG FAULT - There are FAULTS NOW but there are no LAST LEG FAULTS.

(d) FAULTS NOW - There are one or more FAULTS NOW faults, and they are displayed by pressing the DISPLAY button on the faceplate of the yaw damper module.

(e) LAST LEG - There are one or more LAST LEG FAULTS, and they are displayed by pressing the DISPLAY button on the faceplate of the yaw damper module.

(f) TEST RUNNING - Power Up Test (PUT) or Pre-Flight Test (PFT) is running on the yaw damper module.

(g) ACCEL - The yaw damper module has determined that there is a fault with the modal suppression accelerometer.

(h) L ADC, R ADC - The yaw damper module has determined that the data from the indicated air data computer is invalid.

(i) ADC DIFF - The yaw damper module has detected an invalid input from an air data computer but is unable to determine which one is at fault.

(j) AIRGND 1, AIRGND 1/2, AIRGND 2 - The yaw damper module has determined that the indicated air/ground discrete is invalid.

(k) AIR DIFF - The yaw damper module has detected an invalid air/ground discrete but is unable to determine which air/ground relay is at fault.

(l) 26 VAC - The yaw damper module has detected the loss of or the invalid level of the 26v RMS reference voltage.

(m) HYD SWITCH - The hydraulic pressure high signal agreed with the hydraulic pressure low signal for more than 2 seconds.

(n) YD TEST - The Pre-Flight Test (PFT) has failed.

(o) L IRU, C IRU, R IRU - The yaw damper module has determined that the data from the indicated inertial reference unit is invalid.

(p) IRU DIFF - The yaw damper module has detected an invalid input from an inertial reference unit but is unable to determine which one is at fault.

(q) IRU OFF - The IRUs are either off or they are on but do not transmit data.

(r) IRU NCD - The IRUs are on but no data is computed.

(s) 767-300 AIRPLANES;
FMC INPUT - The yaw damper has determined the data being received from the flight management computer to be in error, or the interface between the yaw damper and flight management computer has been interrupted.

(t) LVDT - The yaw damper module has detected that the LVDT common mode voltage is low. During the preflight test, an LVDT fault is recorded if the common mode voltage is low and a position test fault is detected (valid high hydraulic pressure must be present).

(u) YD ACT - During normal operation, a yaw damper actuator fault occurs when the yaw damper module detects a servo loop fault. During the preflight test, a yaw damper actuator fault is recorded if a position test fault occurs. To record a YD ACT fault, valid high hydraulic pressure must be present and the LVDT common mode voltage must be normal.

(v) YDM - A yaw damper module fault is recorded and latched with the following faults: power up test fault, preflight test fault, A/C type code input fault, control or monitor channel command coincidence fault, computing channel fault or an accelerometer excitation voltage interface fault. A YDM fault is also recorded when either control or monitor channel detected auto-disengage relay faults.

1) If no YDM fault exists, the YDM fault latch resets for the following conditions:

a) Successful completion of power up test or maintenance reset on ground.

b) Power up reset, manual engage or YDM autoreset in air or on ground.

(7) Fault messages are displayed on the 12 character alphanumeric LED display located on the faceplate of the yaw damper module. To view the fault messages, the DISPLAY button is pressed on the faceplate of the yaw damper module until all the fault messages are displayed. To display the fault messages one at a time, the DISPLAY button is pressed and released to display each message. The fault messages automatically extinguish from the display after 30 seconds. LAST LEG FAULT messages are distinguished from FAULTS NOW messages by an asterick (*) preceeding the fault message. To view the first fault message, the DISPLAY button is pressed and held for over 2 seconds.

(8) When the RESET button on the faceplate is pressed, all yaw damper module monitors and control logic are reset. To inhibit the LAST LEG FAULT messages from being displayed, the DISPLAY and RESET buttons are pressed at the same time.

C. Yaw Damper Servo (2)

(1) Each yaw damper servo drives the mechanical linkage to the power control actuators. Servo weight is approximately 7.5 pounds and the piston stroke is ±0.75 inches. Normal operating pressure at the pressure port is 3000 psi. The maximum return pressure is 100 psi. The servo rated load at 3000 psi is 1700 pounds maximum with the piston extended or retracted.

(2) Servo component parts consist of the input filter, electrohydraulic solenoid valve, electrohydraulic servo valve, output piston and shaft, and LVDT. A single electrical connector completes the servo engage voltage, command signal voltage, LVDT excitation voltage, and LVDT feedback voltage.

(3) Yaw Damper Servo Schematic (Fig. 5)

(a) Hydraulic pressure is maintained at the servo input port by the airplane hydraulic system. The hydraulic fluid is filtered by a 25 micron filter before entering the servo.

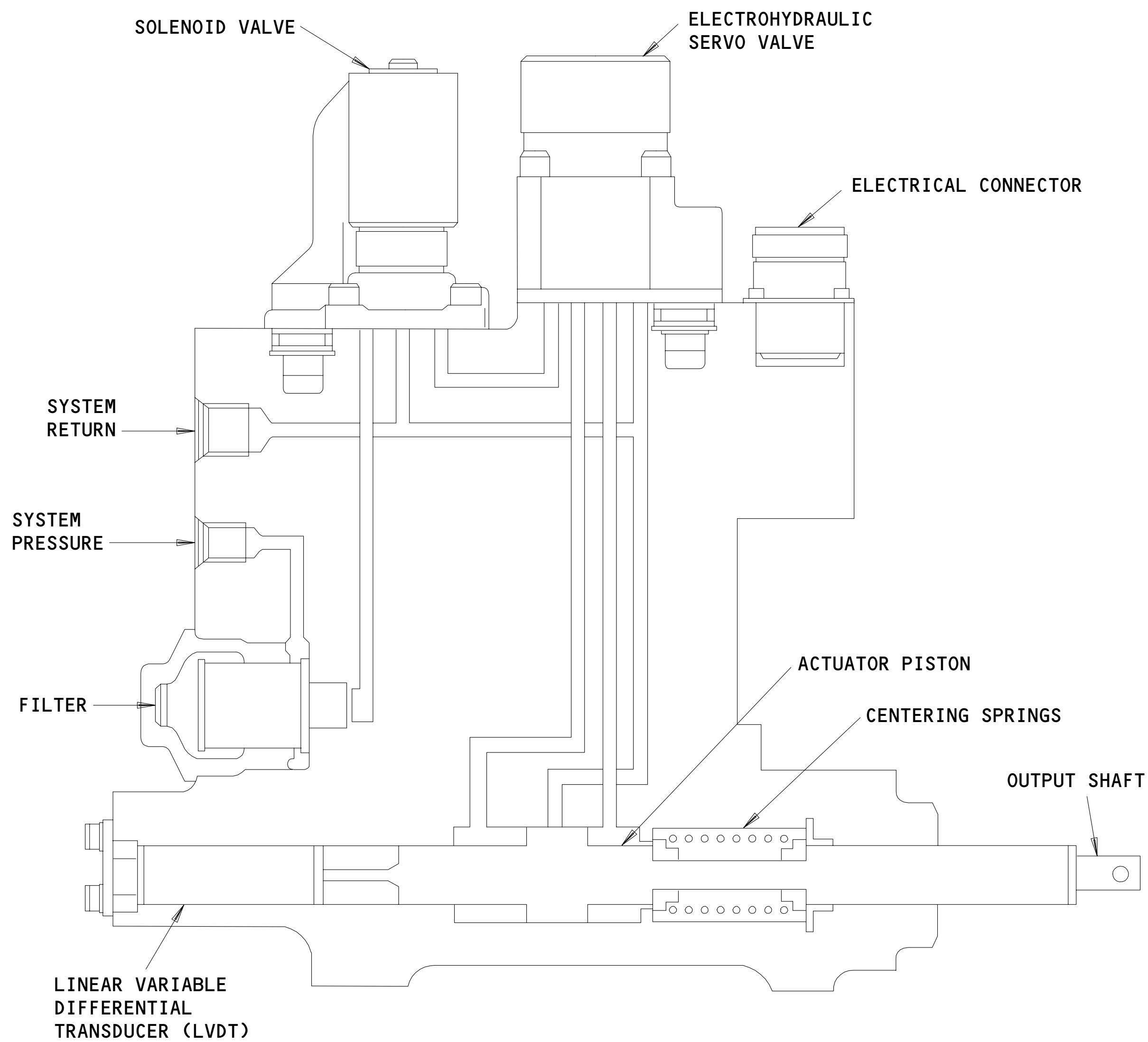
(b) Solenoid Valve

1) The solenoid valve is an electrically operated open-close valve that completes hydraulic pressure through the servo when the yaw damper system is armed. The 28 vdc valve-actuating voltage is controlled by the auto-disengage relay in the yaw damper module. The valve is installed with 4 bolts and sealed with a gasket plate.

(c) Electrohydraulic Servo Valve (EHSV)

1) One EHSV is installed on each yaw damper servo. It contains a sealed torque motor, a feedback spring, a projector jet and a piston. Hydraulic pressure through the valve can be applied to either of two output ports. When no error signal is applied to the torque motor, a small amount of hydraulic fluid flows through a flexpipe attached to the torque motor and out the projector jet. From the jet, equal pressure is applied to opposite ends of the piston, holding it at center and closing both output ports.

2) When an error signal is supplied, the motor armature rotates in proportion to the magnitude and direction of the input signal and moves the projector jet accordingly. The jet directs more hydraulic pressure to one end of the piston causing it to move. This opens the corresponding output port to complete hydraulic fluid to the servo. When the error signal is nulled out, the motor armature and jet return to center. This equalizes the pressures on the piston and, with the aid of the feedback spring, causes the piston to recenter and close both output ports.



Yaw Damper Servo Schematic
Figure 5

3) The EHSV is controlled by a rudder command signal from the yaw damper modules. In response to this command, the EHSV ports hydraulic pressure to the right or left side of the actuator piston. Feedback from the LVDT nulls out the module error signal and piston movement stops. The EHSV is installed with 4 bolts and sealed with a gasket plate.

(d) Actuator Piston Assembly

1) The actuator piston translates yaw damper input commands through the EHSV into mechanical positioning of the rudder. The actuator piston is normally centered by two springs and is moved right or left by hydraulic pressure from the EHSV. The springs center the piston when pressurization is released.

(e) Linear Variable Differential Transducer (LVDT)

1) The actuator piston LVDT functions as a linear follow-up transmitter for closing the loop around the EHSV. It is operated by the actuator piston. The LVDT provides a feedback voltage to the yaw damper modules that varies directly with linear motion. This feedback is used to null the servo loop. The LVDT is not a line replaceable unit.

(f) The electrical connection to the yaw damper servo is provided by one external and two internal connectors. One internal connector is used for the solenoid valve and the other for the EHSV. These electrical pins are mated when the solenoid or EHSV is bolted in position.

(g) The solenoid valve on each servo is controlled by the auto-disengage relay in the yaw damper modules. When 28 vdc is applied, the valve opens and arms the yaw damper system. The servo valve opens in proportion to a dc command signal from the yaw damper modules. The LVDT is excited by 26 vac, 400 Hz and provides a feedback voltage signal proportional to the linear position of the actuator piston.

D. 767-300 AIRPLANES;
Modal Suppression Accelerometer

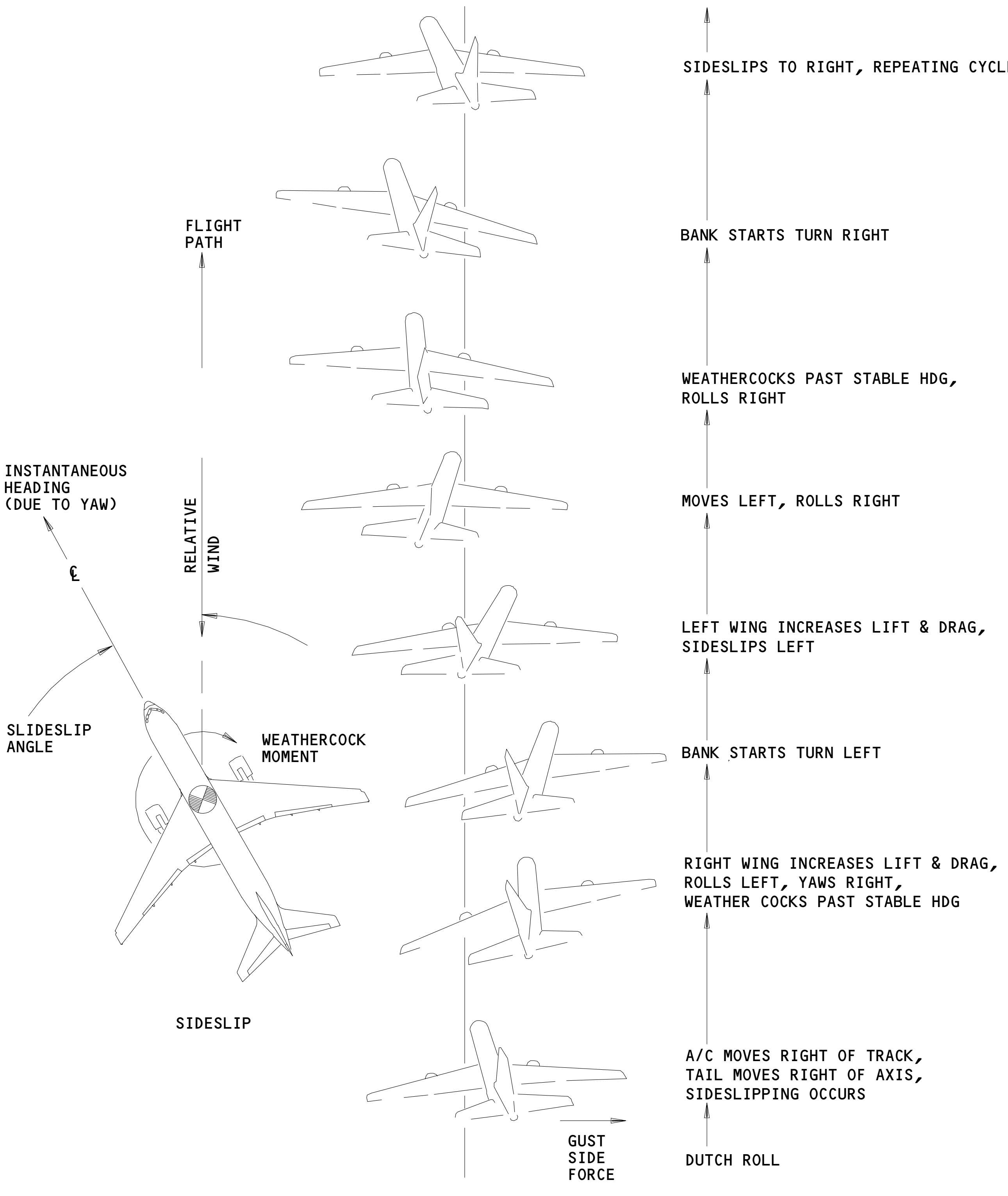
(1) The modal suppression accelerometer provides the yaw damper module with lateral acceleraton data of the aft end of the airplane. This data aids the yaw damper module in controlling the yaw and turn coordination of the airplane.

(2) The modal suppression accelerometer is composed of a detection circuit that consists of a cantilever surrounded on both sides by metal plates. The cantilever moves with the changing lateral velocity of the airplane. An output signal is produced by comparing the difference in the spacial gaps between the metal plates and the cantilever.

3. <u>Operation</u>

A. Functional Description

(1) Yaw Correction Requirements (Fig. 6)



Yaw Correction Requirements
Figure 6

(a) Sideslip

1) An airplane is defined to be in sideslip when the airplane heading is not aligned with the flight path. The sideslip angle is the angle between the airplane heading and the flight path. Airplane sideslip is caused by unstable air conditions, aerodynamic instability, or banking without coordinating rudder movement. When the airplane sideslips, it flies yawed and a side force tends to retard the sideways velocity. This damping force comes largely from increased drag on the faster moving wing, the fuselage, and vertical stabilizer. Damping and the vertical fin tends to correct the sideslip by "weathercocking" the airplane.

2) When a sideslip condition is present, the airflow on the forward wing goes from an outboard point on the leading edge to an inboard point on the trailing edge. The airflow on the trailing wing goes from an inboard point on the leading edge to an outboard point on the trailing edge. This results in an increase in angle of attack of the forward wing relative to that of the trailing wing. A difference in angle means a difference in lift on the two sides of the airplane, which induces a rolling moment.

(b) Dutch Roll

1) Dutch roll is a common oscillatory condition encountered due to low drag high speed aerodynamic design and turbulance which can be created by air mass instability. If this condition is allowed to persist, it causes considerable discomfort. If the aircraft experiences a side force causing yaw relative to the flight path, sideslip is produced. The change in direction of relative wind causes one wing to produce more lift than the other, thus inducing roll.

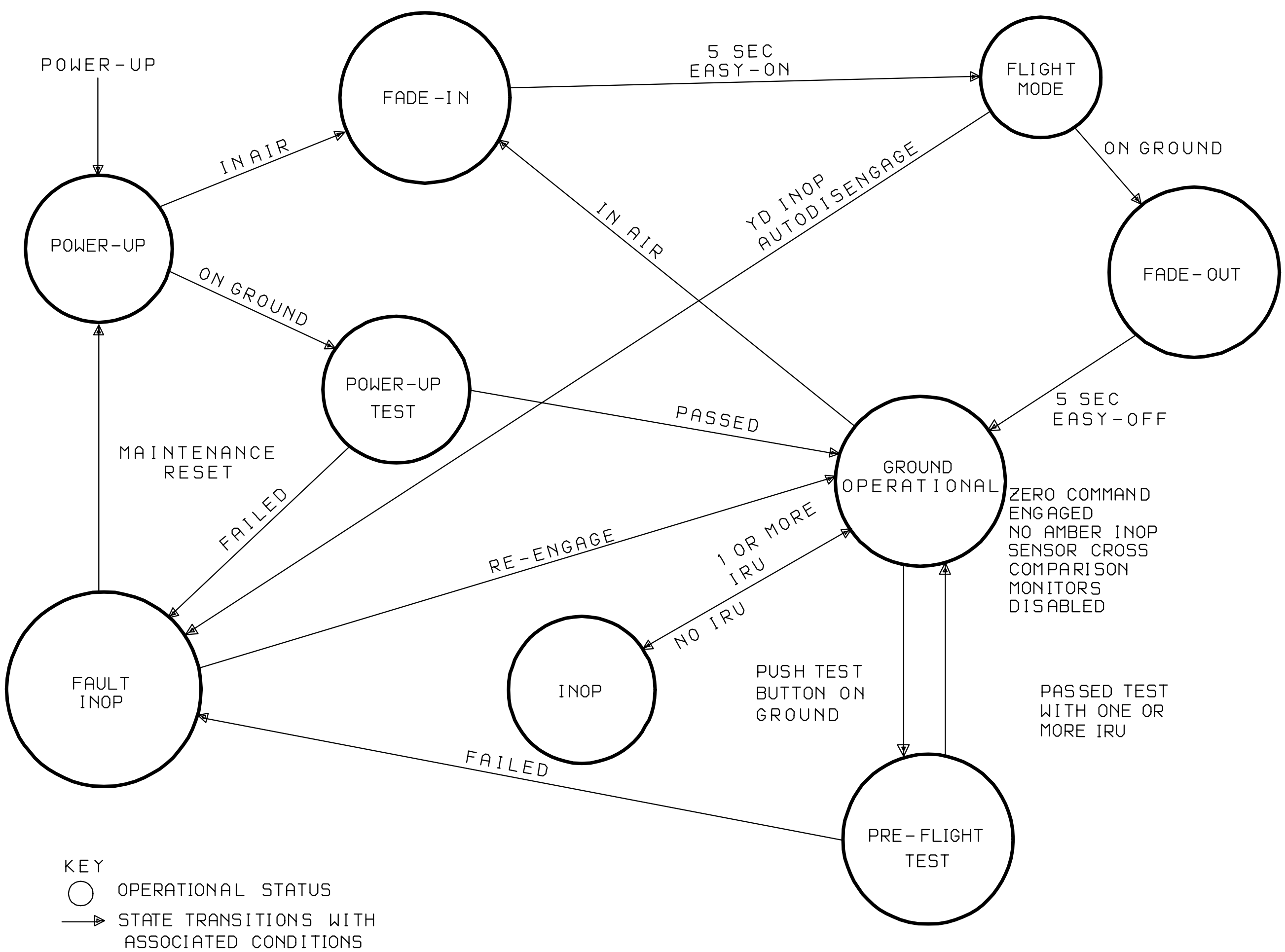
2) The yaw induced roll and weathercocking moment combine to induce yaw in the opposite direction of the roll. This produces an uncoordinated turn. The change in direction of relative wind causes one wing to produce more lift than the other, causing the wings to level. The airplane then banks back in the other direction and repeats the cycle with oscillating motion. The frequency of this oscillation is in the range from 0.5 to 0.1 Hz.

3) The yaw damper provides damping via the rudder to eliminate dutch roll.

(c) Turn Coordination

1) When the airplane is rolled into a turn, a yawing moment is generated by changing lift vectors due to the changes in relative wind. This moment produces a forward component (reduced drag) on the downgoing wing and a rearward component (increased drag) on the upgoing wing. The result of the yaw moments produce yaw opposite to the intended turn direction. Rudder deflections to counter roll induced yaw are required to achieve a coordinated turn.

(2) Yaw Damper Operational Modes (Fig. 7)
   (a) The yaw damper modules provide self-tests, full flight regime yaw damping control, and turn coordination. To accomplish this, each module has eight operational modes described in the following paragraphs.
   (b) Power-Up
      1) The power-up mode is entered when power is supplied to the yaw damper modules (power applied to bus). The CSEU power supply modules must be operative and the bus circuit breakers must be closed. The mode can be entered when on the ground or in the air. If the airplane is on the ground, the system proceeds to the power-up test. If power up is entered in the air, the system proceeds to the flight mode via a 5 second easy-on function. Upon power-up, the yaw damper modules initialize all interfaces, latches, command, and state variables to a safe state. The YDS monitors are reset; however, the fault ball indicators are latched and do not reset during power-up.
   (c) Power-Up Test
      1) The power-up test is performed when the YDS is on the ground. The test lasts approximately 20 seconds. The yaw damper INOP amber light is turned on and an EICAS message displayed during the power-up test. If a fault is detected, the INOP light remains on.
      2) The following tests are conducted during the power-up test:
         a) The channel fault disengagement test is performed for each channel (control and monitor). It monitors disengage relay status and reports a fault if the relay does not disengage.
         b) This causes a YDM fault message to be recorded.
         c) The INOP amber light interface check is performed for each channel (control and monitor).
         d) The hardware paths of both the control and monitor microprocessor yaw damper command coincidence monitor is checked.



Yaw Damper Operational Modes
Figure 7

e) Verifies that the yaw damper is not automatically disengaged.
f) The test performs a Random Access Memory (RAM) read/write test.

3) During the power-up test there is no rudder actuator movement. If a test failure is detected, the system goes to the FAULT/INOP mode (fault message is recorded and INOP light illuminates).
4) If the test is successful, the system proceeds to the ground operational mode.

(d) Ground Operational

1) The ground operational mode is entered when one of the following conditions is valid:
a) Successful completion of power-up or preflight test
b) Successful fault recovery and subsequent re-engagement
c) Sensor loss recovery to at least one IRU
d) Upon landing

2) During this mode, the yaw damper control law computes commands. However, the aircraft on ground status causes the control law output to convert commands to zero. Even though no output commands are possible, the INOP lamps do not illuminate during the ground operational mode. The sensor cross-comparison monitors are also disabled during this mode. The ground operational mode is exited when the airplane becomes airborne, all IRUs are lost, or when the preflight test is initiated.

(e) Preflight Test

1) The preflight test is entered when the ground operational mode is valid and the YAW DMPR test switch on the P61 panel is activated. The INOP indicator is illuminated throughout the test. The preflight test also initiates an IRS self-test. During the IRS self-test, the four annunciator lights for each system illuminate. The preflight test has the following automatic test sequence:
a) Commands the rudder to three degrees each direction and returns to center.
b) The actuator holds in each of the three positions for one second and performs a position test. The position test determines the difference between the commanded actuator position and the feedback position. The position test fails if the difference exceeds approximately 8 percent of the command value.
c) After 10 seconds, the test verifies proper IRU input to the system. If any IRU input is not within limits:
d) The corresponding IRU fault message is recorded.

2) If all the position tests and any one IRU data are satisfactory, the INOP indication is cancelled and the system returns to the ground operation mode. If any position test or all three IRU data tests fail, the INOP indication persists and the auto-disengage relay is actuated. If any IRU self-test fails, the FAULT light for that system remains illuminated.

(f) Fade-In

1) The fade-in mode is entered from the power-up mode or ground operational mode. The mode is entered if the airplane is airborne when power is first applied or after the airplane becomes airborne. The mode prevents sudden rudder transient response. The easy-on control law provides a linear rise in gain from zero to one in 5 seconds. When a gain of one is reached, the full flight mode is engaged.

(g) Flight Mode

1) The full flight mode is entered when the airplane is airborne and the fade-in function is completed. The flight mode performs turn coordination and yaw damping. Yaw correction and turn coordination are computed from yaw rate, lateral acceleration, indicated angle of attack, roll angle, roll rate, and true airspeed (or ground speed when true airspeed is not available). The flight mode also provides fault monitoring, self checks, command limiting, degraded operation as required, auto-disengage, and fault indications.

2) When no IRU data is available or a servo loop fault is detected, a YDS fault is latched and the associated channel automatically disengages. Each channel is also engaged and disengaged by the YDS manual controls. The flight mode automatically exits to the ground operational mode upon landing via the fade-out function.

(h) Fade-Out

1) The fade-out mode is entered directly upon airplane landing. The mode prevents a sudden stop of rudder control. The easy-off control law provides a linear decrease in gain from one to zero in a 5 second time period. The transition begins as soon as a ground state is established.

(i) Fault

1) The fault mode is entered as a result of the following:
   a) Failure of the power-up test
   b) Failure of the preflight test
   c) INOP type failure resulting in automatic disengagement

   d) when the fault mode is entered, the associated channel is disengaged, the INOP lamp is illuminated, and the corresponding fault messages are recorded.

2) The mode exits to the power up mode by a maintenance reset, or to the ground operational mode by automatic re-engagement if no INOP faults remain.

(3) Left and Right Yaw Damper Schematics (Fig. 8 and 9)

(a) Electrical Power

1) Yaw damper module operating power consists of +5 and ±15v dc parallel inputs from CSEU power supply modules 1L and 2L.

2) The 28v dc engage solenoid power is supplied by a YAW DAMPER circuit breaker via the ON switch. The 26v ac, 400-Hz LVDT excitation is supplied by the CSEU power supply modules.

(b) Inputs

1) Airplane Type Code

a) The airplane type code is a five wire ground combination with a sixth wire for parity.

b) The parity monitor checks parity and compares type code with the software issue code. This ensures that the correct software is used for the airplane type in which the yaw damper module is installed. If installed correctly, an APL TYPE CODE VALID signal is issued. A detected failure disengages the yaw damper, illuminates the INOP light, provides an EICAS yaw damper message, and

c) a YDM fault message is recorded.

2) Channel Code

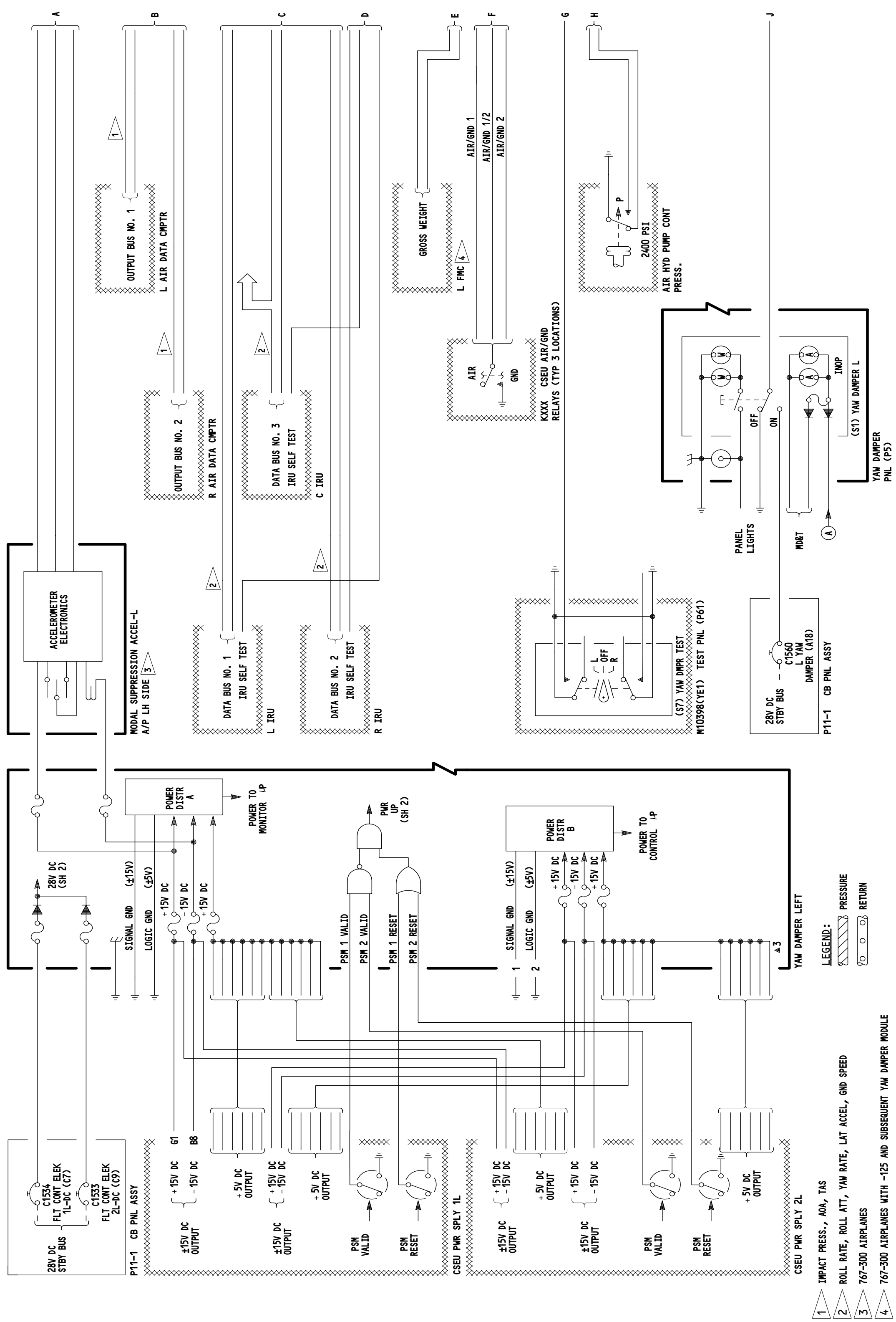
a) The channel code input is a two wire ground/open combination. One pin is grounded for the left installation of the module and the other pin is grounded for right module installation. The monitor checks to ensure only one channel code line is grounded.

3) Hydraulic Pressure Switch

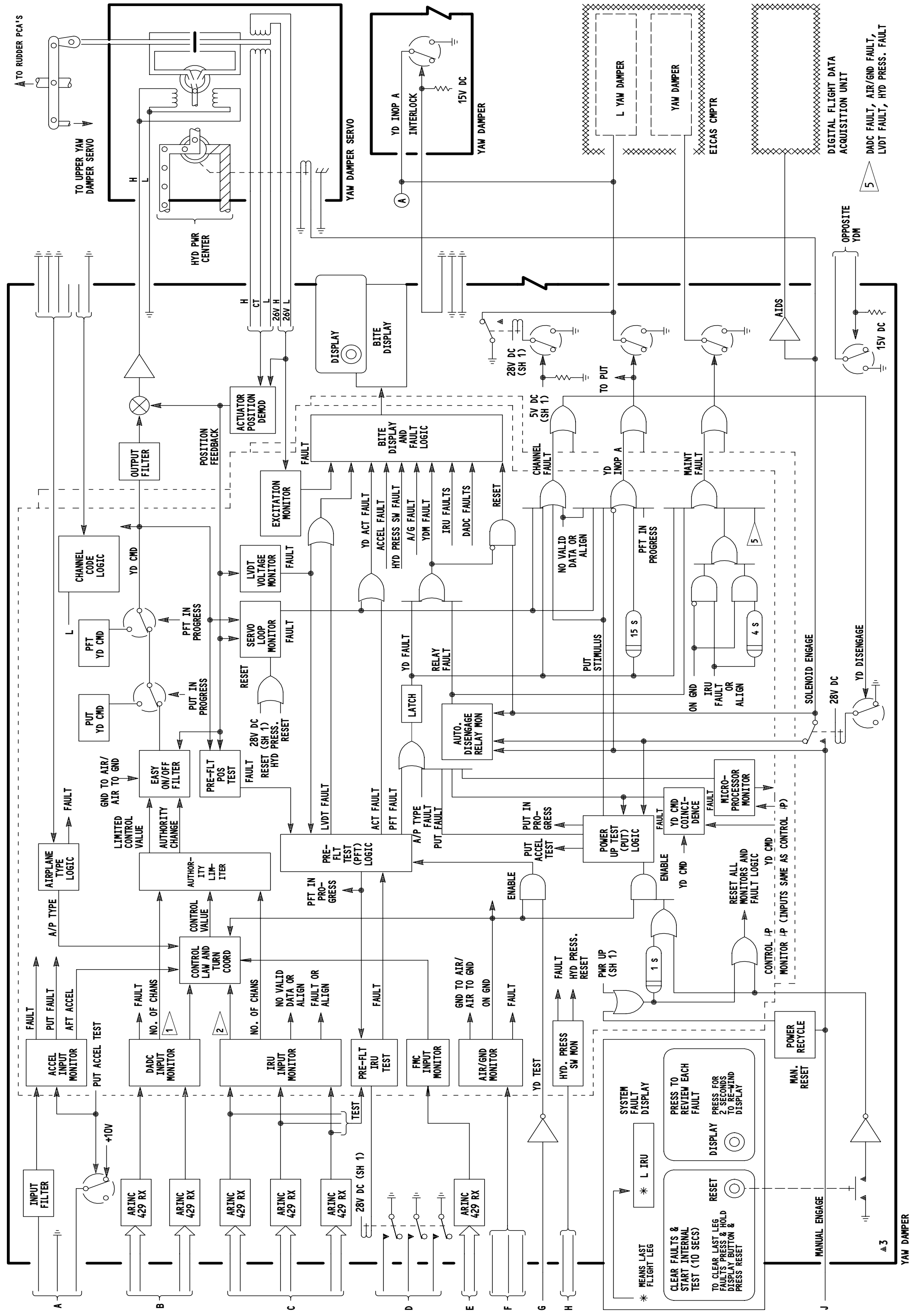
a) The hydraulic pressure switch is a two wire input. One wire is normally grounded at a time, indicating the hydraulic condition (high or low hydraulic pressure state). The failure mode is set if the hydraulic pressure HIGH signal agrees with the hydraulic pressure LOW signal for two seconds.

b) The HYD SWITCH fault message is cleared by the maintenance RESET switch.

c) The comparison monitor provides a VALID HI HYDRAULIC PRESS discrete signal and a HYD PRESS reset when high pressure is restored for 6 seconds after a low pressure indication.



Yaw Damper Schematic
Figure 8 (Sheet 1)



Yaw Damper Schematic
Figure 8 (Sheet 2)

4) Air/Ground Relays
   a) Air/ground information is received on three status lines from the air/ground relays. The status lines indicate: right gear aft tilt switch level (airplane on ground - through master A/G relay), right gear aft tilt switch level (airplane on ground - direct to YDM), and right gear forward tilt switch level (airplane on ground - direct to YDM).
   b) A triplex majority vote signal selector/monitor is used to determine A/G status. The monitor detects when one of three previously valid inputs is different from the selected value for more than 2 seconds. When the two remaining valid inputs disagree for more than 2 seconds following a first failure, a second failure is detected.
5) ADC
   a) The ADC parameter values for impact pressure, indicated angle of attack, and true airspeed are received on the ARINC 429 bus. Input signal processing includes active monitoring, parity monitoring, sign status matrix monitoring, and value substitution. Excessive signal failure rates or lack of activity result in an ADC input fault.
   b) A median value for each parameter is selected for use in the control law. Incoming parameter values are compared with the median value and are fault registered if the comparison is beyond the maximum limits.
6) IRU
   a) The IRU parameter values for roll rate, roll attitude, yaw rate, lateral acceleration and ground speed are received on the ARINC 429 bus. Input signal processing includes: activity monitoring, parity monitoring, sign status matrix monitoring, and value substitution. Excessive failure rates or lack of activity results in an IRU input fault.
   b) A median value for each parameter is selected for use in the control law. Incoming parameter values are compared with the median value and are registered as faults if the comparison is beyond the maximum limits.

7) 767-300 AIRPLANES;
FMC
   a) The FMC data (gross weight) is received on the ARINC 429 BUS. Input signal processing includes active monitoring, parity monitoring, sign status matrix monitoring, and value substitution. Excessive signal failure rates or lack of activity result in a FMC input fault.
   b) The FMC data (gross weight) is received from the left or the right FMC depending upon the position of the FMC source select switch. This data is received for use in the control law. An FMC LRU fault will alert the flight crew to use the data from the valid FMC. No signal selection logic for the FMC ARINC 429 data input is provided within the YDM.

8) Power Supply Modules (PSMs)
   a) The CSEU PSMs (No. 1 and 2 left) each furnish power-up reset and PSM valid signals. All four signals must be true to get a power-up reset of the yaw damper module.

9) Preflight Test Monitor
   a) The IRU parameter test values are monitored during the preflight test to ensure correct test values. Failure of this test generates an IRU input fault.

(c) Computing Channels
1) Each yaw damper module consists of two independent microprocessors with dissimilar software and instruction sets. Each microprocessor and associated software form a separate computing channel. These are a MONITOR channel and a CONTROL channel. Each microprocessor is programmed with the following software functions:
   a) Executive
   b) Power-up test and preflight test
   c) Input, output, and initialization
   d) Foreground software that includes input monitors, signal selectors, redundancy management, comparison monitors, degraded mode logic, control laws, and annunciation logic.
   e) Background monitors that conduct watchdog monitor tests and Read Only Memory (ROM) tests.

2) Control Law Computations
   a) The control law computations are enabled only if the APL type code is valid. The code ensures that the software in the module is correct for the airplane in which it is installed. The control law continually computes output commands. The output is zero on the ground (except during the preflight test) due to the disabled easy-on/easy-off function.

b) The control law computes output commands based on the data from the ADCs and IRUs. The command outputs from the two microprocessors are compared in coincidence monitors to detect when the two channel outputs do not agree. If non-coincidence is detected, the YDM fault is set.
c) Each computing channel of a yaw damper module performs self monitoring. The channel self monitoring consists of a background ROM check, executive activity monitor, and a watchdog monitor. The watchdog monitor checks for total Central Processing Unit (CPU) loss, operating speed, and erroneous ROM sum check values. If the watchdog monitor detects a failure, the YDM fault is set.

(d) Authority Limiting
1) The control law output command from each yaw damper module may never exceed ±3 degrees rudder. The authority is further limited as the number of valid ADC and IRU inputs decreases.

(e) Easy-On/Easy-Off
1) The control law output command is routed through a ramp gain schedule stage. The gain is zero when on the ground and one when in-air. With an air/ground relay indication that the airplane has taken off, the gain linearly rises from zero to one in 5 seconds. With an air/ground relay indication that the airplane has landed, the gain linearly decreases from one to zero in a 5 second time period. If an in-air power-up occurs, the fade-in begins immediately following the power up.

(f) Command Output
1) The control microprocessor output is converted to analog, smoothed, and routed to the yaw damper servo for rudder control. The output command operates the yaw damper electrohydraulic servo valve (EHSV). The EHSV directs hydraulic pressure flow to the servo actuator piston in proportion to the magnitude and sign of the command signal.
2) The EHSV receives hydraulic pressure when the engage solenoid is actuated by 28 vdc from the control panel ON switch (via the engage relay in the yaw damper module). The engage relay is closed unless an auto-disengage signal is generated, or there is a loss of operating power to the yaw damper module.
3) A Linear Variable Differential Transducer (LVDT) is physically attached to the actuator piston. The LVDT uses 400 Hz, 26 vac power for excitation. The LVDT output voltage reflects the position of the servo actuator piston. This data is routed back to the yaw damper module to complete the servo loop.
4) The LVDT feedback signal and the command output signal are compared in the yaw damper module. If the LVDT signal does not track the command signal, the servo fault latch is set.

5) This causes a YD ACT fault to be recorded.
6) The common mode sum voltage from the LVDT is also monitored. If it is less than 7 vrms,
7) a LVDT fault message is recorded.

(g) INOP Indication
1) The yaw damper control panel indicates INOP with the following conditions present:
a) During the power-up test
b) During the preflight test
c) With disengagement of a yaw damper channel (yaw damper servo solenoid valve not actuated)
2) Disengagement of a yaw damper channel occurs under the following conditions:
a) Manual disengagement from the control panel ON switch
b) Auto-disengagement resulting from: any YDM fault, power-up test failure, preflight test actuator failure, preflight test LVDT failure, no IRU data valid, or a servo fault during normal operation.
c) The interlock between modules is broken by one yaw damper module not being installed.

NOTE: One YDM must be installed to convert the opposite YDM open circuit to a ground for the INOP lamp.

(h) Each yaw damper also sends a discrete signal (yaw damper engage) to the digital flight data acquisition unit (DFDAU).

(i) Maintenance Reset
1) Pushing the reset button resets all the yaw damper module monitors and control logic.

(j) Yaw Damper Module Autoreset
1) The YDM fault latch is responsive to transient fault recovery of YDM fault monitors. YDM fault monitors capable of transient fault recovery are as follows:
a) Airplane type code
b) Yaw damper command coincidence
c) Activity monitor
d) Monitor or control watchdog monitor fault
2) The YDM fault latch resets if all YDM monitors exhibit no failures for 10 seconds continuously and within 15 seconds of detecting a YDM fault. Autoreset is limited to a maximum of four resets per flight. There is no limit on autoreset when the airplane is on ground.

3) The following faults are incapable of automatic recovery:
   a) Power-up reset YDM fault
   b) Preflight test YDM fault
   c) Background Read Only Memory (ROM) fault

(4) Yaw Damper Control Law Schematic (Fig. 9)

(a) Yaw Damping Command

1) Sideslip angle rate is synthesised by summing together yaw rate, lateral acceleration scheduled by true airspeed, roll attitude scheduled by true airspeed, and roll rate multiplied by body angle of attack. The lateral acceleration signal from the IRU's, is filtered and corrected for the effect of yaw rate and IRU displacement from center of gravity. Sideslip angle rate is gain scheduled by impact pressure to generate the yaw damping command.

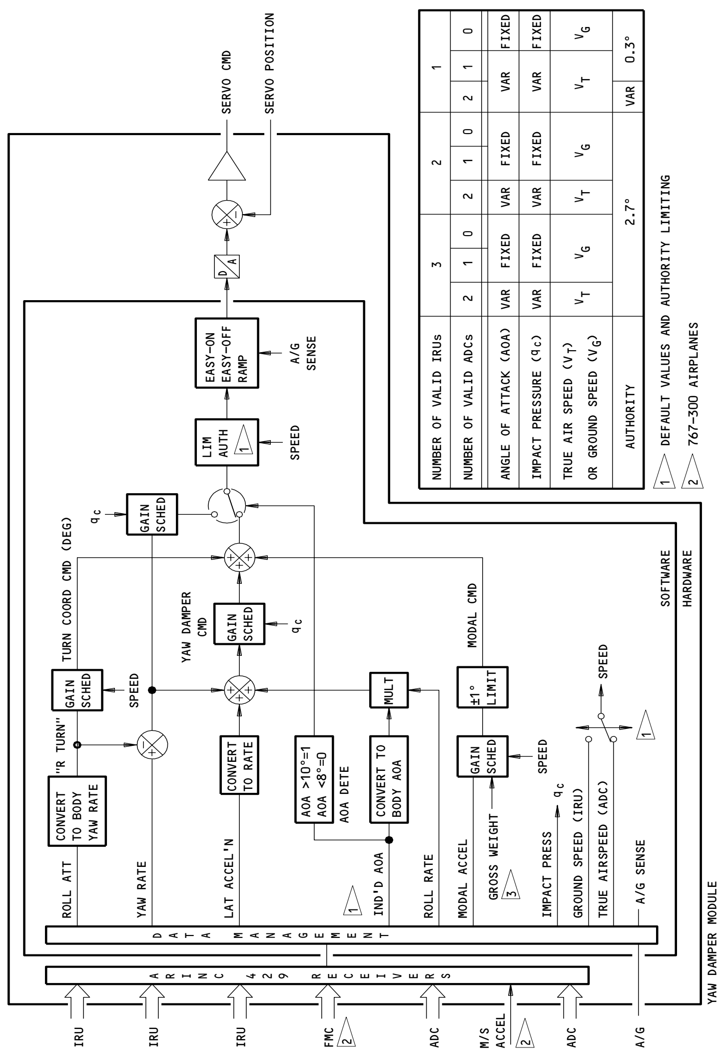
(b) Turn Coordination Command

1) The turn coordination command is derived from the roll attitude gain scheduled by true airspeed. This value is summed with the yaw damping CMD to form the output command processing.

(c) Output Command Processing

1) The output command is authority limited to fixed or variable amounts, depending on the availability of DADC and IRU inputs. Authority is in accordance with the table shown. The variable authority is programmed as a function of true airspeed. True airspeed is used when both DADCs are valid and at least one IRU is valid. For other combinations, ground speed is substituted for true airpseed. The authority is either variable, ±3.0 degrees, or ±0.3 degrees rudder as shown in the table. Commands are applied through an easy-on/easy-off gain schedule (5 second ramp) when the air/ground discrete indicates in-air (takeoff), or on-ground (landing), respectively.

2) Default values for indicated angle of attack (AOA), impact pressure, and true airspeed are used depending upon ADC and IRU availability. As an example, with one ADC and two IRUs available, indicated AOA becomes a constant 7.88 degrees, impact pressure becomes a constant 270 millibars, and ground speed is substituted for true airspeed.



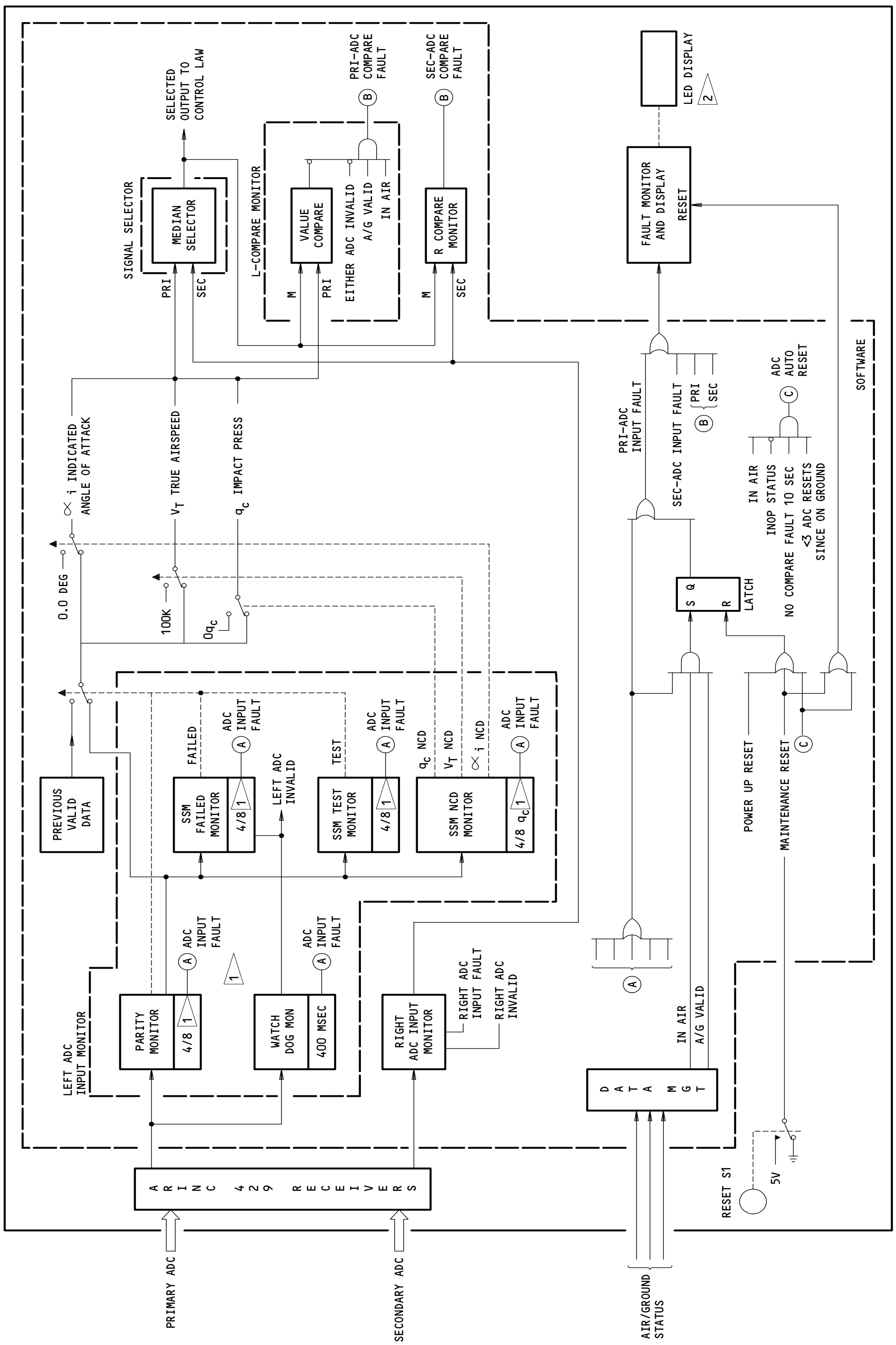
NUMBER OF VALID IRUs	3			2			1		
NUMBER OF VALID ADCs	2	1	0	2	1	0	2	1	0
ANGLE OF ATTACK (AOA)	VAR	FIXED		VAR	FIXED		VAR		FIXED
IMPACT PRESSURE (q_c)	VAR	FIXED		VAR	FIXED		VAR		FIXED
TRUE AIR SPEED (V_T) OR GROUND SPEED (V_G)	V_T	V_G		V_T	V_G		V_T		V_G
AUTHORITY	2.7°						VAR	0.3°	

1 DEFAULT VALUES AND AUTHORITY LIMITING
2 767-300 AIRPLANES

Yaw Damper Control Law Schematic
Figure 9

H57084

(5) ADC Input Signal Processing (Fig. 10)
   (a) Input Monitor
      1) The ADC input monitor consists of a watchdog monitor, parity monitor, and Sign Status Matrix (SSM) monitor. The watchdog monitor transmits an input fault signal if no air data information is received for 400 msec when required by the YDM. The parity monitor transmits a fault signal if a parity error is detected in 4 out of any 8 successive samples. For all samples with parity error, the incoming values are discarded and the last previous valid values are used.
      2) The SSM monitor provides an input fault signal if incoming data is indicated as FAILED for 4 out of 8 successive samples. For all samples indicating FAILED, the incoming values are discarded and the last previous valid value is used. A fault signal is also provided if impact pressure is indicated as No Computed Data (NCD) for 4 out of 8 successive samples. For all samples indicating NCD, the incoming value is discarded and the last previous valid value is used. Any angle of attack sample indicated as NCD, is used as valid data. If a true airspeed sample is indicated as NCD, 100 knots is substituted for the sample value. If incoming data is indicated as TEST for 4 out of 8 successive samples, a fault signal is transmitted. For all samples indicating TEST, the incoming values are discarded and the last previous valid value is used.
   (b) Signal Selector
      1) A median value of parameters from both ADCs is selected for use in the control law computations. Incoming parameter values are compared to the respective signal selector output. If the difference is greater than the specified values shown below for one second, the compare fault is set. (Airplane must be 200 milliseconds after the ground to air transition, A/G discrete valid, and neither ADC invalid.) Following a ADC fault, data from the remaining valid ADC may be used.



1. FOUR OUT OF EIGHT SUCCESSIVE SAMPLES CAUSE FAILURE STATUS
2. POSSIBLE DISPLAYS FOR ADC FAULTS ARE: L ADC, R ADC, ADC DIFF

ADC Input Signal Processing
Figure 10

812999

2) The maximum deviation allowed between the ADC input values and the median ADC value follows:
   a) True airspeed - 60 knots
   b) Angle of attack - 5 degrees
   c) Impact pressure - 27 millibars

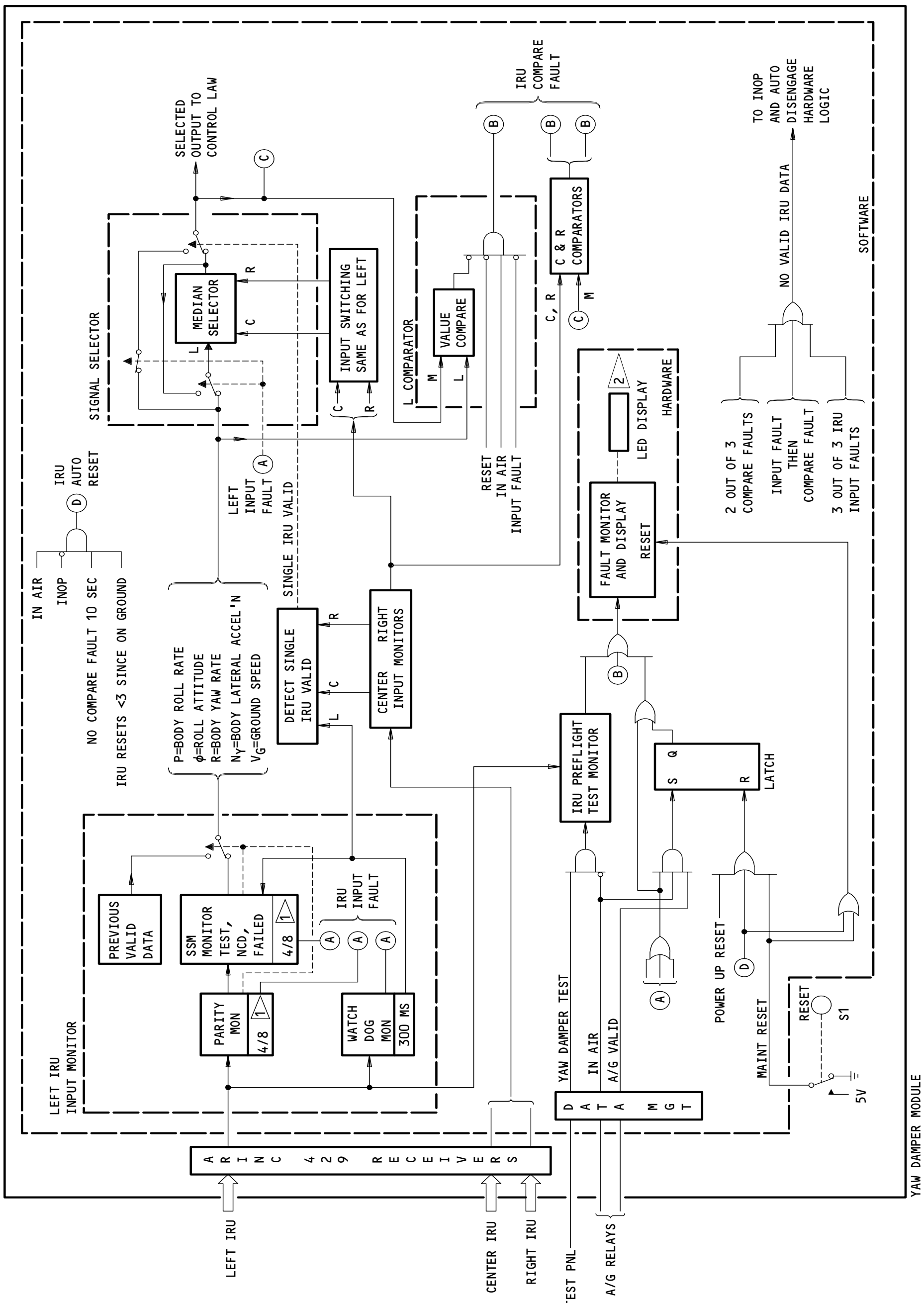
(c) Fault Annunciation

1) Any ADC input or comparison fault causes a corresponding ADC fault message to be recorded and sets the ADC software monitors.
2) If the airplane is in the air and the air/ground (A/G) discrete is valid, input faults are latched. The latch can be reset by the power-up reset, maintenance reset, or DADC automatic reset.
3) The DADC autoreset is generated with the airplane in air. If sensor data within monitor threshold is detected for 10 seconds continuously, the DADC fault latch is reset, provided there have been no more than 4 resets since the airplane was last on ground. Maintenance reset is generated by pressing the RESET switch on the yaw damper module.

(6) IRU Input Signal Processing (Fig. 11)

(a) Input Monitor

1) The IRU input monitor consists of a watchdog monitor, parity monitor, preflight test monitor, and Sign Status Matrix (SSM) monitor. The watchdog monitor input fault is set when no IRU information is received for 300 msec. when required by the YDM. The parity monitor input fault is set if a parity error is detected in four out of any 8 successive samples. For all samples with a parity error, the incoming values are discarded and the last previous valid values are used.
2) The SSM monitor input fault is set when incoming data is indicated as FAILED, TEST or NCD for 4 out of 8 successive samples. For all samples indicating FAILED, TEST or NCD, the incoming values are discarded and the last previous valid values are used. The preflight test input monitor fault is set if the test values from the IRUs are incorrect. The monitor is active only during the airplane preflight test.



1 FOUR OUT OF EIGHT SUCCESSIVE SAMPLES INDICATE MONITOR STATUS TRUE

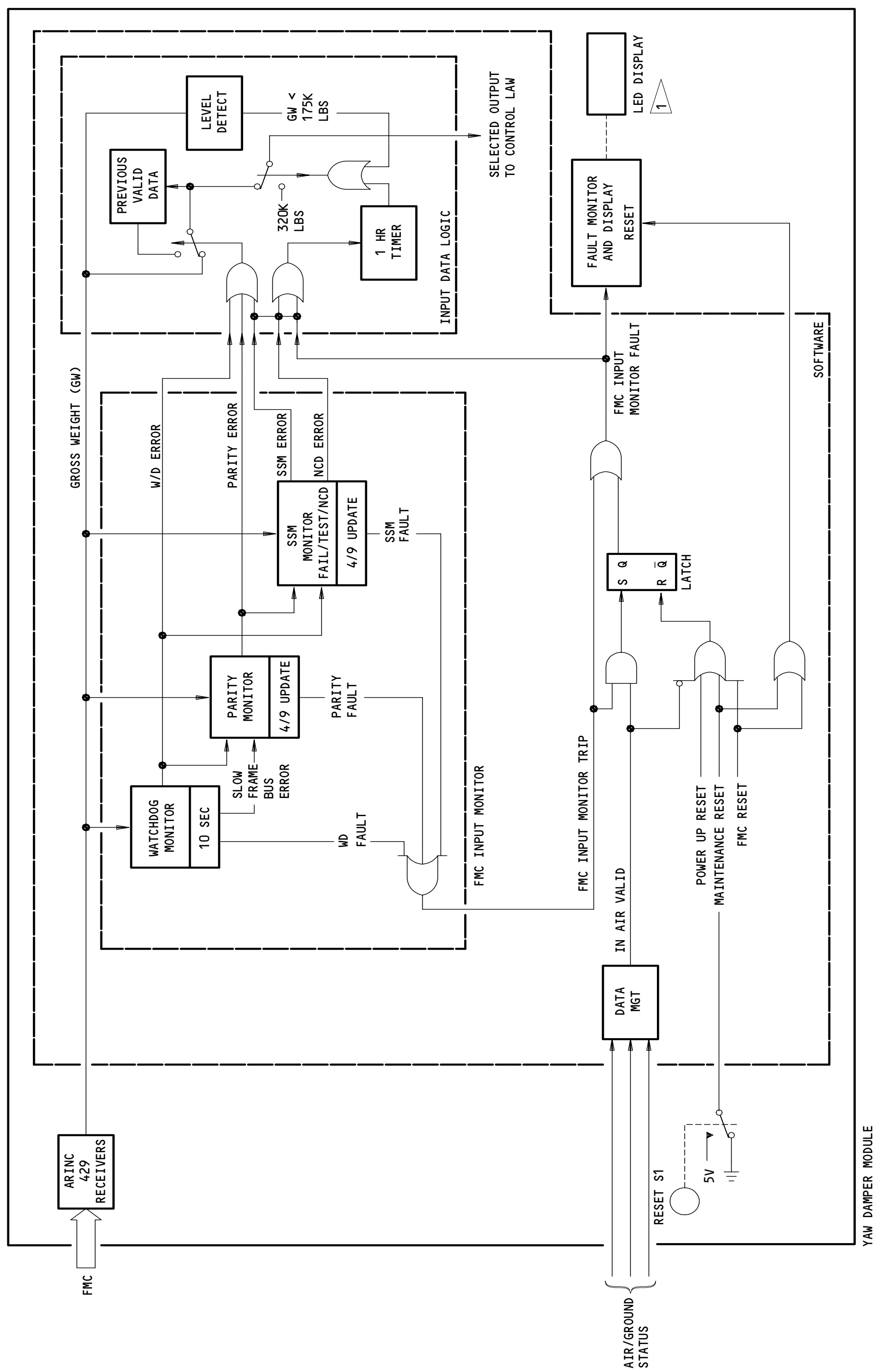
2 POSSIBLE DISPLAYS FOR IRU FAULTS ARE: L IRU, R IRU, C IRU, IRU DIFF

IRU Input Signal Processing
Figure 11

012999

(b) Signal Selector
  1) A median value of parameters from all three IRUs is selected for use in control law computations. If one IRU fails, that input is not used. The two remaining IRU inputs and the last valid median value are used for median selection. With only one IRU remaining valid, median selection is avoided and that single input is used. Each parameter input value is compared to the selected median value. Differences exceeding specified amounts for more than one second result in a comparison fault latch and the affected IRU being declared failed. (The airplane must be 200 millisecond after the ground to air transition, no input fault for that IRU and A/G discrete valid.)
  2) The maximum deviation allowed between IRU input values and the median IRU value follows:
    a) Roll attitude - 5 degrees
    b) Body axis roll rate - 2.5 deg/sec
    c) Body axis yaw rate - 2.5 deg/sec
    d) Body axis lateral acceleration - 0.1 g's
    e) Ground speed - 60 knots

(c) Fault Annunciation
  1) If any IRU input, comparison, or preflight test monitor fault is detected, the corresponding IRU fault message is recorded.
  2) If the airplane is airborne, input faults are latched. The monitors can be reset by the power-up reset, IRU auto-reset, or the maintenance reset.
  3) IRU data is declared unacceptable if:
    a) Two out of three IRUs have comparison faults
    b) An input fault and a comparison fault are detected.
    c) All IRUs have input faults
  4) When IRU data is declared unacceptable, the yaw damper channel goes INOP and is auto-disengaged. An IRU autoreset is generated with the airplane in air. If sensor data within monitor threshold is detected for 10 seconds continuously, the IRU fault latch is reset, provided there have been no more than four resets since the airplane was on the ground. The maintenance reset is generated by the YDM front panel RESET switch.



1 THE DISPLAY FOR FMC FAULT IS: FMC INPUT

FMC Input Signal Processing
Figure 12

A33609

(7) 767-300 AIRPLANES;
FMC Input Signal Processing (Fig. 12)

(a) Input Monitor

1) The FMC input monitor consists of a watchdog monitor, parity monitor and Sign Status Matrix (SSM) monitor. The watchdog monitor transmits an input fault signal if no FMC data is received for more than 10 seconds when required by the yaw damper module. The parity monitor transmits a fault signal if a parity error is detected in 4 out of 9 successive samples. For all samples with the parity error, the incoming values are discarded and the last previous values are used.

2) The SSM monitor provides an input fault signal if the incoming data is indicated as FAILED for 4 out of 9 successive samples. For all the samples indicating FAILED, the incoming values are discarded and the last previous valid value is used. A fault signal is also provided if the incoming data is indicated as No Computed Data (NCD) for 4 out of 9 successive samples. For all the samples indicating NCD, the incoming value is discarded and the last previous valid value is used. If the incoming data is indicated as TEST for 4 out of 9 successive samples, a fault signal is transmitted. For all samples indicating TEST, the incoming values are discarded and the last previous valid value is used.

(b) Fault Annunciation

1) If any FMC input fault is detected, the fault message (FMC INPUT) is recorded.

2) If the air/ground (A/G) discrete is valid (airplane in the air) and the FMC Input Monitor Fault is set, then the FMC Input Monitor Fault is latched, and the default FMC data (gross weight) is used. The latch can be reset by power up reset, maintenance reset, or FMC reset.

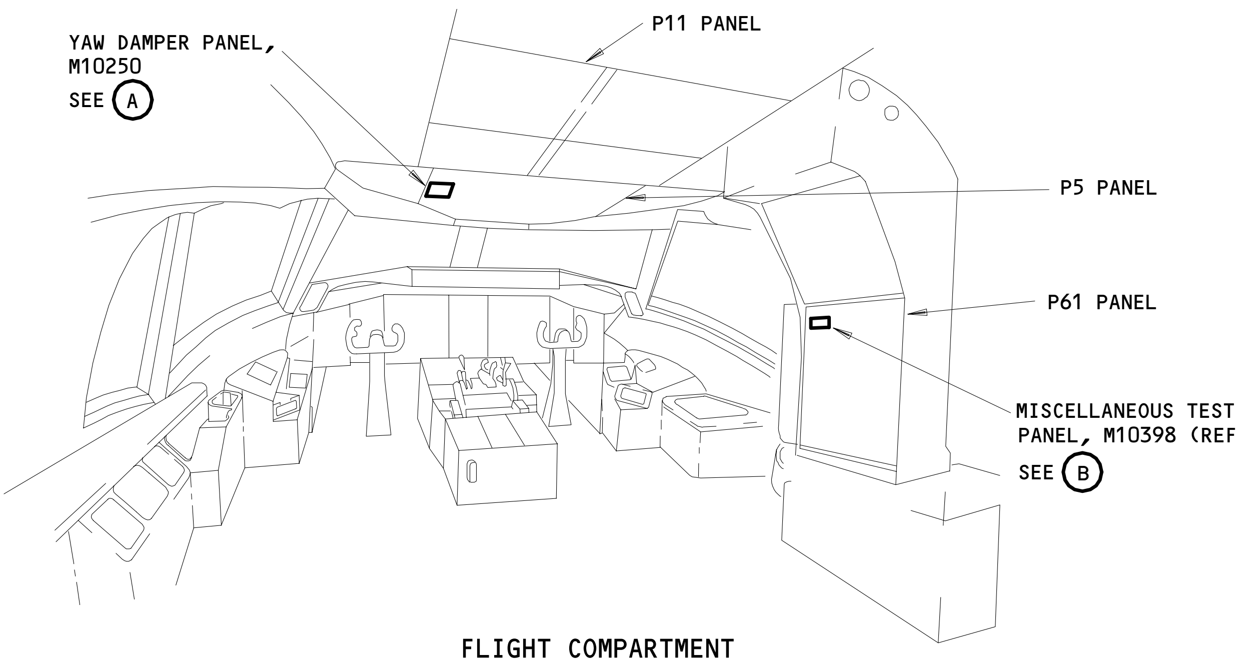
YAW DAMPER SYSTEM

COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
ACCELEROMETER - MODAL SUPPRESSION L, M1458 [1]	2	1	822 AFT CARGO DOOR, AFT CARGO COMPT CEILING	*
ACCELEROMETER - MODAL SUPPRESSION R, M1459 [1]	2	1	822 AFT CARGO DOOR, AFT CARGO COMPT CEILING	*
CIRCUIT BREAKER -	1		FLT COMPT, P11	
FLT CONT ELEC 1L AC, C1538		1	11C6	
FLT CONT ELEC 1L DC, C1534		1	11C7	
FLT CONT ELEC 1R AC, C1536		1	11G17	
FLT CONT ELEC 1R DC, C1531		1	11G18	
FLT CONT ELEC 2L AC, C1537		1	11C8	
FLT CONT ELEC 2L DC, C1533		1	11C9	
FLT CONT ELEC 2R AC, C1535		1	11G26	
FLT CONT ELEC 2R DC, C1532		1	11G27	
YAW DAMPER L, C1560		1	11A18	*
YAW DAMPER R, C1561		1	11F34	*
COMPUTER - (FIM 31-41-00/101)				
EICAS L, M10181				
EICAS R, M10182				
COMPUTER - (FIM 34-12-00/101)				
AIR DATA L, M100				
AIR DATA R, M101				
COMPUTER - (FIM 34-51-00/101) [1]				
FLIGHT MANAGEMENT L, M134				
FLIGHT MANAGEMENT R, M135				
MODULE - (FIM 27-09-00/101)				
LEFT POWER SUPPLY 1, M536				
LEFT POWER SUPPLY 2, M537				
RIGHT POWER SUPPLY 1, M538				
RIGHT POWER SUPPLY 2, M539				
MODULE - L YAW DAMPER, M522	2	1	119AL, MAIN EQUIP CTR, E1-1	22-21-04
MODULE - R YAW DAMPER, M523	2	1	119AL, MAIN EQUIP CTR, E2-1	22-21-04
PANEL - (FIM 28-43-00/101)				
MISC TEST, M10398				
PANEL - YAW DAMPER, M10250	1	1	FLT COMPT, P5	22-21-01
RELAY - (FIM 31-01-33/101)				
AIR/GND SYS NO. 2, K518				
RELAY - (FIM 31-01-36/101)				
AIR/GND BAT SYS NO. 1, K529				
AIR/GND SYS NO. 1, K199				
RELAY - (FIM 31-01-37/101)				
AIR/GND SYS NO. 2, K293				
SERVO - LEFT YAW DAMPER, M510	2	1	324EL, VERT STAB, APL R SIDE	22-21-02
SERVO - RIGHT YAW DAMPER, M509	2	1	324EL, VERT STAB, APL L SIDE	22-21-02
SWITCH - YAW DMPR TEST	1	1	FLT COMPT, P61, MISC TEST PNL	*
UNIT - (FIM 34-21-00/101)				
INERTIAL REFERENCE C, M160				
INERTIAL REFERENCE L, M159				
INERTIAL REFERENCE R, M161				
VALVE - YDS ELECTROHYDRAULIC SERVO	2	1	324EL, EA YAW DAMPER SERVO	22-21-03
VALVE - YDS ELECTROHYDRAULIC SOLENOID	2	1	324EL, EA YAW DAMPER SERVO	22-21-03

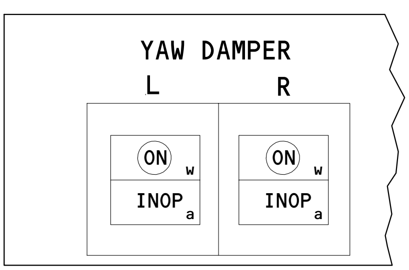
* SEE THE WDM EQUIPMENT LIST

[1] 767-300 AIRPLANES

Yaw Damper System - Component Index
Figure 101

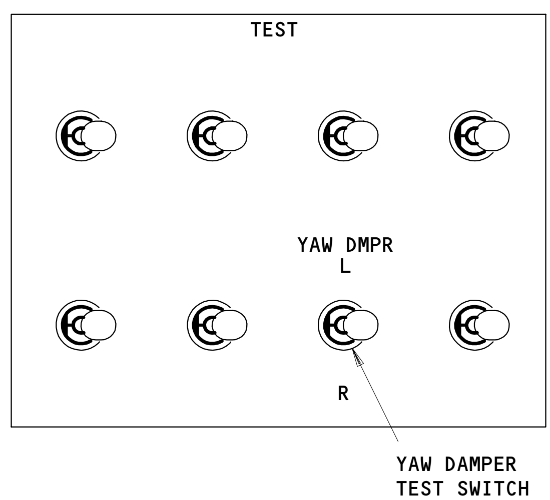


FLIGHT COMPARTMENT



YAW DAMPER PANEL, M10250

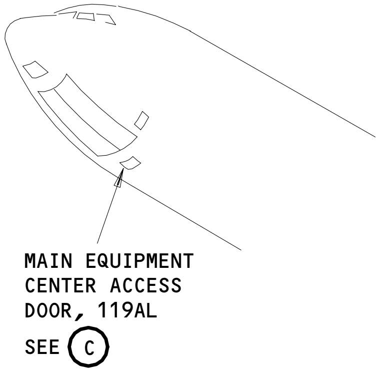
A

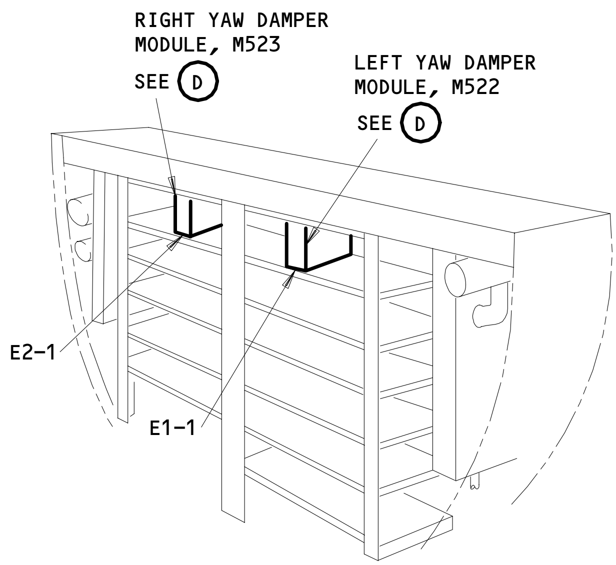


MISCELLANEOUS TEST PANEL, M10398 (REF)

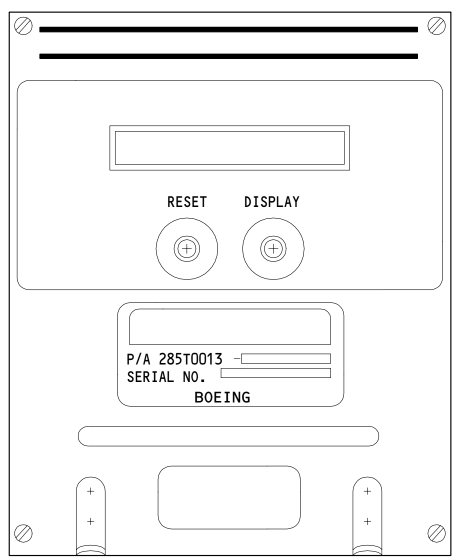
B

Yaw Damper System - Component Location
Figure 102 (Sheet 1)



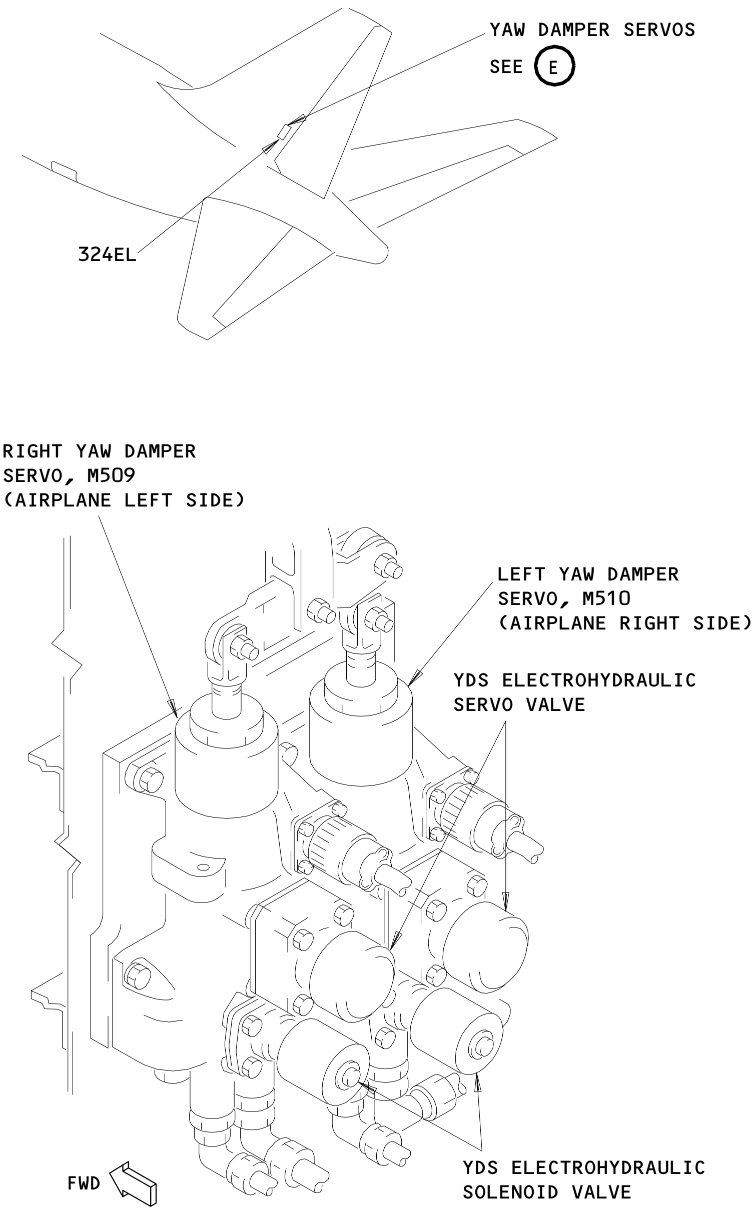


MAIN EQUIPMENT CENTER
C



LEFT OR RIGHT YAW DAMPER MODULE, M522 OR M523
D

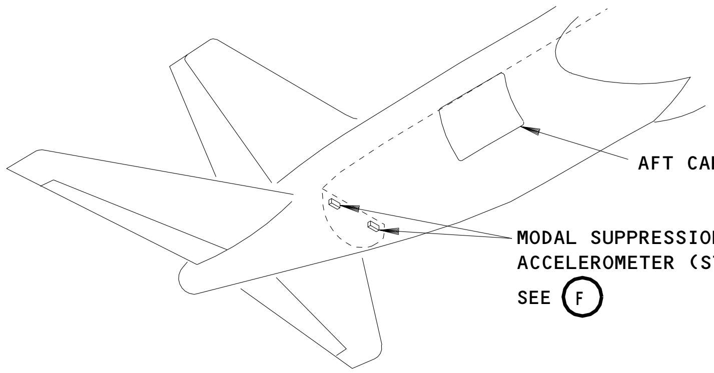
Yaw Damper System - Component Location
Figure 102 (Sheet 2)

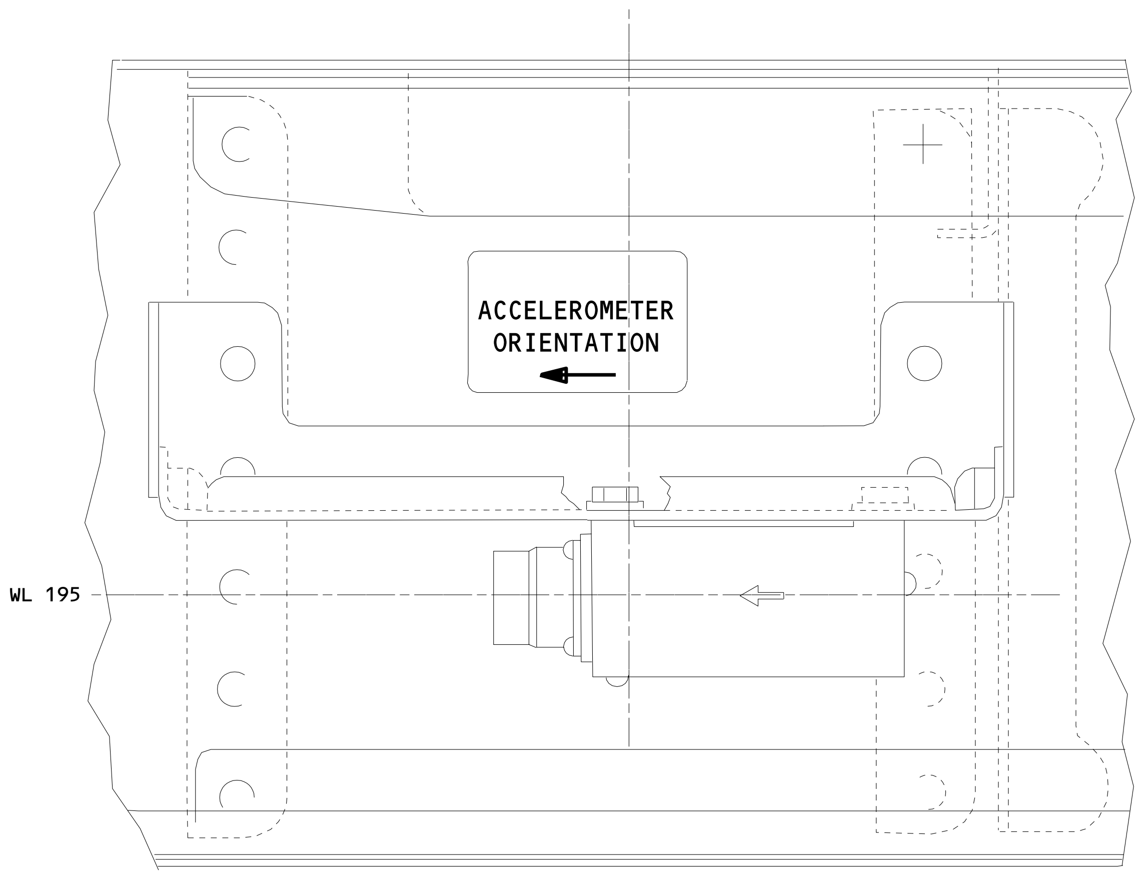


YAW DAMPER SERVOS

E

Yaw Damper System - Component Location
Figure 102 (Sheet 3)





MODAL SUPPRESSION ACCELEROMETER
(VIEW IN THE FORWARD DIRECTION)
(RIGHT SIDE IS SHOWN, LEFT SIDE IS EQUIVALENT)
F

Yaw Damper System - Component Location
Figure 102 (Sheet 4)

YAW DAMPER SYSTEM - ADJUSTMENT/TEST

1. General
   A. The paragraphs that follow give an operational test and a system test of the Yaw Damper System. No external test equipment is needed for these tests. The operational test is a preflight test of the Yaw Damper System. The system test is a detailed check of the Yaw Damper System interfaces to make sure the system performs correctly.

TASK 22-21-00-715-895

2. Operational Test - Yaw Damper System
   A. General
      (1) The operational test uses only the built-in test equipment. The YAW DMPR test switch on right side panel, P61 is pushed to start the operational test. The test makes sure the yaw damper modules, the yaw damper servos, and the switches on the yaw damper panel operate correctly.
   B. References
      (1) AMM 24-22-00/201, Electrical Power - Control
      (2) AMM 27-21-00/501, Rudder and Rudder Trim Control System
      (3) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
      (4) AMM 31-41-00/501, Engine Indication and Crew Alerting System
      (5) AMM 32-09-02/201, Air/Ground Relays
      (6) AMM 33-16-00/501, Master Dim and Test
      (7) AMM 34-12-00/501, Air Data Computing System
      (8) AMM 34-21-00/501, Inertial Reference System
      (9) AMM 34-61-00/501, Flight Management Computer system
   C. Access
      (1) Location Zones
         119/120 Main Equipment Center
         211/212 Control Cabin

      (2) Access Panel
         119AL Main Equipment Center

   D. Prepare for Test

      S 865-002
      (1) Supply electrical power (AMM 24-22-00/201).

      S 865-003
      (2) Make sure that these circuit breakers on the overhead panel, P11, are closed:
         (a) 11A18, YAW DAMPER L
         (b) 11C6, FLT CONT ELEC 1L AC
         (c) 11C7, FLT CONT ELEC 1L DC
         (d) 11C8, FLT CONT ELEC 2L AC
         (e) 11C9, FLT CONT ELEC 2L DC
         (f) 11F34, YAW DAMPER R
         (g) 11G17, FLT CONT ELEC 1R AC

(h) 11G18, FLT CONT ELEC 1R DC
(i) 11G26, FLT CONT ELEC 2R AC
(j) 11G27, FLT CONT ELEC 2R DC

S 865-004

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(3) Pressurize the left and center hydraulic systems (AMM 29-11-00/201).

S 865-006
(4) Make sure that these P11 panel circuit breakers are closed:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11H17, FLT CONT SHUTOFF TAIL L
(d) 11H18, FLT CONT SHUTOFF TAIL CENTER
(e) 11H27, FLT CONT SHUTOFF TAIL R

S 985-007
(5) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches, on the P61 panel, in the ON position.

S 985-008
(6) Put the two STAB TRIM switches on control stand panel, P10, in the NORMAL position.

S 285-009
(7) Make sure these systems operate:
(a) Rudder and Rudder Trim Control System (AMM 27-21-00/501).
(b) Air Data Computing System (AMM 34-12-00/501).
(c) Inertial Reference System (AMM 34-21-00/501).
(d) Air/Ground Relays (AMM 32-09-02/201).
(e) Master Dim and Test System (AMM 33-16-00/501).
(f) Engine Indication and Crew Alerting System (AMM 31-41-00/501).

(g) 767-300 AIRPLANES;
Flight Management Computer System (AMM 34-61-00/501)

S 985-010

(8) Push the STATUS switch on the EICAS display select panel on the P9 panel.

S 985-011

(9) Rotate the rudder trim switch on the P8 panel to put the rudder in the neutral position. (Rudder is in neutral when the two sides of the rudder trailing edge aligned in the rudder index plate groove.)

E. Test Yaw Damper System Operation

S 985-012

(1) Push the RESET button on the faceplate of the left and right yaw damper modules. Stop for 30 seconds before you continue.

S 285-013

(2) Make sure that the left and right YAW DAMPER ON switch/lights on the yaw damper panel are on.

<u>NOTE</u>: If the lights are not on, push the L and R YAW DAMPER ON switches.

S 285-014

(3) Make sure that the YAW DAMPER INOP L and INOP R lights are off.

S 985-015

(4) Put the YAW DMPR test switch on the right side panel, P61, in the L position and hold until the YAW DAMPER INOP L light comes on. Return the switch to the center. The light will stay on for the remainder of the test. (The YAW DAMPER INOP R test light may also come on.)

S 285-016

(5) Make sure that the rudder position indicator on the lower EICAS display shows that the rudder moves approximately 3 degrees trailing edge right, then approximately 3 degrees trailing edge left, and then returns to center in less than 10 seconds.

S 285-017

(6) Make sure that the YAW DAMPER INOP L light goes off in less than 15 seconds after the action of two steps before.

S 985-018

(7) Put the YAW DMPR test switch in the R position and hold until YAW DAMPER INOP R light comes on. Return the switch to the center. The light will stay on for the remainder of test. (The YAW DAMPER INOP L light may come on.)

S 285-019
(8) Make sure that the rudder position indicator shows that the rudder moves approximately 3 degrees trailing edge right, then approximately 3 degrees trailing edge left, and then returns to center in less than 10 seconds.

S 285-020
(9) Make sure that the YAW DAMPER INOP R light goes off in less than 15 seconds after the action of the two steps before.

S 985-024
(10) Push the DISPLAY button on the left YAW DAMPER module and make sure that the message NO FAULTS is shown on the left YAW DAMPER module's fault display.

S 985-930
(11) Push the DISPLAY button on the right YAW DAMPER module and make sure that the message NO FAULTS is shown on the right YAW DAMPER module's fault display.

F. Put the Airplane Back to Its Usual Condition

S 865-026
(1) Remove pressure from the left and center hydraulic systems.

S 865-027
(2) Remove electrical power if is not necessary.

TASK 22-21-00-735-237
3. System Test - Yaw Damper System
A. General
(1) The system test makes sure these system interfaces operate correctly:
(a) Modal Suppression Accelerometer
(b) 767-300 AIRPLANES;
Flight Management Computers
(c) Yaw Damper servos
(d) Inertial Reference Units
(e) Air Data Computers
(f) EICAS
(g) Air/Ground relays
(h) Hydraulic pressure switches and sources
(i) System electrical power sources
(2) The system test also makes sure these conditions occur:
(a) Channel fault disengagement
(b) Disengage relay
(c) Ability to detect servo loop faults
(d) Operation of monitors
B. Equipment
(1) Four Landing Gear Proximity Sensor Actuators, P/N A32102-3

C. References
   (1) AMM 24-22-00/201, Electrical Power - Control
   (2) AMM 27-21-00/501, Rudder and Rudder Trim Control System
   (3) AMM 27-61-00/201, Spoiler/Speedbrake Control System
   (4) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
   (5) AMM 31-41-00/501, Engine Indication and Crew Alerting System
   (6) AMM 32-09-02/201, Air/Ground Relays
   (7) AMM 33-16-00/501, Master Dim and Test
   (8) AMM 34-12-00/501, Air Data Computing System
   (9) AMM 34-21-00/501, Inertial Reference System
   (10) AMM 34-61-00/501, Flight Management Computer System

D. Access
   (1) Location Zones
       119/120 Main Equipment Center
       211/212 Control Cabin

   (2) Access Panel
       119AL Main Equipment Center

E. Prepare for Test

   S 865-238
   (1) Supply electrical power (AMM 24-22-00/201).

   S 285-239
   (2) Make sure that these P11 panel circuit breakers are closed:
       (a) 11A18, YAW DAMPER L
       (b) 11C6, FLT CONT ELEC 1L AC
       (c) 11C7, FLT CONT ELEC 1L DC
       (d) 11C8, FLT CONT ELEC 2L AC
       (e) 11C9, FLT CONT ELEC 2L DC
       (f) 11C12, STAB TRIM SHUTOFF L
       (g) 11C13, STAB TRIM SHUTOFF CENTER
       (h) 11F34, YAW DAMPER R
       (i) 11G17, FLT CONT ELEC 1R AC
       (j) 11G18, FLT CONT ELEC 1R DC
       (k) 11G26, FLT CONT ELEC 2R AC
       (l) 11G27, FLT CONT ELEC 2R DC
       (m) 11H17, FLT CONT SHUTOFF TAIL L
       (n) 11H18, FLT CONT SHUTOFF TAIL CENTER
       (o) 11H27, FLT CONT SHUTOFF TAIL R

S 865-240

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(3) Pressurize the left and center hydraulic systems (AMM 29-11-00/201).

S 985-242

(4) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches on the P61 panel in the ON position.

S 985-243

(5) Put the left and right STAB TRIM switches on the control stand, P10, in the NORMAL position.

S 285-244

(6) Make sure these systems are operational:
   (a) Rudder and Rudder Trim Control System (AMM 27-21-00/501).
   (b) Engine Indication and Crew Alerting System (AMM 31-41-00/501).
   (c) Air Data Computing System (AMM 34-12-00/501).
   (d) Inertial Reference System (AMM 34-21-00/501).
   (e) Air/Ground Relays (AMM 32-09-02/201).
   (f) Master Dim and Test System (AMM 33-16-00/501).
   (g) 767-300 AIRPLANES;
       Flight Management Computer System (AMM 34-61-00/501)

S 985-245

(7) Push the STATUS switch on the EICAS display select panel on P9.

F. Power-Up-Test

S 985-246

(1) Push both YAW DAMPER-ON switch/lights and make sure that the ON legend comes on.

S 865-270

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(2) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 865-247
(3) Open for 1 second then close these P11 panel circuit breakers:
(a) 11C6, FLT CONT ELEC 1L AC
(b) 11C8, FLT CONT ELEC 2L AC
(c) 11G17, FLT CONT ELEC 1R AC
(d) 11G26, FLT CONT ELEC 2R AC

S 285-248
(4) Make sure that the YAW DAMPER INOP (L, R) lights on the overhead panel, P5, come on momentarily and then go off within 30 seconds.

S 985-249
(5) Push the RESET button on the left and right yaw damper modules.

S 285-250
(6) Stop for a minimum of 30 seconds, then make sure that the YAW DAMPER INOP L, R lights are off.

S 985-251
(7) Push the DISPLAY button on the left YAW DAMPER module. Make sure that the message NO FAULTS is shown on the left YAW DAMPER module's fault display.

S 985-252
(8) Push the DISPLAY button on the right YAW DAMPER module. Make sure that the message NO FAULTS is shown on the right YAW DAMPER module's fault display.

G. Power Supply Interface Test

S 865-269

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 865-253
(2) Open these P11 panel circuit breakers:
(a) 11G17, FLT CONT ELEC 1R AC
(b) 11G26, FLT CONT ELEC 2R AC

S 285-254
(3) Make sure that the YAW DAMPER INOP R light is on.

S 865-255
(4) Close this P11 panel circuit breaker:
(a) 11G26, FLT CONT ELEC 2R AC

S 285-256
(5) Stop for a minimum of 20 seconds then make sure that the YAW DAMPER INOP R light goes off.

S 865-268

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(6) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 865-257
(7) Open this P11 panel circuit breaker:
(a) 11G26, FLT CONT ELEC 2R AC

S 285-258
(8) Make sure that the YAW DAMPER INOP R light is on.

S 865-259
(9) Close this P11 panel circuit breaker:
(a) 11G17, FLT CONT ELEC 1R AC

S 285-260
(10) Stop for a minimum of 20 seconds then make sure that the YAW DAMPER INOP R light goes off.

S 865-267

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(11) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 865-261
(12) Close this P11 panel circuit breaker:
(a) 11G26, FLT CONT ELEC 2R AC

S 865-262
(13) Open these P11 panel circuit breakers:
(a) 11C6, FLT CONT ELEC 1L AC
(b) 11C8, FLT CONT ELEC 2L AC

S 285-263
(14) Make sure that the YAW DAMPER INOP L light is on.

S 865-264

(15) Close this P11 panel circuit breaker:
(a) 11C8, FLT CONT ELEC 2L AC

S 285-265

(16) Stop for a minimum of 20 seconds then make sure that the YAW DAMPER INOP L light goes off.

S 865-266

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(17) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 985-271

(18) Put the EICAS computer select switch, on the pilot's EICAS DISPLAY SELECT PANEL, to the L position.

S 865-272

(19) Open this P11 panel circuit breaker:
(a) 11C8, FLT CONT ELEC 2L AC

S 285-273

(20) Make sure that the YAW DAMPER INOP L light is on.

S 865-274

(21) Close this P11 panel circuit breaker:
(a) 11C6, FLT CONT ELEC 1L AC

S 285-275

(22) Stop for a minimum of 20 seconds then make sure that the YAW DAMPER INOP L light goes off.

S 865-276

(23) Close this P11 panel circuit breaker:
(a) 11C8, FLT CONT ELEC 2L AC

H. 28V DC Power Test

S 865-277

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 865-278
(2) Open these P11 panel circuit breakers:
(a) 11G18, FLT CONT ELEC 1R DC
(b) 11G27, FLT CONT ELEC 2R DC

S 285-279
(3) Make sure that the YAW DAMPER INOP R light is on.

S 865-280
(4) Close this P11 panel circuit breaker:
(a) 11G18, FLT CONT ELEC 1R DC

S 285-281
(5) Make sure that the YAW DAMPER INOP R light goes off.

S 865-282

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(6) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 865-416
(7) Open this P11 panel circuit breaker:
(a) 11G18, FLT CONT ELEC 1R DC

S 285-284
(8) Make sure that the YAW DAMPER INOP R light is on.

S 865-285
(9) Close this P11 panel circuit breaker:
(a) 11G27, FLT CONT ELEC 2R DC

S 285-286
(10) Make sure that the YAW DAMPER INOP R light goes off.

S 865-287

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(11) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 865-288
(12) Close this P11 panel circuit breaker:
   (a) 11G18, FLT CONT ELEC 1R DC

S 865-289
(13) Open these P11 panel circuit breakers:
   (a) 11C7, FLT CONT ELEC 1L DC
   (b) 11C9, FLT CONT ELEC 2L DC

S 285-417
(14) Make sure that the YAW DAMPER INOP L light is on.

S 865-291
(15) Close this P11 panel circuit breaker:
   (a) 11C9, FLT CONT ELEC 2L DC

S 285-292
(16) Make sure that the YAW DAMPER INOP L light goes off.

S 865-293

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(17) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 865-294
(18) Open this P11 panel circuit breaker:
(a) 11C9, FLT CONT ELEC 2L DC

S 285-295
(19) Make sure that the YAW DAMPER INOP L light is on.

S 865-296
(20) Close this P11 panel circuit breaker:
(a) 11C7, FLT CONT ELEC 1L DC

S 285-297
(21) Make sure that the YAW DAMPER INOP L light goes off.

S 865-298
(22) Close this P11 panel circuit breaker:
(a) 11C9, FLT CONT ELEC 2L DC

I. Yaw Damper Engage Test

S 865-299
(1) Open this P11 panel circuit breaker:
(a) 11A18, YAW DAMPER L

S 985-300
(2) Put the EICAS computer select switch on the pilot's EICAS DISPLAY SELECT PANEL in the L position.

S 285-301
(3) Make sure that the YAW DAMPER INOP L light is on and that the yellow L YAW DAMPER message is shown on the upper EICAS display.

S 985-302
(4) Put the EICAS computer select switch in the R position.

S 285-303
(5) Make sure that the upper EICAS display shows the L YAW DAMPER message in yellow.

S 865-304
(6) Close this P11 panel circuit breaker:
(a) 11A18, YAW DAMPER L

S 285-305
(7) Stop for a minimum of 30 seconds then make sure that the YAW DAMPER INOP L light goes off and the L YAW DAMPER message is not shown on the upper EICAS display.

S 985-306
(8) Put the EICAS computer select switch in the L position.

S 285-307

(9) Make sure that the L YAW DAMPER message is not shown on the upper EICAS display.

S 865-313

(10) Open this P11 panel circuit breaker:
   (a) 11F34, YAW DAMPER R

S 285-312

(11) Make sure that the YAW DAMPER INOP R light is on and that the yellow R YAW DAMPER message is shown on the upper EICAS display.

S 985-311

(12) Put the EICAS computer select switch in the R position.

S 285-310

(13) Make sure that the upper EICAS display shows the R YAW DAMPER message in yellow.

S 865-309

(14) Close this P11 panel circuit breaker:
   (a) 11F34, YAW DAMPER R

S 285-308

(15) Stop for a minimum of 30 seconds then make sure that the YAW DAMPER INOP R light goes off and the R YAW DAMPER message is not shown on the top EICAS display.

S 985-314

(16) Put the EICAS computer select switch in the L position.

S 285-315

(17) Make sure that the R YAW DAMPER message is not shown on the upper EICAS display.

J. Air/Ground System Interface Test

S 865-362

WARNING: PREPARE THE SAFETY-SENSITIVE SYSTEMS FOR THE AIR MODE BEFORE YOU OPEN THE AIR/GROUND CIRCUIT BREAKERS. IN THE AIR MODE, MANY OF THE AIRPLANE SYSTEMS CAN OPERATE AND CAUSE INJURIES TO PERSONS AND DAMAGE TO EQUIPMENT.

(1) Prepare the safety sensitive systems for air mode simulation (AMM 32-09-02/201).

S 435-317

(2) In less than 2 seconds put the two Landing Gear Sensor Actuators against the proximity sensors, S245 and S267 (Left Gear Sys No. 1 and No. 2) with the simulated gap facing the sensor.

S 985-319

(3) Put the YAW DMPR test switch on the P61 panel in the L position, then R, and return to center.

S 285-422

(4) Make sure that the YAW DAMPER INOP L and R lights on the overhead panel, P5, are off and the rudder position indicator on the lower EICAS display shows no rudder movement for 10 seconds.

S 985-321

(5) Push the DISPLAY button on the left YAW DAMPER module. Make sure that the message AIRGND 1/2 is shown on the left YAW DAMPER module's fault display.

S 985-322

(6) Push the DISPLAY button on the right YAW DAMPER module. Make sure that the message AIRGND 1/2 is shown on the right YAW DAMPER module's fault display.

S 435-324

(7) Put the Landing Gear Sensor Actuator against the proximity sensor S246 (R GEAR SYS NO. 1) with the simulated gap facing the sensor.

S 285-325

(8) Stop for a minimum of 15 seconds, then push the DISPLAY button on the left YAW DAMPER module. Make sure that these fault messages are not shown on the left yaw damper module's fault display:
  (a) AIRGND 1
  (b) AIRGND 1/2
  (c) AIRGND 2

S 985-326

(9) Push the DISPLAY button on the right YAW DAMPER module. Make sure that these fault messages are not shown on the right yaw damper module's fault display:
  (a) AIRGND 1
  (b) AIRGND 1/2
  (c) AIRGND 2

S 435-327

(10) Put the Landing Gear Sensor Actuator against the proximity sensor S268 (R GEAR SYS NO. 2) with the simulated gap facing the sensor. Remove the actuator from sensor S245 (L GEAR SYS NO. 1).

S 285-328

(11) Stop for a minimum of 15 seconds, then push the DISPLAY button on the left YAW DAMPER module. Make sure that these fault messages are not shown on the left yaw damper module's fault display:
  (a) AIRGND 1
  (b) AIRGND 1/2
  (c) AIRGND 2

S 285-329

(12) Push the DISPLAY button on the right YAW DAMPER module. Make sure that these fault messages are not shown on the right yaw damper module's fault display:
  (a) AIRGND 1

(b) AIRGND 1/2
(c) AIRGND 2

S 035-330
(13) Remove the actuator from sensor S267 (L GEAR SYS NO. 2).

S 285-331
(14) Push the DISPLAY button on the left YAW DAMPER module. Make sure that the message AIRGND 1/2 is shown on the left YAW DAMPER module's fault display.

S 285-332
(15) Push the DISPLAY button on the right YAW DAMPER module. Make sure that the message AIRGND 1/2 is shown on the right YAW DAMPER module's fault display.

S 035-333
(16) In less than 2 seconds, remove the actuators from sensors S246 and S268 (R GEAR SYS NO. 1 and NO. 2).

S 985-334
(17) Push and release the reset button on both YAW DAMPER modules.

S 285-335
(18) Stop for a minimum of 15 seconds, then push the DISPLAY button on the left YAW DAMPER module. Make sure that these fault messages are not shown on the left yaw damper module's fault display:
(a) AIRGND 1
(b) AIRGND 1/2
(c) AIRGND 2

S 285-336
(19) Push the DISPLAY button on the right YAW DAMPER module. Make sure that these fault messages are not shown on the right yaw damper module's fault display:
(a) AIRGND 1
(b) AIRGND 1/2

(c) AIRGND 2

S 865-325

(20) Put the safety sensitive systems back to their initial condition (AMM 32-09-02/201).

K. Hydraulic Interface Test

S 865-326

(1) Remove pressure from the center hydraulic system (AMM 29-11-00/201).

S 985-283

(2) Put the YAW DMPR test switch, on the P61 panel, in the L position, then put the switch back to the center.

S 285-327

(3) Make sure that the rudder position indicator on the lower EICAS display shows no rudder movement.

S 285-285

(4) Stop for a minimum of 15 seconds from step (2), and make sure that the YAW DAMPER INOP L light is on.

S 285-344

(5) Stop until there are no messages on the fault display of the left yaw damper module:

(a) Push the display button on the left yaw damper module.

(b) Make sure that the FAULTS NOW message is shown on the fault display of the left yaw damper module.

S 285-287

(6) Push the display button on the left yaw damper module:

(a) Make sure the YD TEST message is shown on the fault display of the left yaw damper module.

S 865-329

(7) Supply hydraulic pressure to the center hydraulic system.

S 985-289
(8) Push the left YAW DAMPER switch/light on the overhead panel, P5, two times to engage the left system again (ON-OFF-ON).

S 285-290
(9) Stop for a minimum of 30 seconds and make sure that the YAW DAMPER INOP L light goes off.

S 865-330
(10) Push the RESET button on the left and right yaw damper modules.

S 285-345
(11) Stop until there are no messages on the fault display of the left yaw damper module:
- (a) Push the display button on the left yaw damper module.
- (b) Make sure that the NO FAULTS or NO FAULT NOW message shows on the fault display of the left yaw damper module.

S 865-331
(12) Remove pressure from the left hydraulic system (Ref 29-11-00).

S 865-333
(13) Put the YAW DMPR test switch on the P61 panel in the R position, then put the switch back to the center.

S 985-294
(14) Make sure that the rudder position indicator on the lower EICAS display shows no rudder movement.

S 285-334
(15) Stop for a minimum of 15 seconds from step (13), and make sure that the YAW DAMPER INOP R light is on.

S 285-296
(16) Stop until there are no messages on the fault display of the right yaw damper module.
- (a) Push the display button on the right yaw damper module.
- (b) Make sure that the FAULTS NOW message is shown on the fault display of the right yaw damper module.

S 285-346
(17) Push the display button on the right yaw damper module:
- (a) Make sure that the YD TEST message is shown on the fault display of the left yaw damper module.

S 865-336
(18) Supply hydraulic prssure to the left hydraulic system.

S 985-299
(19) Push the right yaw damper switch/light two times to engage the right system again (ON-OFF-ON).

S 285-337
(20) Stop for a minimum of 30 seconds and make sure that the YAW DAMPER INOP R light goes off.

S 865-302
(21) Push the RESET switch on the left and right yaw damper modules.

S 285-300
(22) Stop for a minimum of 15 seconds and push the display button on the right yaw damper module:
(a) Make sure that the NO FAULTS or NO FAULT NOW message is shown on the fault display of the right yaw damper module.

S 285-338
(23) Push the display button on the left yaw damper module:
(a) Make sure that the NO FAULTS or NO FAULT NOW message is shown on the fault display of the left yaw damper module.

L. Air Data Computer Interface Test

S 865-355

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(1) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 985-356
(2) Put the EICAS computer select switch on the pilot's EICAS DISPLAY SELECT PANEL in the L position.

S 985-357
(3) Push and hold the test button on the L ADC.

NOTE: When you push the test button on the ADC, a self-test signal and a failure signal go to the YDM. The failure signal is used by the YDM as a fault message. This message will be shown on the fault display when you push the YDMs DISPLAY button.

S 985-358
(4) Push the DISPLAY button on the left YAW DAMPER module. Make sure that the message L ADC is shown on the left YAW DAMPER module's fault display.

S 985-359
(5) Push the DISPLAY button on the right YAW DAMPER module. Make sure that the message L ADC is shown on the right yaw damper module's fault display.

S 985-360
(6) Push the ECS/MSG button on the EICAS MAINTENANCE PNL.

S 285-421
(7) Make sure that the YAW DAMPER maintenance message is shown on the lower EICAS display.

S 985-363
(8) Put the EICAS computer select switch in the R position.

S 285-362
(9) Make sure that the YAW DAMPER maintenance message is shown on the lower EICAS display.

S 865-423
(10) Open these P11 panel circuit breakers:
(a) 11C6, FLT CONT ELEC 1L AC
(b) 11C8, FLT CONT ELEC 2L AC

S 285-365
(11) Make sure that the YAW DAMPER maintenance message is shown on the lower EICAS display.

S 865-366
(12) Close these P11 panel circuit breakers:
(a) 11C6, FLT CONT ELEC 1L AC
(b) 11C8, FLT CONT ELEC 2L AC

S 865-367
(13) Open these P11 panel circuit breakers:
(a) 11G17, FLT CONT ELEC 1R AC
(b) 11G26, FLT CONT ELEC 2R AC

S 285-368
(14) Make sure that the YAW DAMPER maintenance message is shown on the bottom EICAS display.

S 865-369
(15) Close these P11 panel circuit breakers:
(a) 11G17, FLT CONT ELEC 1R AC

(b) 11G26, FLT CONT ELEC 2R AC

S 985-370
(16) Release the test button on the L ADC.

S 985-371
(17) Push the DISPLAY button on the left YAW DAMPER module. Make sure that these fault messages are not shown on the left yaw damper module's fault display:
(a) L ADC
(b) R ADC

S 985-372
(18) Push the DISPLAY button on the right YAW DAMPER module. Make sure that these fault messages are not shown on the right yaw damper module's fault display:
(a) L ADC
(b) R ADC

S 285-373
(19) Make sure that the YAW DAMPER maintenance message is not shown on the bottom EICAS display.

S 985-374
(20) Put the EICAS computer select switch in the L position.

S 285-375
(21) Make sure that the YAW DAMPER maintenance message is not shown on the bottom EICAS display.

S 985-380
(22) Push and hold the test button on the R ADC.

S 985-379
(23) Push the DISPLAY button on the left YAW DAMPER module. Make sure that the message R ADC is shown on the left yaw damper module's fault display.

S 985-378
(24) Push the DISPLAY button on the right YAW DAMPER module. Make sure that the message R ADC is shown on the right yaw damper module's fault display.

S 985-377
(25) Release the test button on the R ADC.

S 985-376
(26) Push the DISPLAY button on the left YAW DAMPER module. Make sure that these fault messages are not shown on the left yaw damper module's fault display:
(a) L ADC
(b) R ADC

S 985-381
(27) Push the DISPLAY button on the right YAW DAMPER module. Make sure that these fault messages are not shown on the right yaw damper module's fault display:
(a) L ADC
(b) R ADC

M. Inertial Reference Unit (IRU) Interface Test

NOTE: When you push the TEST button on an IRU, the IRU initiates a 10 second self-test. The yaw damper stores this as a fault message that can be shown on the YDM when you push the YDM DISPLAY button.

S 985-382
(1) Push and release the TEST button on front of the left IRU.

S 285-383
(2) Stop for a minimum of 5 seconds, then push the DISPLAY button on the left YAW DAMPER module. Make sure that the message L IRU is shown on the left yaw damper module's fault display.

S 985-384
(3) Push the DISPLAY button on the right YAW DAMPER module. Make sure that the message L IRU is shown on the right YAW DAMPER module's fault display.

S 285-385
(4) Stop for a minimum of 10 seconds, then push the DISPLAY button on the left YAW DAMPER module. Make sure that the message L IRU is not shown on the left YAW DAMPER module's fault display.

S 985-386
(5) Push the DISPLAY button on the right YAW DAMPER module. Make sure that the message L IRU is not shown on the right YAW DAMPER module's fault display.

S 985-387
(6) Push and release the TEST button on front of the right IRU.

S 285-392

(7) Stop for a minimum of 5 seconds, then push the DISPLAY button on the left YAW DAMPER module. Make sure that the message R IRU is shown on the left YAW DAMPER module's fault display.

S 985-391

(8) Push the DISPLAY button on the right YAW DAMPER module. Make sure that the message R IRU is shown on the right YAW DAMPER module's fault display.

S 285-390

(9) Stop for a minimum of 10 seconds, then push the DISPLAY button on the left YAW DAMPER module. Make sure that the message R IRU is not shown on the left YAW DAMPER module's fault display.

S 985-389

(10) Push the DISPLAY button on the right YAW DAMPER module. Make sure that the message R IRU is not shown on the right YAW DAMPER module's fault display.

S 985-388

(11) Push and release the TEST button on front of center the IRU.

S 285-396

(12) Stop for a minimum of 5 seconds, then push the DISPLAY button on the left YAW DAMPER module. Make sure that the message C IRU is shown on the left YAW DAMPER module's fault display.

S 985-395

(13) Push the DISPLAY button on the right YAW DAMPER module. Make sure that the message C IRU is shown on the right YAW DAMPER module's fault display.

S 285-394

(14) Stop for a minimum of 10 seconds, then push the DISPLAY button on the left YAW DAMPER module. Make sure that the message C IRU is not shown on the left YAW DAMPER module's fault display.

S 985-393

(15) Push the DISPLAY button on the right YAW DAMPER module. Make sure that the message C IRU is not shown on the right YAW DAMPER module's fault display.

N. 767-300 AIRPLANES;
Flight Management Computer Interface Test

S 285-304
(1) Push the reset button on the right FMC:
(a) Make sure that the fault ball on the right FMC is not on.

S 285-347
(2) Push the reset button on the left FMC:
(a) Make sure that the fault ball on the left FMC is not on.

S 285-306
(3) Stop for a minimum of 30 seconds:
(a) Push the display button on the right YDM.
(b) Make sure that the message "NO FAULTS" or "NO FAULTS NOW" shows on the fault display of the right YDM.
(c) Push the display button on the left YDM.
(d) Make sure that the message "NO FAULT" or "NO FAULTS NOW" shows on the fault display of the left YDM.

S 865-307
(4) Open these circuit breakers:
(a) P11 Circuit Breaker Panel
1) 11E9 L-FMC COMPUTER
2) 11E30 R-FMC COMPUTER

S 285-348
(5) Push the display button on the right YDM:
(a) Make sure that the message "FAULTS NOW" shows on the fault display of the right YDM.

S 285-309
(6) Push the display button on the right YDM:
(a) Make sure that the message "FMC INPUT" shows on the fault display of the right YDM.

S 285-349
(7) Push the display button on the left YDM:
(a) Make sure that the message "FAULTS NOW" shows on the fault display of the left YDM.

S 285-311
(8) Push the display button on the left YDM:
(a) Make sure that the message "FMC INPUT" shows on the fault display of the left YDM.

S 865-312
(9) Close these circuit breakers:
(a) P11 Circuit Breaker Panel
1) 11E9 L-FMC COMPUTER

2) 11E30 R-FMC COMPUTER

S 285-350

(10) Push the reset button on the right FMC:
  (a) Make sure that the fault ball on the right FMC is not on.

S 285-314

(11) Push the reset button on the left FMC:
  (a) Make sure that the fault ball on the left FMC is not on.

S 285-351

(12) Stop for a minimum of 30 seconds:
  (a) Push the display button on the right YDM.
  (b) Make sure that the message "NO FAULTS" or "NO FAULTS NOW" shows on the fault display of the right YDM.
  (c) Push the display button on the left YDM.
  (d) Make sure that the message "NO FAULTS" or "NO FAULTS NOW" shows on the fault display of the left YDM.

O. 767-300 AIRPLANES;
Yaw Damper Modal Suppression Accelerometer Interface Test

S 865-397

(1) Open these P11 Panel circuit breakers:
  (a) 11C6, FLT CONT ELEC 1L AC
  (b) 11C8, FLT CONT ELEC 2L AC
  (c) 11G17, FLT CONT ELEC 1R AC
  (d) 11G26, FLT CONT ELEC 2R AC

S 035-398

(2) Disconnect the accelerometer electrical connectors D1937 and D1939.

S 865-399

(3) Close these P11 panel circuit breakers:
  (a) 11C6, FLT CONT ELEC 1L AC
  (b) 11C8, FLT CONT ELEC 2L AC
  (c) 11G17, FLT CONT ELEC 1R AC
  (d) 11G26, FLT CONT ELEC 2R AC

S 985-400

(4) Push the reset button on the left yaw damper module.

S 285-401
(5) Stop for a minimum of 15 seconds, then push the DISPLAY button on the left YAW DAMPER module. Make sure that the message ACCEL is shown on the left yaw damper module's fault display.

S 985-402
(6) Push the reset button on the right yaw damper module.

S 285-403
(7) Stop for a minimum of 15 seconds, then push the DISPLAY button on the right YAW DAMPER module. Make sure that the message ACCEL is shown on the right yaw damper module's fault display.

S 865-404
(8) Open these P11 Panel circuit breakers:
(a) 11C6, FLT CONT ELEC 1L AC
(b) 11C8, FLT CONT ELEC 2L AC
(c) 11G17, FLT CONT ELEC 1R AC
(d) 11G26, FLT CONT ELEC 2R AC

S 435-931
(9) Connect the accelerometer electrical connectors D1937 and D1939.

S 865-932
(10) Close these P11 panel circuit breakers:
(a) 11C6, FLT CONT ELEC 1L AC
(b) 11C8, FLT CONT ELEC 2L AC
(c) 11G17, FLT CONT ELEC 1R AC
(d) 11G26, FLT CONT ELEC 2R AC

S 985-933
(11) Push the reset button on the left yaw damper module.

S 285-934
(12) Stop for a minimum of 15 seconds, then push the DISPLAY button on the left YAW DAMPER module. Make sure that the message NO FAULTS is shown on the left yaw damper module's fault display.

S 985-935

(13) Push the reset button on the right yaw damper module.

S 285-936

(14) Stop for a minimum of 15 seconds, then push the DISPLAY button on the right YAW DAMPER module. Make sure that the message NO FAULTS is shown on the right yaw damper module's fault display.

P. Return the airplane to its usual condition.

S 865-937

(1) Remove the pressure from the left and center hydraulic systems (AMM 29-11-00/201).

S 865-938

(2) Remove electrical power if it is not necessary.

YAW DAMPER PANEL - REMOVAL/INSTALLATION

1. General
   A. The yaw damper panel is on the left side of overhead panel P5. The yaw damper panel is held in place by four 1/4 turn panel screws. Two electrical connectors are at the rear of the unit.

TASK 22-21-01-014-002

2. Remove the Yaw Damper Panel
   A. Reference
      (1) AMM 24-22-00/201, Electrical Power - Control
   B. Access
      (1) Location Zone
          211 Flight Compartment (Left)

   C. Prepare for Removal

      S 844-001
      (1) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11A18, YAW DAMPER L
          (b) 11A33, IND LIGHTS 1
          (c) 11A35, IND LIGHTS 3
          (d) 11F34, YAW DAMPER R
          (e) 11R29, RIGHT IND LTS 2
          (f) 11P3, INSTRUMENT & PANEL OVHD
   D. Procedure

      S 034-003
      (1) Hold the yaw damper panel in position and loosen the four quick-release screws.

      S 024-004
      (2) Carefully lower the yaw damper panel from the overhead panel.

      S 034-005
      (3) Remove the two electrical connectors attached to the rear of the panel.

TASK 22-21-01-414-006

3. Install the Yaw Damper Panel
   A. Reference
      (1) 24-22-00/201, Electrical Power - Control

B. Access

(1) Location Zone

211 Flight Compartment (Left)

C. Install the Yaw damper Panel

S 284-007

(1) Make sure that the yaw damper panel is correctly aligned, and connect the two electrical connectors to the rear of the unit.

S 424-008

(2) Carefully lift the panel into position and install it with four quick-release screws.

D. Yaw Damper Panel Test

S 864-010

(1) Supply electrical power (Ref 24-22-00).

S 864-011

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:

(a) 11A18, YAW DAMPER

(b) 11A33, IND LTS 1

(c) 11A35, IND LTS 3

(d) 11F34, YAW DAMPER R

(e) 11R29, RIGHT IND LTS 2

(f) 11P3, INSTRUMENT & PANEL OVHD

S 864-012

(3) Push the IND LIGHT TEST button on the overhead panel P5.

S 714-013

(4) Make sure that the INOP lights on the yaw damper panel come on.

S 864-014

(5) Push the two yaw damper switch/lights.

S 714-016

(6) Make sure the ON light comes on.

E. Put the airplane back to its initial condition.

S 864-015

(1) Remove electrical power it is not necessary (Ref 24-22-00).

YAW DAMPER SERVO - REMOVAL/INSTALLATION

1. General (Fig. 401)
   A. The two yaw damper servos are on the rear spar of the vertical stabilizer. You can access the servos through the trailing edge service access panel 324EL, on the vertical stabilizer.

TASK 22-21-02-014-001

2. Remove the Yaw Damper Servo

   A. References
      (1) AMM 06-42-00/201, Vertical Stabilizer and Rudder Access Doors and Panels
      (2) AMM 24-22-00/201, Electrical Power - Control
      (3) AMM 27-61-00/201, Spoiler/Speedbrake Control System
      (4) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
   B. Access
      (1) Location Zone
          320 Vertical Stabilizer and Rudder

      (2) Access Panel
          324EL Vertical Stabilizer, Rear Spar to Trailing Edge

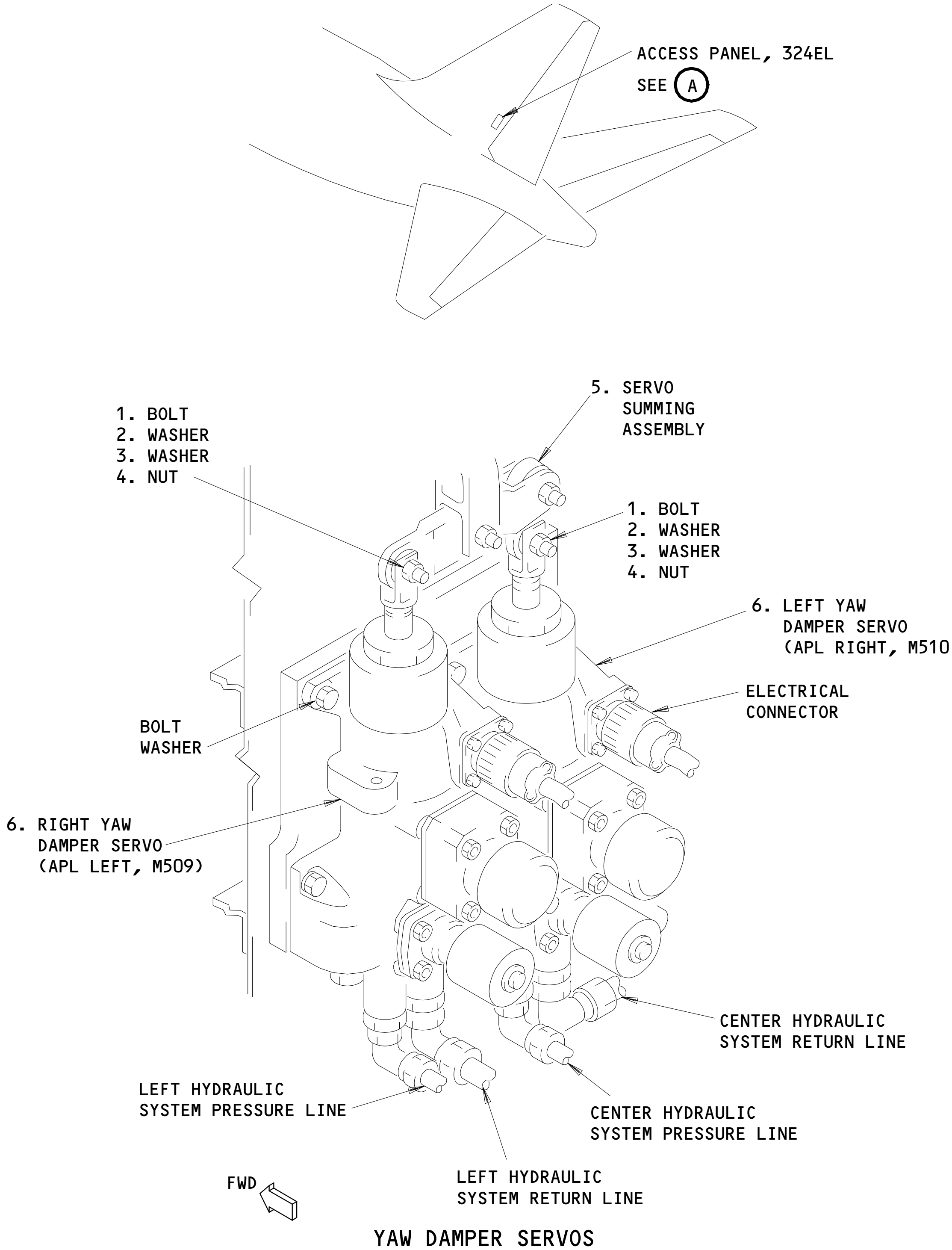
   C. Prepare For Removal

      S 864-002

      WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

      (1) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

      S 864-003
      (2) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11A18, YAW DAMPER L



Yaw Damper Servo Installation
Figure 401

(b) 11C6, FLT CONT ELEC 1L AC
(c) 11C7, FLT CONT ELEC 1L DC
(d) 11C8, FLT CONT ELEC 2L AC
(e) 11C9, FLT CONT ELEC 2L DC
(f) 11F34, YAW DAMPER R
(g) 11G17, FLT CONT ELEC 1R AC
(h) 11G18, FLT CONT ELEC 1R DC
(i) 11G26, FLT CONT ELEC 2R AC
(j) 11G27, FLT CONT ELEC 2R DC

S 864-004

(3) Supply electrical power (AMM 24-22-00/201).

S 864-005

(4) For the right yaw damper system servo (left side of airplane), remove the pressure in the left hydraulic system (AMM 29-11-00/201).

S 864-006

(5) For the left yaw damper system servo (right side of airplane), remove the pressure in the center hydraulic system (AMM 29-11-00/201).

S 864-007

(6) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches on the right side panel P61 in the OFF position.

S 864-008

(7) Put the two STAB TRIM switches on the control stand panel, P10, in the CUT OUT position.

S 864-009

(8) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11H17, FLT CONT SHUTOFF TAIL L

(d) 11H18, FLT CONT SHUTOFF TAIL CENTER
(e) 11H27, FLT CONT SHUTOFF TAIL R

S 864-059

WARNING: MAKE SURE PERSONNEL STAY A MINIMUM OF 6 FEET AWAY FROM THE VERTICAL STABILIZER WHEN THE HF SYSTEM TRANSMITS. RF ENERGY FROM THE HF COMMUNICATION ANTENNA CAN CAUSE INJURIES TO PERSONNEL.

(9) Make sure the HF system does not transmit.

S 014-010
(10) Open the service access panel 324EL (AMM 6-42-00/201).

D. Remove the Yaw Damper Servo

S 034-056
(1) Remove the electrical connector from the yaw damper servo.

S 034-011
(2) Remove the hydraulic lines from the yaw damper servo, and plug the lines.

S 414-012
(3) Plug the hydraulic ports on the yaw damper servo.

S 034-013
(4) Remove the bolt, washers and nut and disconnect the servo output from the servo summing assembly.

S 034-014
(5) Hold the yaw damper servo in position and remove the bolts and the washers.

S 024-015
(6) Remove the servo from the vertical stabilizer rear spar.

TASK 22-21-02-414-016

3. Install the Yaw Damper Servo

A. Equipment
(1) Bonding Meter, Avtron T477W - Avtron Manufacturing Co., 10409 Meech Ave., Cleveland, OH 44105

B. Parts

AMM		NOMENCLATURE	AIPC		
FIG	ITEM		SUBJECT	FIG	ITEM
401	1	Bolt	22-21-02	01	35
	2	Washer (Under Head)			45
	3	Washer (Under Nut)			50
	4	Nut			70
	5	Servo Summing Assy			227
	6	Yaw Damper Servo			310

C. References
   (1) AMM 06-42-00/201, Vertical Stabilizer and Rudder Access Doors and Panels
   (2) AMM 20-10-21/601, Electrical Bonding
   (3) AMM 24-22-00/201, Electrical Power - Control
   (4) AMM 27-21-00/501, Rudder and Rudder Trim Control System
   (5) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
   (6) AMM 31-41-00/501, Engine Indicating and Crew Alerting System
   (7) AMM 32-09-02/201, Air/Ground Relays
   (8) AMM 33-16-00/501, Master Dim and Test
   (9) AMM 34-12-00/501, Air Data Computing System
   (10) AMM 34-21-00/501, Inertial Reference System
   (11) AMM 34-61-00/501, Flight Management Computer System

D. Access
   (1) Location Zone
      320 Vertical Stabilizer and Rudder

   (2) Access Panel
      324EL Vertical Stabilizer, Rear Spar to Trailing Edge

E. Procedure

S 864-060

WARNING: MAKE SURE PERSONNEL STAY A MINIMUM OF 6 FEET AWAY FROM THE VERTICAL STABILIZER WHEN THE HF SYSTEM TRANSMITS. RF ENERGY FROM THE HF COMMUNICATION ANTENNA CAN CAUSE INJURIES TO PERSONNEL.

(1) Make sure the HF system does not transmit.

S 434-017
(2) Remove the plugs from the hydraulic ports on the servo and hydraulic lines.

S 434-018
(3) Connect the hydraulic and tighten by hand.

S 434-019
(4) Put the servo in position on the vertical stabilizer rear spar.

S 434-020
(5) Install the servo with washers and bolts.

S 434-021
(6) Connect the servo output to the servo summing assembly with the bolt, washers, and nut.

S 434-022
(7) Tighten the hydraulic lines.

S 434-023
(8) Connect the electrical connector to the servo.

S 764-058
(9) Use the bonding meter to make sure that the maximum resistance between the solenoid valve and ground is 6 milliohms (0.006 ohms) (AMM 20-10-21/601).

F. Yaw Damper Servo Test

S 864-026
(1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11A18, YAW DAMPER L
(b) 11C6, FLT CONT ELEC 1L AC
(c) 11C7, FLT CONT ELEC 1L DC
(d) 11C8, FLT CONT ELEC 2L AC
(e) 11C9, FLT CONT ELEC 2L DC
(f) 11F34, YAW DAMPER R
(g) 11G17, FLT CONT ELEC 1R AC
(h) 11G18, FLT CONT ELEC 1R DC
(i) 11G26, FLT CONT ELEC 2R AC

(j) 11G27, FLT CONT ELEC 2R DC

S 864-027

WARNING: ALL PERSONS AND STANDS MUST BE CLEAR OF CONTROL SURFACES AND CONTROL COLUMN WHEN HYDRAULIC SYSTEMS ARE PRESSURIZED. ALL CONTROL SURFACES ARE HYDRAULICALLY POWERED AND MAY MOVE WHEN ANY HYDRAULIC SYSTEMS ARE PRESSURIZED, OR IF ANY CONTROLS ARE MOVED. THIS CAN CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
   (a) 11C12, STAB TRIM SHUTOFF L
   (b) 11C13, STAB TRIM SHUTOFF CENTER
   (c) 11H17, FLT CONT SHUTOFF TAIL L
   (d) 11H18, FLT CONT SHUTOFF TAIL CENTER
   (e) 11H27, FLT CONT SHUTOFF TAIL R

S 864-028

(3) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches in the ON position.

S 864-029

(4) Put the two STAB TRIM switches on the control stand panel, P10, in the NORM position.

S 714-030

(5) Make sure that these systems operate:
   (a) Rudder and Rudder Trim Control System (AMM 27-21-00/501)
   (b) Air Data Computing System (AMM 34-12-00/501)
   (c) Inertial Reference System (AMM 34-21-00/501)
   (d) Air/Ground Relays (AMM 32-09-02/201)
   (e) Master Dim and Test System (AMM 33-16-00/501)
   (f) Engine Indication and Crew Alerting System (AMM 31-41-00/501)

(g) 767-300 AIRPLANES;
Flight Management Computer System (AMM 34-61-00/501)

S 864-031
(6) Select STATUS on the EICAS display select panel on P9.

S 864-032
(7) For the left yaw damper system servo, pressurize the center hydraulic system (AMM 29-11-00/201).

S 864-033
(8) For the right yaw damper system servo, pressurize the left hydraulic system (AMM 29-11-00/201).

S 864-034
(9) Push the left or the right YAW DAMPER switchlight on P5.

S 284-035
(10) Make sure that the ON light comes on.

S 284-036
(11) Make sure that the appropriate YAW DAMPER L or R INOP light is OFF.

S 864-037
(12) For the left yaw damper system servo (right side of airplane), put the YAW DMPR test switch on the right side panel P61 to L, then return the switch to center.

S 284-038
(13) Make sure that the YAW DAMPER L INOP light is on.

S 284-039
(14) Make sure that the rudder position indicator, on the bottom EICAS display, shows this sequence of rudder movements in less than 10 seconds.
(a) The rudder moves approximately 3 degrees trailing edge right.
(b) The rudder moves approximately 3 degrees trailing edge left.
(c) The rudder goes back to the center.

S 284-040
(15) Make sure that the YAW DAMPER L INOP light goes off within 15 seconds after the rudder goes back to the center.

S 864-041
(16) For the right yaw damper system servo (left side of airplane), put the YAW DMPR test switch on the right side panel, P61, to R, then return the switch to center.

S 284-042
(17) Make sure that YAW DAMPER R INOP light is on.

S 284-043

(18) Make sure that the rudder position indicator, on the bottom EICAS display, shows this sequence of rudder movements in less than 10 seconds.

(a) The rudder moves approximately 3 degrees trailing edge right.

(b) The rudder moves approximately 3 degrees trailing edge left.

(c) The rudder goes back to the center.

S 284-044

(19) Make sure that the YAW DAMPER R INOP light goes off within 15 seconds after the rudder goes back to center.

S 864-065

(20) Push the display button on the right yaw damper module:

(a) Make sure that the message NO FAULTS shows on the fault display of the right Yaw Damper Module.

S 864-068

(21) Push the display button on the left yaw damper module:

(a) Make sure that the message NO FAULTS shows on the fault display of the left Yaw Damper Module.

G. Look for hydraulic leakage.

S 864-048

(1) Put the L, C and R TAIL FLT CONTROL SHUTOFF switches in the OFF position.

S 864-049

(2) Put the two STAB TRIM switches in the CUT OUT position.

S 864-050

(3) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:

(a) 11C12, STAB TRIM SHUTOFF L

(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11H17, FLT CONT SHUTOFF TAIL L
(d) 11H18, FLT CONT SHUTOFF TAIL CENTER
(e) 11H27, FLT CONT SHUTOFF TAIL R

S 864-061

WARNING: MAKE SURE PERSONNEL STAY A MINIMUM OF 6 FEET AWAY FROM THE VERTICAL STABILIZER WHEN THE HF SYSTEM TRANSMITS. RF ENERGY FROM THE HF COMMUNICATION ANTENNA CAN CAUSE INJURIES TO PERSONNEL.

(4) Make sure the HF system does not transmit.

S 794-051
(5) Make an inspection of the servos for hydraulic leakage and correct as necessary.

S 414-052
(6) Close service access panel 324EL (AMM 06-42-00/201).

H. Put the Airplane Back to its Initial Condition

S 864-053
(1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11H17, FLT CONT SHUTOFF TAIL L
(d) 11H18, FLT CONT SHUTOFF TAIL CENTER
(e) 11H27, FLT CONT SHUTOFF TAIL R

S 864-062
(2) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches in the ON position.

S 864-063
(3) Put the two L and C STAB TRIM switches in the NORM position.

S 864-054
(4) Remove the power from the left and center hydraulic systems (AMM 29-11-00/201).

S 864-055
(5) Remove electrical power if it is not necessary (AMM 24-22-00/201).

YDS ELECTROHYDRAULIC SERVOVALVE AND SOLENOID VALVES - MAINTENANCE PRACTICES

1. General (Fig. 201)
   A. Two Yaw Damper Servos are in the vertical stabilizer. Each servo unit has one electrohydraulic servovalve (EHSV) and one solenoid valve. You can access the servos through the trailing edge service access panel, 324EL, on the vertical stabilizer (Ref 06-42-00).

TASK 22-21-03-022-082

2. Remove Valve
   A. General
      (1) You can use this procedure to remove either the electrohydraulic servovalve or the solenoid valve.
   B. Equipment
      (1) Bonding Meter (AMM 20-10-21/401)
   C. Consumable Materials
      (1) D00633 Grease - BMS 3-33 (Preferred)
      (2) D00013 Grease - MIL-PRF-23827 (Supersedes MIL-G-23827) (Alternate)
      (3) D00153 Fluid - Hydraulic, Fire Resistant BMS 3-11
   D. References
      (1) AMM 06-42-00/201, Vertical Stabilizer and Rudder Access Doors and Panels.
      (2) AMM 24-22-00/201, Electrical Power - Control
      (3) AMM 27-61-00/201, Spoiler/Speedbrake Control System
      (4) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
   E. Access
      (1) Location Zone

         324 Vertical Stabilizer - Rear Spar to Trailing Edge

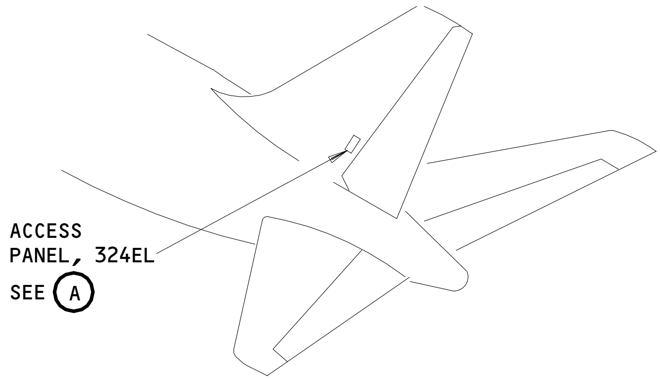
      (2) Access Panel

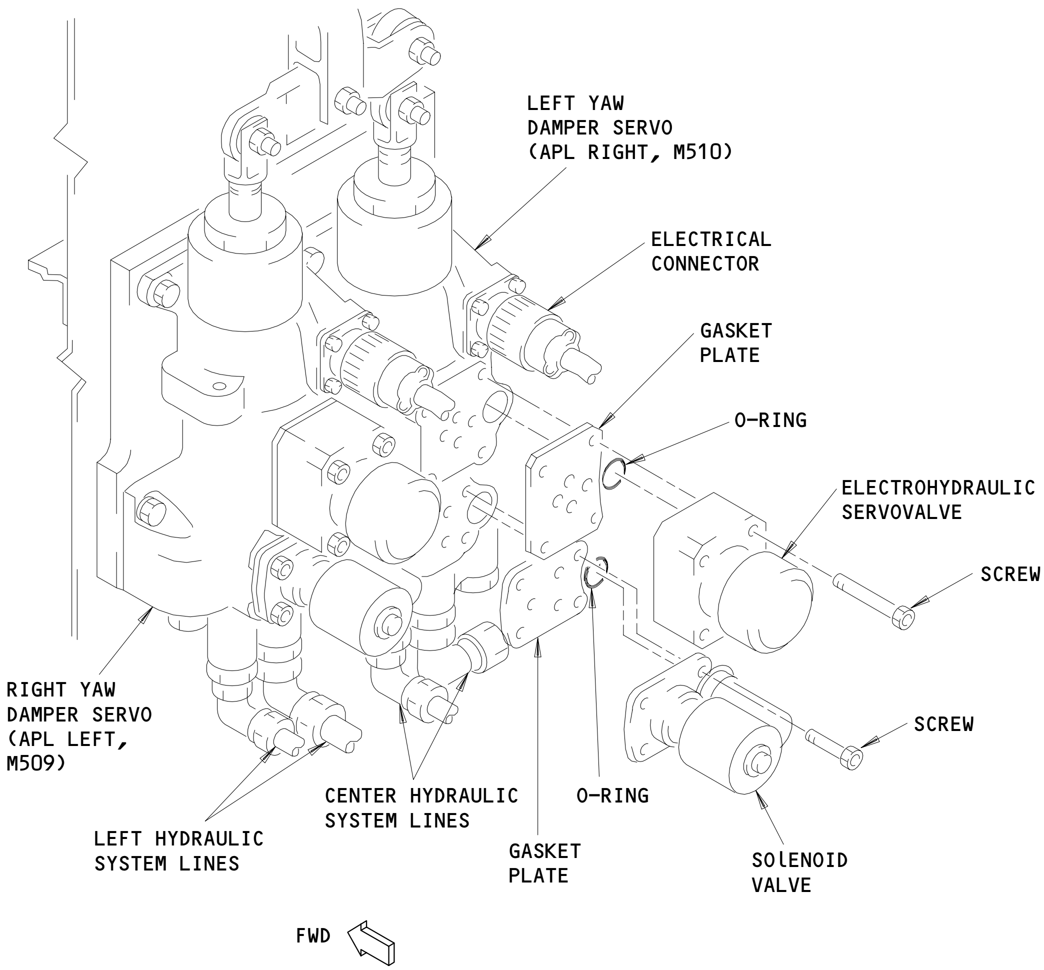
         324EL Stabilizer Service Access Panel

   F. Prepare For Removal

      S 862-001

      (1) Supply electrical power (AMM 24-22-00/201)





YDS ELECTROHYDRAULIC SERVOVALVE AND SOLENOID VALVES

A

YDS Electrohydraulic Servovalve and Solenoid Valves
Figure 201

S 842-083

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(2) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 862-002

(3) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags. Make sure you open the FLT CONT ELEC AC circuit breakers before you open the FLT CONT ELEC DC circuit breakers for each power supply.
   (a) 11A18, YAW DAMPER L
   (b) 11C6, FLT CONT ELEC 1L AC
   (c) 11C7, FLT CONT ELEC 1L DC
   (d) 11C8, FLT CONT ELEC 2L AC
   (e) 11C9, FLT CONT ELEC 2L DC
   (f) 11F34, YAW DAMPER R
   (g) 11G17, FLT CONT ELEC 1R AC
   (h) 11G18, FLT CONT ELEC 1R DC
   (i) 11G26, FLT CONT ELEC 2R AC
   (j) 11G27, FLT CONT ELEC 2R DC

S 862-003

(4) For the left yaw damper system servo valves (right side of airplane), remove pressure from the center hydraulic system. For the right yaw damper system servo valves (left side of airplane), remove pressure from the left hydraulic system (AMM 29-11-00/201).

S 862-004

(5) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches on the right side panel P61 in the OFF position.

S 982-005

(6) Put the two STAB TRIM switches on the control stand panel, P10, in the CUT OUT position.

S 982-006

(7) Open these P11 panel circuit breakers and attach DO-NOT-CLOSE tags:
   (a) 11C12, STAB TRIM SHUTOFF L
   (b) 11C13, STAB TRIM SHUTOFF CENTER

(c) 11H17, FLT CONT SHUTOFF TAIL L
(d) 11H18, FLT CONT SHUTOFF TAIL CENTER
(e) 11H27, FLT CONT SHUTOFF TAIL R

S 862-103

WARNING: MAKE SURE PERSONNEL STAY A MINIMUM OF 6 FEET AWAY FROM THE VERTICAL STABILIZER WHEN THE HF SYSTEM TRANSMITS. RF ENERGY FROM THE HF COMMUNICATION ANTENNA CAN CAUSE INJURIES TO PERSONNEL.

(8) Make sure the HF system does not transmit.

S 012-007
(9) Open the service access panel, 324EL (AMM 06-42-00/201).

G. Remove Valve

S 162-008
(1) Clean the external areas of the valve and around the valve mounting surface.

S 032-009
(2) Remove the lockwire from the mounting screws.

S 032-010
(3) Remove the mounting screws from the valve assembly.

S 022-011
(4) Remove the valve, the O-ring and the gasket plate from the servo assembly.

S 032-012
(5) Cover the servo openings to prevent contamination from entering hydraulic or electrical passages.

TASK 22-21-03-402-095

3. Install Valve

A. General
(1) You can use this procedure to install either the electrohydraulic servovalve or the solenoid valve.

B. Equipment
(1) Bonding Meter, Avtron T477W - Avtron Manufacturing Co., 10409 Meech Ave., Cleveland, OH 44105

C. Consumable Materials
(1) D00013 Grease, MIL-G-23827

D. References
(1) AMM 20-10-21/601, Electrical Bonding
(2) AMM 20-10-23/401, Lockwires - Removal/Installation

E. Access
(1) Location Zone
324 Vertical Stabilizer - Rear Spar to Trailing Edge

(2) Access Panel
324EL Stabilizer Service Access Panel

F. Install Valve

S 862-104

WARNING: MAKE SURE PERSONNEL STAY A MINIMUM OF 6 FEET AWAY FROM THE VERTICAL STABILIZER WHEN THE HF SYSTEM TRANSMITS. RF ENERGY FROM THE HF COMMUNICATION ANTENNA CAN CAUSE INJURIES TO PERSONNEL.

(1) Make sure the HF system does not transmit.

S 032-013
(2) Remove the covers from the openings.

S 032-014
(3) Clean the mating surface of the valve, the new gasket plate and the servo assembly with clean hydraulic fluid. Do not allow hydraulic fluid to go into the electrical connector.

S 412-015
(4) Install the new O-ring on the valve electrical connector.

S 422-016
(5) Install the valve and the new gasket plate.

S 432-017
(6) Coat the mounting screws with grease and install the screws.

S 432-109
(7) Tighten the mounting screws to 50-80 pound-inches.

S 432-018
(8) Install lockwire on the mounting screws (AMM 20-10-23/401).

S 482-094

(9) Use the bonding meter to make sure that the maximum resistance between the replaced valve and ground is 6 milliohms (0.006 ohms) (AMM 20-10-21/601) .

TASK 22-21-03-712-020

4. Test Yaw Damper Electrohydraulic Servovalve And Solenoid Valve

A. References

(1) AMM 24-22-00/201, Electrical Power - Control
(2) AMM 27-21-00/501, Rudder and Rudder Trim Control System
(3) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
(4) AMM 31-41-00/501, Engine Indication and Crew Alerting System.
(5) AMM 32-09-02/201, Air/Ground Relays
(6) AMM 33-16-00/501, Master Dim and Test
(7) AMM 34-12-00/501, Air Data Computing System
(8) AMM 34-21-00/501, Inertial Reference System
(9) AMM 34-61-00/501, Flight Management Computer System

B. Prepare for test.

S 862-021

(1) Supply electrical power (AMM 24-22-00/201).

S 862-022

(2) Remove DO-NOT-CLOSE tags and close these P11 panel circuit breakers. Make sure you close the FLT CONT ELEC DC circuit breakers before you close the FLT CONT ELEC AC circuit breakers for each power supply.
(a) 11A18, YAW DAMPER L
(b) 11C6, FLT CONT ELEC 1L AC
(c) 11C7, FLT CONT ELEC 1L DC
(d) 11C8, FLT CONT ELEC 2L AC
(e) 11C9, FLT CONT ELEC 2L DC
(f) 11F34, YAW DAMPER R
(g) 11G17, FLT CONT ELEC 1R AC
(h) 11G18, FLT CONT ELEC 1R DC
(i) 11G26, FLT CONT ELEC 2R AC
(j) 11G27, FLT CONT ELEC 2R DC

S 862-023

(3) Close the six EICAS circuit breakers on the P11 panel.

S 982-025
(4) Select STATUS on EICAS display select panel on P9.

S 862-026
(5) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
   (a) 11C12, STAB TRIM SHUTOFF L
   (b) 11C13, STAB TRIM SHUTOFF C
   (c) 11H17, FLT CONT SHUTOFF TAIL L
   (d) 11H18, FLT CONT SHUTOFF TAIL CENTER
   (e) 11H27, FLT CONT SHUTOFF TAIL R

S 982-027
(6) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches on right side panel, P61, in the ON position.

S 982-028
(7) Put the two STAB TRIM switches on control panel, P10, in the NORMAL position.

S 862-096

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(8) For the left yaw damper system, pressurize the center hydraulic system (AMM 29-11-00/201).

S 862-112

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(9) For the right yaw damper system, pressurize the left hydraulic system (AMM 29-11-00/201).

C. Test the Valves

S 282-031
(1) Make sure that these systems are operational:
   (a) Rudder and Rudder Trim Control System (AMM 27-21-00/501)
   (b) Air Data Computing System (AMM 34-12-00/501)

(c) Inertial Reference System (AMM 34-21-00/501)
(d) Air/Ground Relays (AMM 32-09-02/201)
(e) Master Dim and Test System (AMM 33-16-00/501)
(f) Engine Indication and Crew Alerting System (AMM 31-41-00/501).
(g) 767-300 AIRPLANES;
Flight Management Computer System (AMM 34-61-00/501)

S 982-032
(2) Push the left or right YAW DAMPER switch/light on the P5 panel and make sure that ON is lit.

S 282-033
(3) Make sure that the appropriate YAW DAMPER L or R INOP light is off.

NOTE: Two persons are necessary to do the steps that follow: one in the flight compartment and one in the main equipment center.

S 862-034
(4) To test the left yaw damper system servovalve and solenoid valve (right side of airplane), open this circuit breaker on the P11 panel:
(a) 11A18, YAW DAMPER L

S 282-035
(5) Make sure that the YAW DAMPER L INOP light is on.

S 982-036
(6) Put the YAW DMPR test switch on the P61 panel to the L position, then return the switch to the center.

S 282-037
(7) Make sure that the rudder position indicator on the EICAS display unit shows no rudder movement.

NOTE: AIRPLANES WITH YDM;
Faults indicated in this test should be clear after you do the reset procedure that follows.

S 862-038
(8) Close this P11 circuit breaker.
(a) 11A18, YAW DAMPER L

S 282-187
(9) AIRPLANES WITH YDM;
Push the RESET button on the front of the left yaw damper module.

S 982-188
(10) AIRPLANES WITH YDM;
Push the DISPLAY button on the left yaw damper module. There is a fault unless the NO FAULTS message is shown.

S 982-044
(11) Put the YAW DMPR test switch on the P61 panel in the L position, then return it to the center. Make sure that the YAW DAMPER INOP light is on.

S 282-085
(12) Make sure that the rudder position indicator, on the bottom EICAS display, shows this sequence of rudder movements in less than 10 seconds:
(a) The rudder moves approximately 3 degrees trailing edge right.
(b) The rudder moves approximately 3 degrees trailing edge left.
(c) The rudder goes back to the center.

S 282-045
(13) Make sure that the YAW DAMPER L INOP light goes out in less than 15 seconds.

S 282-052
(14) The NO FAULTS message should be the only message shown on the yaw damper module after completion of this test if there are no faults.

S 862-053
(15) To test the right yaw damper system servovalve and solenoid valve (left side of airplane), open this circuit breaker on the P11 panel:
(a) 11F34, YAW DAMPER R

S 282-054
(16) Make sure that the YAW DAMPER R INOP light is on.

S 982-055
(17) Put the YAW DMPR test switch on the P61 panel in the R position, then return it to the center position.

S 282-056
(18) Make sure that the rudder position indicator on the EICAS display unit shows no rudder movement.

NOTE: During this test a YD ACT fault may be shown and a YDM fault will be shown on the right Yaw Damper module.
Faults indicated in this test should be clear after you push the YDM RESET button in the next step.

S 862-057
(19) Close this P11 panel circuit breaker:
(a) 11F34, YAW DAMPER R

S 282-058
(20) Push the RESET button on the front of the right yaw damper module.

S 982-062
(21) Push the DISPLAY button on the right yaw damper module. There is a fault unless the NO FAULTS message is shown.

S 982-063
(22) Put the YAW DAMPER test switch on the P61 panel in the R position, then return it to the center.

S 282-064
(23) Make sure that the YAW DAMPER R INOP light is on.

S 282-108
(24) Make sure that the rudder position indicator, on the bottom EICAS display (AMM 31-41-00/001), shows this sequence of rudder movements in less than 10 seconds:
(a) The rudder moves approximately 3 degrees trailing edge right.

(b) The rudder moves approximately 3 degrees trailing edge left.
(c) The rudder goes back to the center.

S 282-065
(25) Make sure that the YAW DAMPER R INOP light goes off in less than 15 seconds after the action of the step before.

S 282-072
(26) The NO FAULTS message should be the only message shown on the yaw damper module after completion of this test if there are no faults.

D. Put the Airplane Back to Its Usual Condition

S 862-090
(1) Remove hydraulic power if it is not necessary (AMM 29-11-00/201).

S 862-087
(2) Remove electrical power if it is not necessary (AMM 24-22-00/201).

TASK 22-21-03-202-088

5. <u>Examine the Yaw Damper Electrohydraulic Servovalve and the Solenoid Valve Leaks</u>

A. References
(1) AMM 06-42-00/201, Vertical Stabilizer and Rudder Access Doors and Panels.
(2) 24-22-00/201, Electrical Power - Control
(3) 29-11-00/201, Pressurize/Depressurize Main Hydraulic System

B. Access
(1) Location Zones

211/212	Flight Compartment
324	Vertical Stabilizer

(2) Access Panel

32EL	Rudder LE Spar

C. Examine for Leakage.

S 862-089
(1) Supply electrical power (AMM 24-22-00/201).

S 862-110

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(2) For the left yaw damper system, pressurize the center hydraulic system (AMM 29-11-00/201).

S 862-111

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(3) For the right yaw damper system, pressurize the left hydraulic system (AMM 29-11-00/201).

S 862-073

(4) Open this circuit breaker on the P11 panel and attach DO-NOT-CLOSE tags before you do the next step. This will prevent rudder ratio failure annunciation:
   (a) 11G10, RUDDER RATIO

S 982-074

(5) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches in the OFF position.

S 982-086

(6) Put the two STAB TRIM switches in the CUT OUT position.

S 862-075

(7) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
   (a) 11C12, STAB TRIM SHUTOFF L
   (b) 11C13, STAB TRIM SHUTOFF CENTER
   (c) 11H17, FLT CONT SHUTOFF TAIL L
   (d) 11H18, FLT CONT SHUTOFF TAIL CENTER
   (e) 11H27, FLT CONT SHUTOFF TAIL R

S 862-105

WARNING: MAKE SURE PERSONNEL STAY A MINIMUM OF 6 FEET AWAY FROM THE VERTICAL STABILIZER WHEN THE HF SYSTEM TRANSMITS. RF ENERGY FROM THE HF COMMUNICATION ANTENNA CAN CAUSE INJURIES TO PERSONNEL.

(8) Make sure the HF system does not transmit.

S 792-106
(9) Inspect valves for hydraulic leakage and correct as necessary.

S 862-107
(10) Close the service access panel 324EL (AMM 06-42-00/201).

D. Put the Airplane Back To Its Usual Condition

S 862-076
(1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C12, STAB TRIM SHUTOFF L
(b) 11C13, STAB TRIM SHUTOFF CENTER
(c) 11H17, FLT CONT SHUTOFF TAIL L
(d) 11H18, FLT CONT SHUTOFF TAIL CENTER
(e) 11H27, FLT CONT SHUTOFF TAIL R

S 982-077
(2) Put the L, C, and R TAIL FLT CONTROL SHUTOFF switches in the ON position.

S 862-078
(3) Remove the DO-NOT-CLOSE tag and close this circuit breaker on the P11 panel:
(a) 11G10, RUDDER RATIO

S 982-079
(4) Put the L and the C STAB TRIM switches in the NORMAL position.

S 862-080
(5) Remove hydraulic power if it is not necessary (AMM 29-11-00/201).

S 862-081
(6) Remove electrical power if it is not necessary (AMM 24-22-00/201).

YAW DAMPER MODULE - REMOVAL/INSTALLATION

1. General
   A. This section contains the removal and installation procedures for the Yaw Damper Modules (YDM).

TASK 22-21-04-014-001

2. Remove the Yaw Damper Module
   A. General
      (1) The Yaw Damper Modules are in the E/E rack and are electrostatic sensitive. Special procedures are used to make sure they are not damaged.
   B. References
      (1) AMM 20-10-01/401, E/E Rack Mounted Components
      (2) AMM 20-41-01/201, Electrostatic Sensitive Devices
      (3) AMM 27-61-00/201, Spoiler/Speedbrake Control System
   C. Access
      (1) Location Zone
          119/120 Main Equipment Center (Left and Right)

      (2) Access Panel
          119AL Main Equipment Center Access

   D. Prepare To Remove Yaw Damper Module

      S 044-004

      WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

      (1) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

      S 864-010
      (2) Make sure you open the AC circuit breakers before you open the DC circuit breakers for each power supply (FLT CONT ELEC).

S 864-006

(3) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
   (a) 11A18, YAW DAMPER L
   (b) 11C6, FLT CONT ELEC 1L AC
   (c) 11C7, FLT CONT ELEC 1L DC
   (d) 11C8, FLT CONT ELEC 2L AC
   (e) 11C9, FLT CONT ELEC 2L DC
   (f) 11F34, YAW DAMPER R
   (g) 11G17, FLT CONT ELEC 1R AC
   (h) 11G18, FLT CONT ELEC 1R DC
   (i) 11G26, FLT CONT ELEC 2R AC
   (j) 11G27, FLT CONT ELEC 2R DC

E. Procedure

S 914-036

CAUTION: DO NOT TOUCH THE YAW DAMPER MODULE BEFORE YOU DO THE PROCEDURE FOR THE ELECTROSTATIC DISCHARGE SENSITIVE DEVICES. ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE YAW DAMPER MODULE.

(1) Do the procedure for electrostatic discharge sensitive devices (AMM 20-41-01/201).

S 024-003

(2) Remove the Yaw Damper Module (AMM 20-10-01/401).

TASK 22-21-04-414-007

3. Install the Yaw Damper Module

A. General
   (1) The Yaw Damper Modules are in the E/E rack and are electrostatic sensitive. Special procedures are used to make sure they are not damaged.

B. References
   (1) AMM 20-10-01/401, E/E Rack Mounted Components
   (2) AMM 20-41-01/201, Electrostatic Sensitive Devices
   (3) AMM 24-22-00/201, Electrical Power - Control
   (4) AMM 27-21-00/501, Rudder and Rudder Trim Control System
   (5) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
   (6) AMM 31-41-00/501, Engine Indication and Crew Alerting System
   (7) AMM 32-09-02/201, Air/Ground Relays
   (8) AMM 33-16-00/501, Master Dim and Test
   (9) AMM 34-12-00/501, Air Data Computing System
   (10) AMM 34-21-00/501, Inertial Reference System

(11) AMM 34-61-00/501, Flight Management Computer System

C. Access

(1) Location Zone

119/120 Main Equipment Center (Left and Right)

(2) Access Panel

119AL Main Equipment Center Access

D. Procedure

S 914-037

CAUTION: DO NOT TOUCH THE YAW DAMPER MODULE BEFORE YOU DO THE PROCEDURE FOR THE ELECTROSTATIC DISCHARGE SENSITIVE DEVICES. ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE YAW DAMPER MODULE.

(1) Do the procedure for electrostatic discharge sensitive devices (AMM 20-41-01/201).

S 424-003

(2) Install the Yaw Damper Module (AMM 20-10-01/401).

E. Yaw Damper Module Test

S 984-048

WARNING: PERFORM THE FOLLOWING TWO STEPS BEFORE APPLYING EXTERNAL POWER; OTHERWISE THE PITOT PROBE CAN BECOME VERY HOT AND CAN BURN.

(1) Push the BAT switch on the pilot's overhead panel P5 to the ON position.

S 984-047

(2) Turn the STBY POWER swith on the P5 panel to the AUTO position.

S 864-009

(3) Supply electrical power (AMM 24-22-00/201).

S 864-011

(4) Make sure you close the DC circuit breakers before you close the AC circuit breakers for each power supply (FLT CONT ELEC).

S 864-026

(5) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:

(a) 11A18, YAW DAMPER L

(b) 11C6, FLT CONT ELEC 1L AC
(c) 11C7, FLT CONT ELEC 1L DC
(d) 11C8, FLT CONT ELEC 2L AC
(e) 11C9, FLT CONT ELEC 2L DC
(f) 11F34, YAW DAMPER R
(g) 11G17, FLT CONT ELEC 1R AC
(h) 11G18, FLT CONT ELEC 1R DC
(i) 11G26, FLT CONT ELEC 2R AC
(j) 11G27, FLT CONT ELEC 2R DC

S 864-033

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO A PERSON OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(6) For the left yaw damper system, pressurize the center hydraulic system (AMM 29-11-00/201).

S 864-013

(7) For the right yaw damper system, pressurize the left hydraulic system (AMM 29-11-00/201).

S 284-014

(8) Make sure these systems operate:
(a) Rudder and Rudder Trim Control System (AMM 27-21-00/501)
(b) Air Data Computing System (AMM 34-12-00/501)
(c) Inertial Reference System (AMM 34-21-00/501)
(d) Air/Ground Relays (AMM 32-09-02/201)
(e) Master Dim and Test System (AMM 33-16-00/501)
(f) Engine Indication and Crew Alerting System (AMM 31-41-00/501)
(g) 767-300 AIRPLANES;
Flight Management Computer System (AMM 34-61-00/501)

S 984-015
(9) Push the STATUS switch on the EICAS DISPLAY select panel on P9.

S 984-016
(10) Push the left or right YAW DAMPER switch/light on P5.

S 714-017
(11) Make sure the ON light comes on.

S 284-018
(12) Make sure that the applicable YAW DAMPER L or R INOP light is off.

S 984-019
(13) For the left yaw damper module, put the YAW DMPR test switch on P61 to L, then return switch to center.

S 284-027
(14) Make sure that the YAW DAMPER L INOP light is on.

S 284-028
(15) Make sure that the rudder position indicator on the lower EICAS display shows this sequence of rudder movement in less than 10 seconds:
  (a) the rudder moves approximately three degrees trailing edge right
  (b) the rudder moves approximately three degrees trailing edge left
  (c) the rudder goes back to the center

S 284-032
(16) Make sure that the YAW DAMPER L INOP light is not on 15 seconds after the action of the step before.

S 984-020
(17) For the right yaw damper module, put the YAW DMPR test switch on P61 to R, then return switch to center.

S 284-029
(18) Make sure that the YAW DAMPER R INOP light is on.

S 284-030
(19) Make sure that the rudder position indicator on the lower EICAS display shows this sequence of rudder movement in less than 10 seconds.
  (a) the rudder moves approximately three degrees trailing edge right
  (b) the rudder moves approximately three degrees trailing edge left
  (c) the rudder goes back to the center

S 284-031
(20) Make sure that the YAW DAMPER R INOP light is not on 15 seconds after the action of the step before.

S 714-038
(21) Push the DISPLAY button on the right Yaw Damper Module and make sure that the message NO FAULTS shows on the right Yaw Damper Module's fault display.

S 714-039
(22) Push the DISPLAY button on the left Yaw Damper Module and make sure that the message NO FAULTS shows on the left Yaw Damper Module's fault display.

F. Put the airplane back to its usual condition.

S 864-024
(1) Remove hydraulic power if it is not necessary (AMM 29-11-00/201).

S 864-025
(2) Remove electrical power if it is not necessary (AMM 24-22-00/201).

MODAL SUPPRESSION ACCELEROMETER - REMOVAL/INSTALLATON

1. General
   A. Two modal suppression accelerometers (MSA) are installed in the ceiling of the aft cargo compartment at station 1518, WL195, LBL72 and RBL72. You can access the MSAs through the aft cargo door.

TASK 22-21-05-014-003
2. Remove the Modal Suppression Accelerometer
   A. References
      (1) AMM 24-22-00/201, Electrical Power - Control
      (2) AMM 25-52-02/401, Lining - Containerized Cargo Compartment Ceiling
      (3) AMM 32-09-02/201, Air/Ground Relays
      (4) AMM 34-12-00/501, Air Data Computing System
      (5) AMM 34-21-00/501, Inertial Reference System
   B. Access
      (1) Location Zones
          822 Aft Cargo Door
          153/154 Aft Cargo Compartment

   C. Prepare for Removal

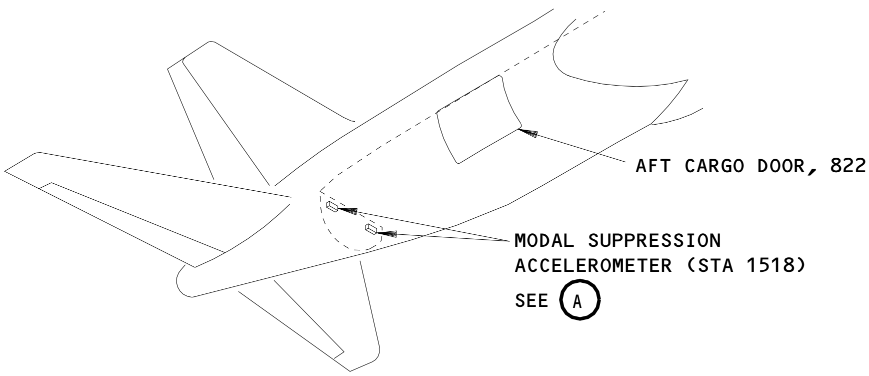
      S 864-021
      (1) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11C6, FLT CONT ELEC 1L AC
          (b) 11C8, FLT CONT ELEC 2L AC
          (c) 11G17, FLT CONT ELEC 1R AC
          (d) 11G26, FLT CONT ELEC 2R AC
   D. Procedure

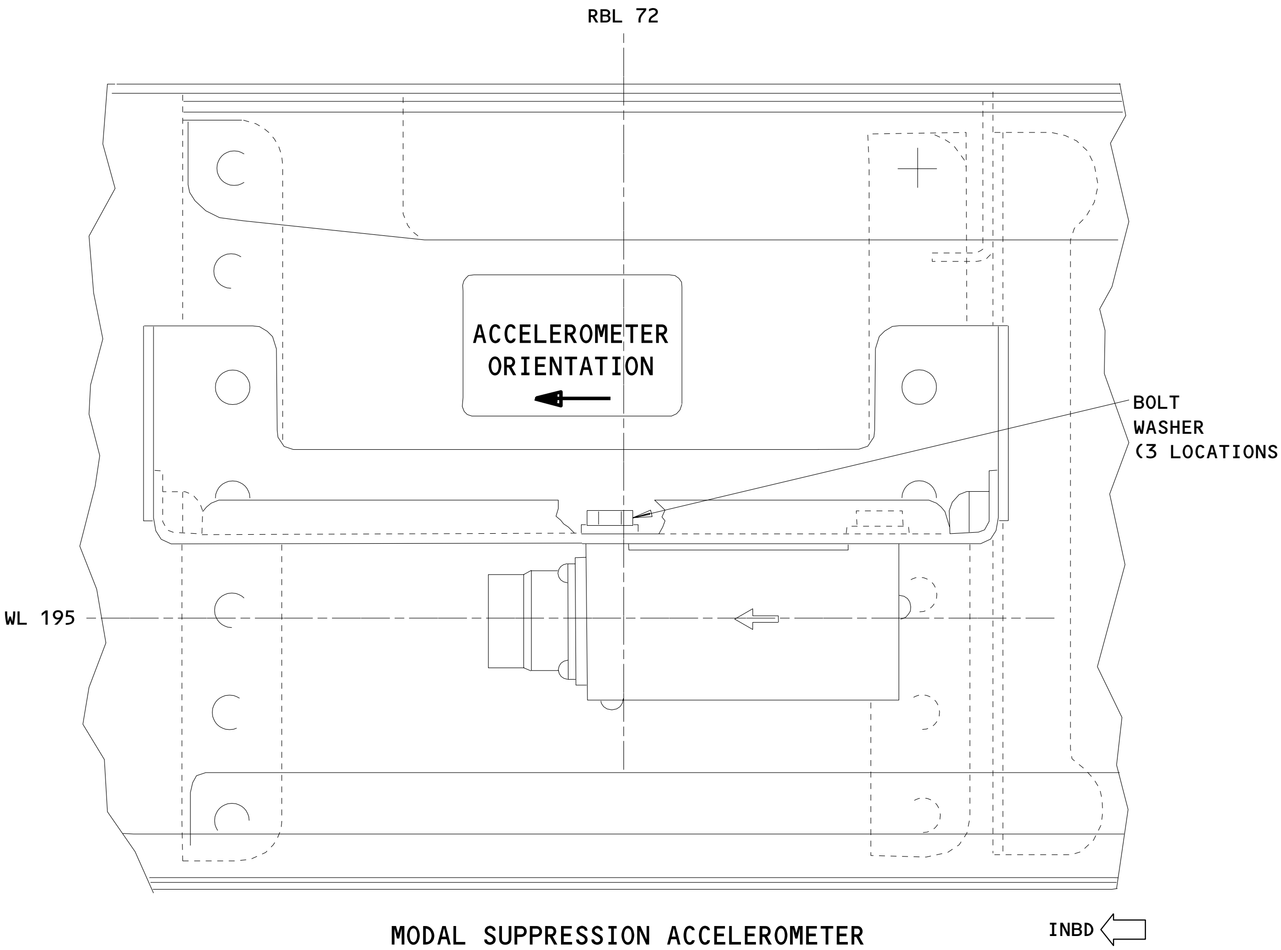
      S 034-004
      (1) Remove the electrical connector from the modal suppression accelerometer.

      S 024-019
      (2) Hold the modal suppression accelerometer in position and remove the mounting bolts and washers.

      S 024-011
      (3) Remove the modal suppression accelerometer from the aft cargo compartment.

TASK 22-21-05-414-010
3. Install the Modal Suppression Accelerometer
   A. References
      (1) AMM 24-22-00/201, Electrical Power - Control





MODAL SUPPRESSION ACCELEROMETER
(VIEW IN THE FORWARD DIRECTION)
(RIGHT SIDE IS SHOWN, LEFT SIDE IS EQUIVALENT)
A

Aft Modal Suppression Accelerometer Installation
Figure 401

(2) AMM 25-52-02/401, Lining - Containerized Cargo Compartment Ceiling
(3) AMM 32-09-02/201, Air/Ground Relays
(4) AMM 34-12-00/501, Air Data Computing System
(5) AMM 34-21-00/501, Inertial Reference System
(6) AMM 34-61-00/501, Flight Management Computer System

B. Access

(1) Location Zones

822	Aft Cargo Door
153/154	Aft Cargo Compartment

C. Prepare for Installation

S 864-020

(1) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
(a) 11C6, FLT CONT ELEC 1L AC
(b) 11C8, FLT CONT ELEC 2L AC
(c) 11G17, FLT CONT ELEC 1R AC
(d) 11G26, FLT CONT ELEC 2R AC

D. Procedure

S 014-002

(1) Open the aft cargo door.

S 024-001

(2) Remove the ceiling liner from the aft cargo compartment (Ref 25-52-02).

S 424-009

(3) Put the modal suppression accelerometer into position in the ceiling of the aft cargo compartment.

S 424-008

(4) Install the modal suppression accelerometer with washers and bolts.

S 434-007

(5) Connect the electrical connector to the modal suppression accelerometer.

E. Modal Suppression Accelerometer Test

S 864-005

(1) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
   (a) 11C6, FLT CONT ELEC 1L AC
   (b) 11C8, FLT CONT ELEC 2L AC
   (c) 11G17, FLT CONT ELEC 1R AC
   (d) 11G26, FLT CONT ELEC 2R AC

S 864-013

(2) Supply electrical power (AMM 24-22-00).

S 284-012

(3) Make sure that these systems operate:
   (a) Air/Ground Relays (AMM 32-09-02)
   (b) Air Data Computing System (AMM 34-12-00)
   (c) Inertial Reference System (AMM 34-21-00)
   (d) 767-300 AIRPLANES;
       Flight Management Computer System (AMM 34-61-00/501)

S 984-018

(4) Push the RESET button on the front panel of the yaw damper module.

S 984-017

(5) After 10 seconds, push the display button on the yaw damper module front panel.

S 284-006

(6) Make sure that ACCEL is not shown as a PRESENT FAULT message on the front panel of the yaw damper module.

F. Put the Airplane Back to Its Usual Condition

S 424-016

(1) Install the ceiling cargo liner in the aft cargo compartment (Ref 25-52-02).

S 414-015

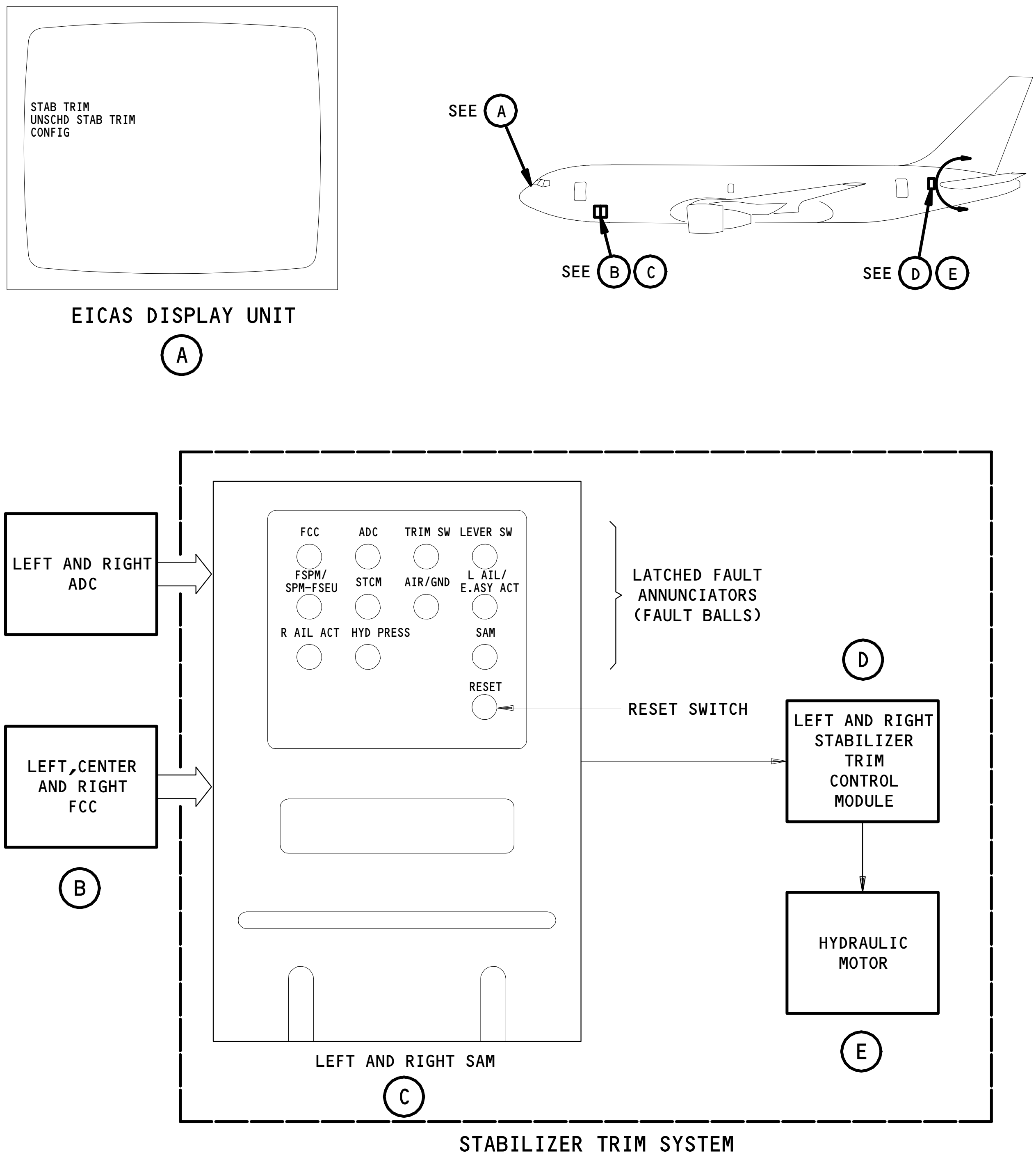
(2) Close the aft cargo door.

S 864-014

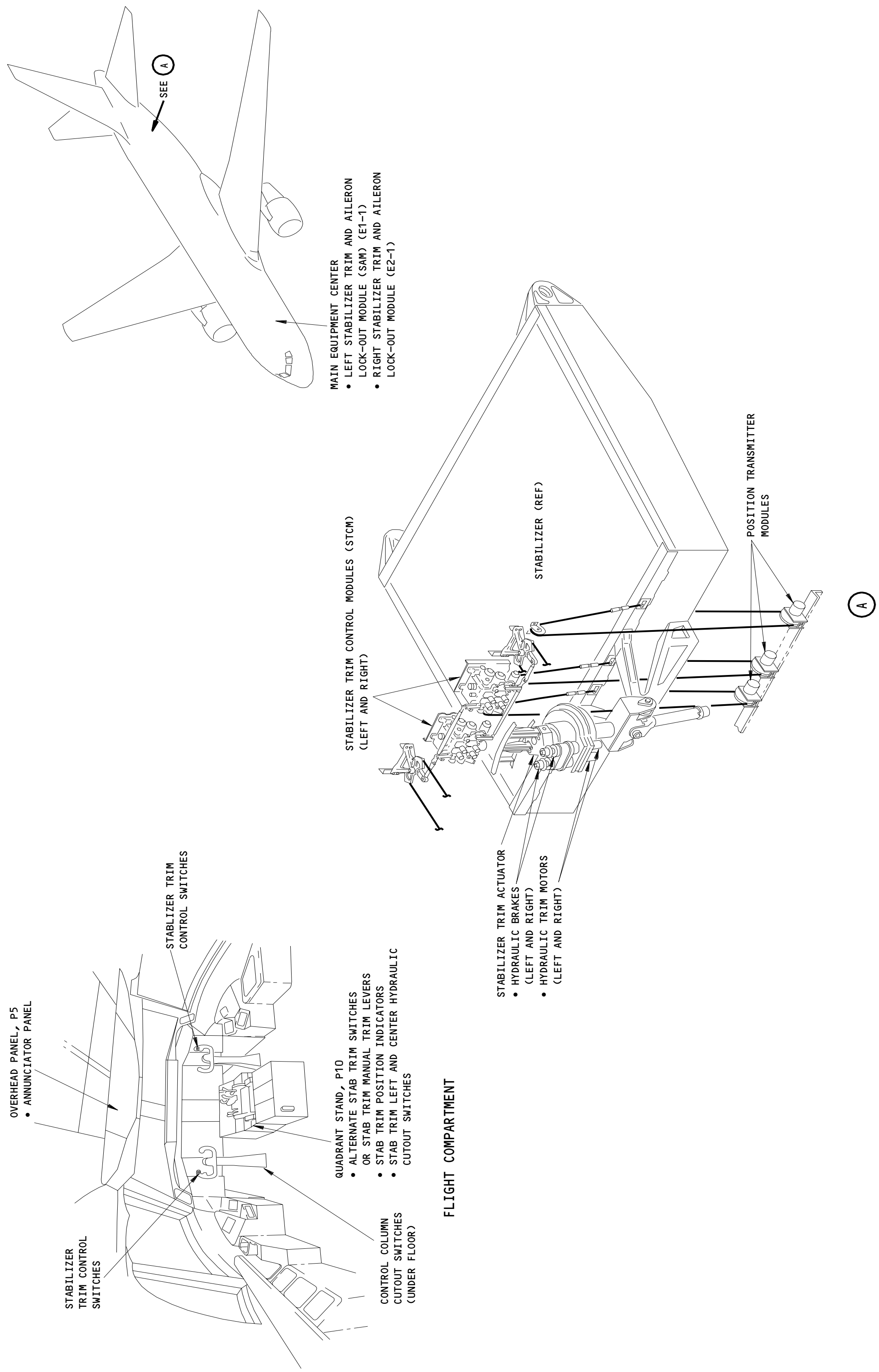
(3) Remove electrical power if it is not necessary (AMM 24-22-00).

AUTOMATIC STABILIZER TRIM SYSTEM - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The automatic stabilizer trim system provides automatic control of horizontal stabilizer position. When the autopilot is engaged, the flight control computer outputs stabilizer position commands. There are no line replaceable units that are only automatic stabilizer trim components. All components are part of the horizontal stabilizer trim system (Ref Chapter 27).
   B. This section covers autotrim operation. A general description of horizontal stabilizer trim system components is added to help understand the automatic stabilizer trim operation. For component details and complete stabilizer system description, refer to 27-41-00/001, Horizontal Stabilizer Trim System.
   C. The stabilizer trim system maintains the airplane in a longitudinally trimmed condition. The optimum pitch attitude is maintained during various flight conditions. The stabilizer may be trimmed manually or automatically. The stabilizer trim system has the following operational modes:
      (1) Manual and manual electric modes - the stabilizer is trimmed by pilot commands.
      (2) Autotrim mode - The autopilot Flight Control Computers (FCCs) trim the stabilizer to relieve sustained elevator loads.
      (3) The stabilizer trim system also provides a Mach Trim mode which provides trim commands as a function of MACH when the flaps are up and the following conditions are present:
         (a) The autopilot is not engaged.
         (b) The stabilizer is not being manually trimmed.
   D. Automatic Stabilizer Trim Component Locations (Fig. 2)
      (1) Flight Deck Components
         (a) The following stabilizer trim system components are in the control cabin. Stabilizer trim (STAB TRIM) manual trim levers (or alternate STAB TRIM switches), position indicators, and hydraulic cutout switches are adjacent to the throttle quadrant on the control stand (P10). The annunciator panel on overhead panel P5 contains the amber STAB TRIM and UNSCHED STAB TRIM lights. The stabilizer trim control switches are on the outboard horn of the captain's and first officer's control wheels. The control column stabilizer trim cutout switches are under the cabin floor.



Automatic Stabilizer Trim System
Figure 1



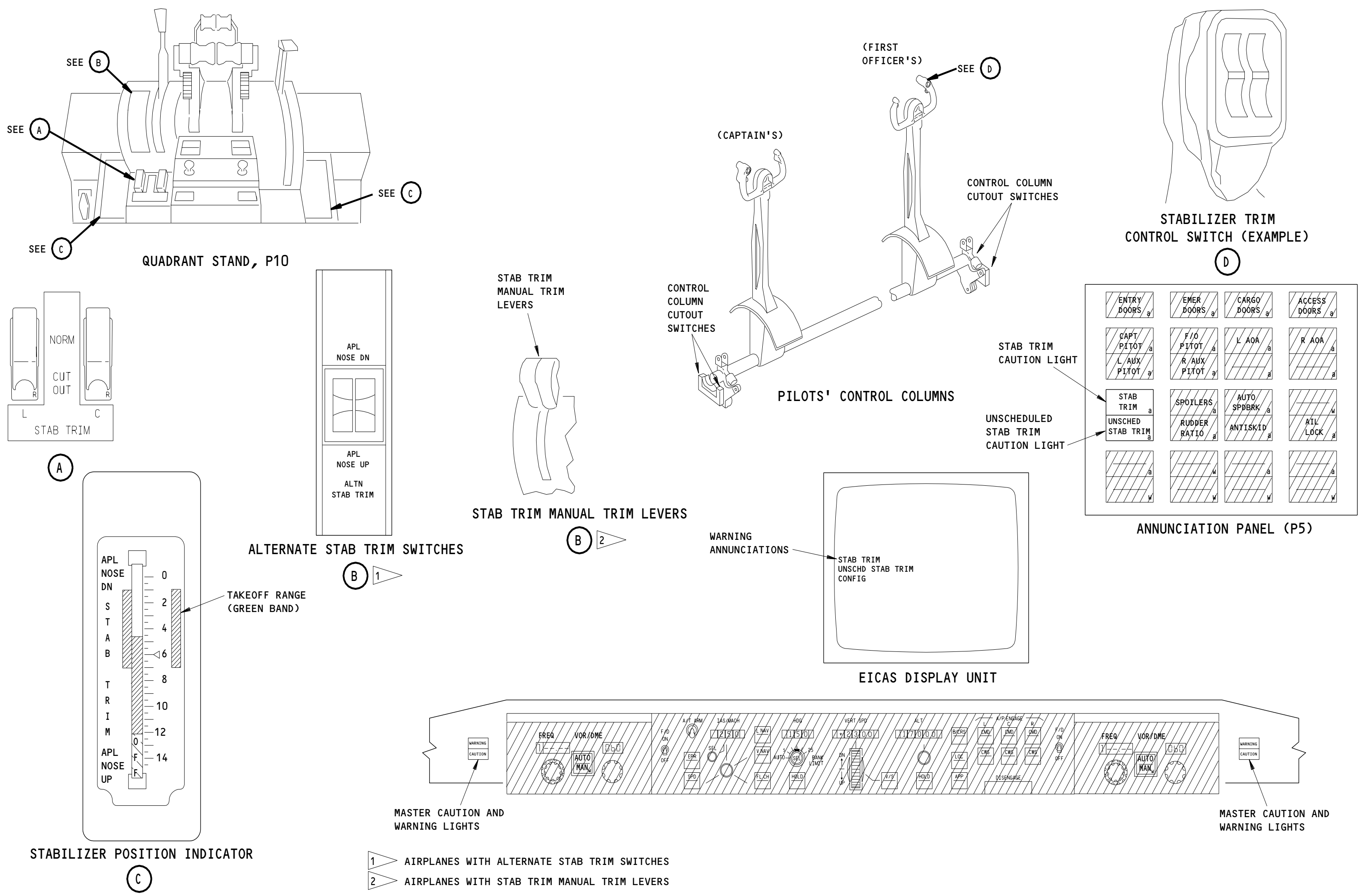
Stabilizer Trim Component Locations
Figure 2

(2) Main Equipment Center
  (a) The main equipment center contains the following stabilizer trim system components. Two Stabilizer Trim and Aileron Lock out Modules (SAMs) are in the left E1-1 and right E2-1 shelves, respectively. The SAM interfacing components in the main Elec/Elex equipment center are the Flight Control Computers (FCCs), Air Data Computers (ADCs), and the Flap/Stab Position Modules (FSPMs). The left, right, and center FCCs are in shelves E1-3, E1-4, and E1-5, respectively. The left and right ADCs are installed on shelves E1-3 and E2-3, respectively. The left, center, and right FSPMs are installed in the card file of equipment rack P50.

(3) Stabilizer Jackscrew Access Area
  (a) The stabilizer jackscrew area is accessible through the aft service access door (Ref 52-49-00/001 Exterior Service Doors). The area contains the Stabilizer Trim Actuator Assembly (STAA), left and right Stabilizer Trim Control Modules (STCMs), and left, center and right position transmitter modules. The STAA consists of two hydraulic motors and brakes coupled to a mechanical differential drive device.

E. Stabilizer Trim Flight Deck Components (Fig. 3)
(1) Stab Trim Position Indicators
  (a) The control stand contains two linear indicators that display stabilizer trim position. One is driven by the left stabilizer position transmitter. The other is driven by the right stabilizer position transmitter.
  (b) The size of the indicator panel is 6.5 inches long and 1.5 inches wide. The weight is approximately 2 pounds. The indicator has a vertical scale that displays stabilizer position. The scale reads from 0 units to 14.5 units. A 0 units indication represents a +2 degree, stabilizer leading edge up (airplane nose down) condition. A 14.5 unit reading represents a -12.5 degree, stabilizer leading edge down (airplane nose up) condition. A green band on each side of the scale indicates the allowable takeoff range. When the system is off, a black and orange striped OFF flag is in view at the bottom of the scale.

(2) Stab Trim Cutout Switches
  (a) The cutout switches enable system hydraulic pressure to be removed from the Stabilizer Trim Control Modules (STCMs). Two guarded switches are on the aisle stand, one for the left and one for the right STCMs. In the NORM position, 28 vdc standby bus power activates a motor-operated valve in the STCM that completes hydraulic pressure to the STCM solenoid valves. The CUTOUT position shuts off hydraulic pressure.



Stabilizer Trim Flight Deck Components
Figure 3

K18267

(3) Stab Trim Control Switches
 (a) The stab trim control switches enable the pilot or fist officer to input manual trim commands to both SAMs. Manual electric stabilizer trim control is provided by two single pole, three position, center off, thumb-operated rocker switches. The switches are on the outboard horn of each control wheel. Actuating both switches in the up direction issues simultaneous airplane nose up ARM and CONTROL commands. Down actuation issues airplane nose down ARM and CONTROL commands. When the trim switches are energized, both trim channels are activated and the stabilizer is driven at full-rate trim.

(4) Control Column Cutout Switches
 (a) 767-300 AIRPLANES;
 Each control column operates a pair of switches under the cabin floor, which allow electric trim commands to be interrupted by column action. The switches are wired in series with the SAMs arm command outputs. A column forward action greater than 1.5 degrees opens the switch in the airplane nose-up stabilizer trim command circuit. A column aft action greater than 1.5 degrees opens a switch in the airplane nose-down stabilizer trim circuit.
 (b) 767-200 AIRPLANES;
 Each control column operates a set of three switches under the cabin floor, which allow electric trim commands to be interrupted by column action. The switches are wired in series with the SAMs arm command outputs. A column forward action greater than 1.5 degrees opens the switch in the airplane nose-up stabilizer trim command circuit. Two switches wired in parallel interrupt airplane nose-down commands at two different control column angles. This is required because the elevator neutral shift program moves the columns aft when operating at low speeds. A column aft action greater than 1.5 at high speeds, or 3.0 degrees at low speeds opens a switch in the airplane nose-down stabilizer trim circuit.

(5) Failure Indicators
 (a) When stabilizer movement of more than 0.3 degree occurs without a valid stabilizer trim command the following indications are activated.
  1) Aural caution and discrete amber UNSCHED STAB TRIM light on the annunciator panel illuminates.
  2) An amber UNSCHD STAB TRIM caution message is displayed on the Engine Indication and Crew Alerting System (EICAS) display unit.
  3) Master caution light on glareshield panel P7 illuminates.
 (b) When a loss of full rate trim occurs during manual electric trim due to SAM, actuation, or airplane wiring faults, the following indications are activated:
  1) A discrete amber STAB TRIM light on the annunciator panel illuminates.
  2) An amber STAB TRIM advisory message is displayed on the EICAS display unit.

(c) When the stabilizer is positioned outside the takeoff greenband range and the throttles are advanced, the following indications are activated:
   1) Warning siren and discrete CONFIG red warning light on the P1 panel illuminates.
   2) A red STABILIZER warning message is displayed on the EICAS display unit.
   3) Master warning light on panel P7 illuminates.

(d) When maintenance is required, each SAM provides a first fail discrete to the EICAS computers. This will cause a STAB TRIM message to be generated on the EICAS maintenance page.

(e) When a single autopilot is engaged and an autotrim failure (dead trim, unscheduled trim, or AUTOTRIM signal from SAM is invalid as detected by the FCC) cannot be cleared by autosequencing, the following indications are activated.

   NOTE: Autosequencing can occur only when center FCC is engaged.

   1) A discrete amber AUTOPILOT caution light on P1 panel illuminates.
   2) Master caution light on panel P7 illuminates and an aural tone (owl) is sounded.
   3) A pitch mode fail (EFIS mode lined - out) is displayed on EADI display units.

(f) When two or more FCCs are engaged and an autotrim failure (detected by the FCC) during an approach is cleared by autosequencing, NO LAND 3 is displayed on Autoland Status Annunciators (ASA) after touchdown and disconnect of the autopilot. The ASAs are on the P1 and P3 panels.

(g) When two or more FCCs are engaged and an autotrim failure (detected by the FCC) cannot be cleared by autosequencing while the airplane is above Decision Height (DH), the following indications are activated.
   1) NO AUTOLAND is displayed on each ASA.
   2) A pitch mode fail (EFIS mode lined - out) is displayed on EADI display units.
   3) A discrete amber AUTOPILOT light on the panel P1 illuminates.
   4) Master caution light on panel P7 illuminates and the owl aural warning sounds.

(h) When two or more FCCs are engaged and an autotrim failure (detected by the FCC) cannot be cleared by autosequencing while the airplane is below Decision Height (DH), the following indications are activated after touchdown and disconnect of the autopilot.
   1) NO AUTOLAND is displayed on ASAs.
   2) A pitch mode fail (EFIS mode - out) is displayed on EADIs.
   3) A discrete amber AUTOPILOT light on panel P1 illuminates.
   4) Master caution light on panel P7 illuminates and the owl aural warning sounds.

F. Stabilizer Trim and Aileron Lock-Out Module (SAM) (Fig. 4)

(1) SAM Description

(a) Each SAM is part of a dual redundant system. The SAM performs the following functions:

1) Generates Mach trim commmands in response to input Mach number.

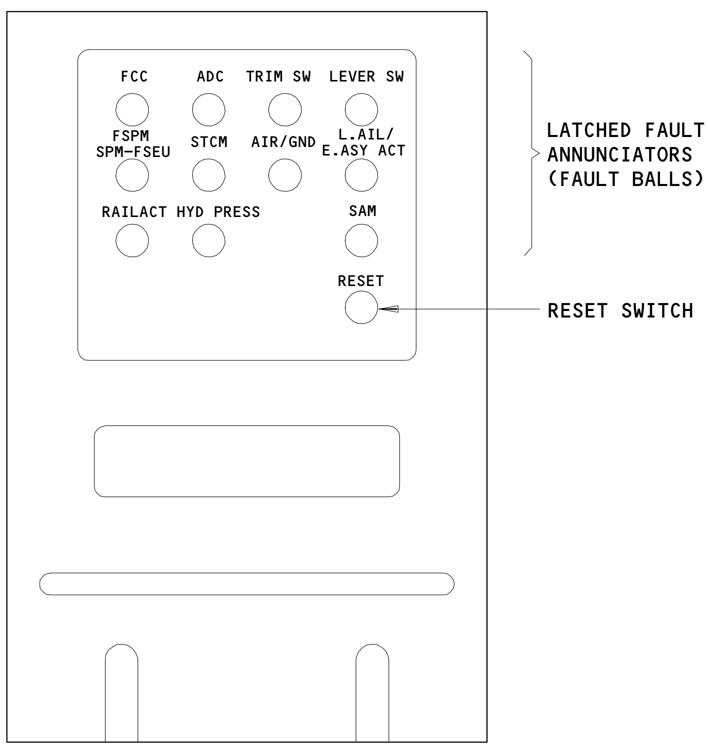
2) Selects the active trim source; either the Flight Control Computers (FCCs), Mach function of the SAM, or the control wheel trim switches.

3) Outputs both ARM and CONTROL electrical commands to the Stabilizer Trim Control Modules (STCMs).

4) Provides redundancy management, failure monitoring, and annunciation.

(b) Each module includes two parallel, non-identical microprocessors (ARM and CONTROL) that develop simultaneous trim commands. These commands are based on input data from the trim sources.

(c) The SAM provides logic circuits for system engage/disengage, stabilizer trim mode, and trim-source command selection. Other internal circuits are digital data ARINC 429 receivers, analog discrete input and output buffers, and cross-channel and self monitoring. Built-In-Test (BIT) fault indications are output to the EICAS display unit and annunciator panel.



Stabilizer Trim and Aileron Lockout Module (SAM)
Figure 4

(d) The front panel of the SAM contains 12 fault balls that indicate specific failures within the stabilizer trim system. All except the SAM fault ball are set by faults identified within the CONTROL microprocessor. The SAM fault ball is set by either ARM or CONTROL microprocessor. Other fault balls cannot be set after the SAM fault ball is set. The fault balls are black when not set, and yellow when set. Fault balls do not latch for input faults when airplane is on ground. The following fault ball indications are provided:
1) FCC - set by a left, center, or right flight control computer input fault.
2) ADC - set by a left or right Air Data Computer (ADC) fault discrete, or ADC input comparison fault.
3) COL TRIM SW - set by a manual-trim coincidence monitor detected failure (manual electric trim switches fault).
4) MAN LEVER SW - set by a standby trim lever switch fault (fault ball is set if the switch is closed or the lever is out of detent for longer than 30 seconds).
5) FSPM/SPM-FSEU - set by a continuously failed output from the stabilizer position level monitor for more than two seconds.
6) STCM - set by a fault detected by hydraulic brake pressure switch monitor or unscheduled trim monitor. (Fault ball can only be set if either of the hydraulic pressure discretes from the associated hydraulic system pressure switch indicate high hydraulic pressure).
7) AIR-GND RELAY - set by a first or second air/ground fault.
8) L. AIL LOCK ACT - set by a left aileron lockout actuator fault (not used by the stabilizer trim system).
9) R. AIL LOCK ACT - set by a right aileron lockout actuator fault (not used by the stabilizer trim system).
10) HYD PRESS SW - set by a system hydraulic pressure switch fault.
11) Growth provision fault ball (spare).
12) SAM - The monitor compares CONTROL and ARM trim commands. The SAM fault ball is set if a difference is detected for longer than two seconds.

(e) The SAM is able to reset itself if it is not in control and a SAM fault occurs. The SAM can also reset itself if a SAM fault has occured and the other SAM picks up control. If a SAM fault occurs again within one minute of an autoreset, the SAM will not be able to perform another autoreset. The SAM fault ball can reset itself if an autoreset is successful and no other SAM faults occur within one minute.

(f) When pressed, the RESET switch on the front panel of the SAM clears (resets) all fault balls unless the fault is still present. (Pressing RESET switch can clear STCM and TRIM SW fault balls even if fault still exist. LEVER SW fault ball cannot be clear until SAM goes through power-up test, fault no longer exist and RESET switch is pressed). The RESET switch is a single pole, momentary-pushbutton switch.

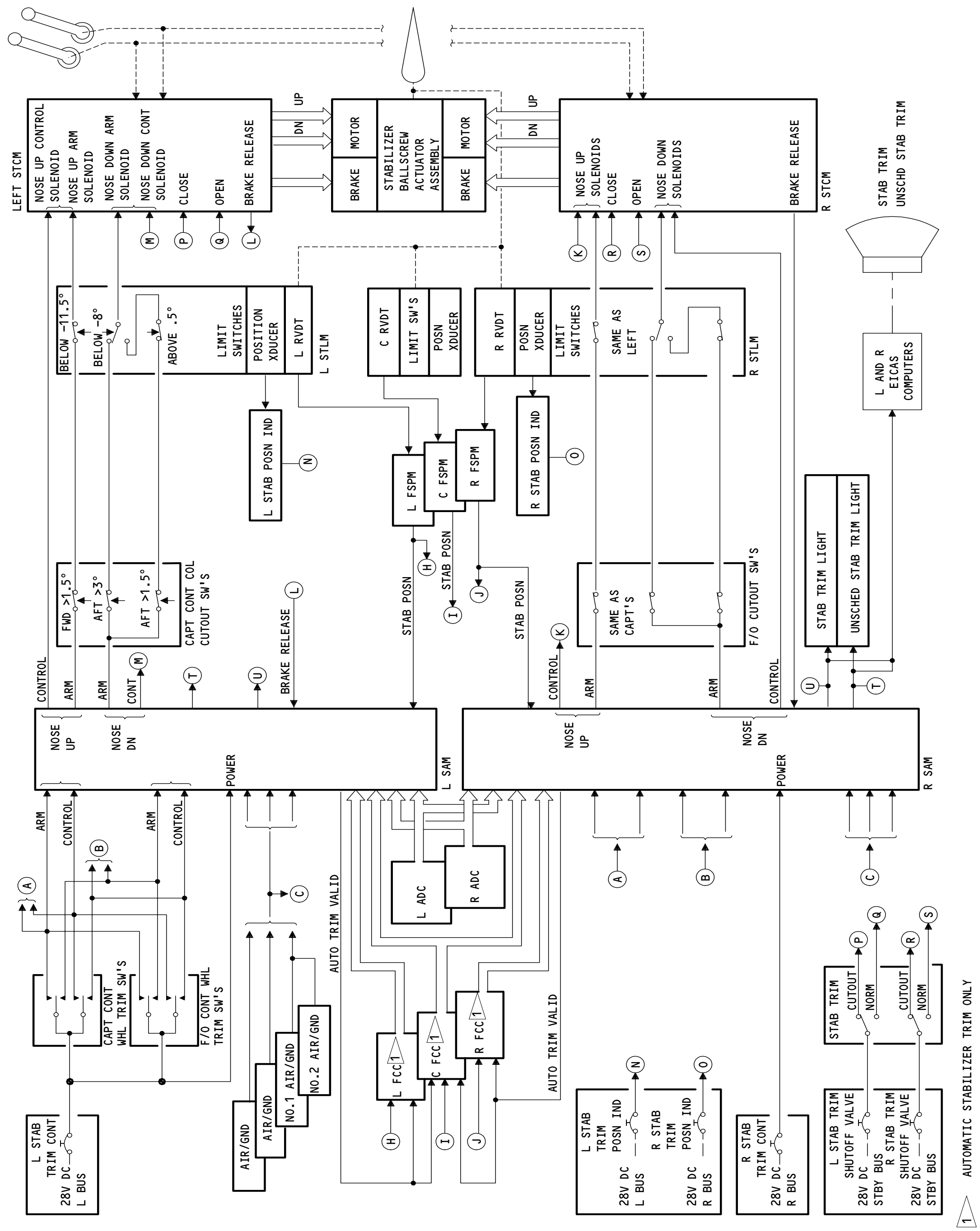
G. Stabilizer Trim System Block Diagram (Fig. 5)

(1) Stab Trim Manual Control

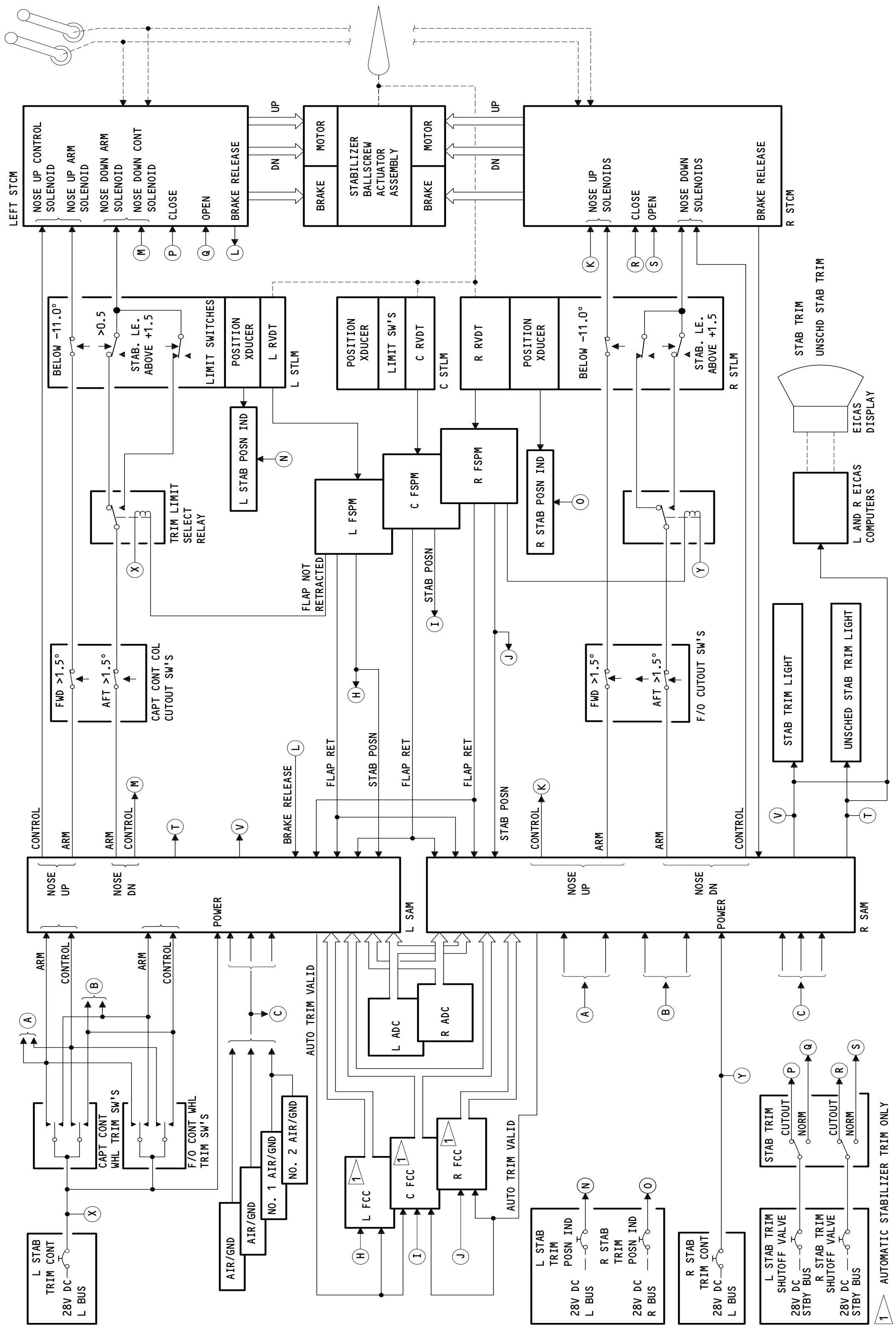
(a) The stabilizer ball-screw actuator consists of two independent hydraulic motors and brakes coupled to a mechanical drive device. The assembly moves the stabilizer at half-rate trim when powered by either the left or right Stabilizer Trim Control Module (STCM). Pressure supplied simultaneously from both STCMs produces full-rate trim. The brakes prevent air loads on the stabilizer from either back driving or overrunning the actuator.

(b) Each STCM has an ARM and a CONTROL hydraulic valve. The manual stabilizer trim controls (P10) are connected to the STCMs. One control activates the ARM valve in both STCMs and the other control activates the CONTROL valves. Actuation of the manual trim controls trim the stabilizer at full rate and overrides all electric trim commands.

(c) When the electric trim switches are engaged, both trim channels are activated, and the stabilizer is driven at full-rate trim. The STCM responds to electrical trim commands by applying hydraulic pressure to the motors on the actuator. A hydraulic brake on the actuator assembly is released by the same hydraulic pressure. A hydraulic pressure signal from the elevator feel computer modulates the flow rate of fluid to the actuator via a rate valve within the STCM. The hydraulic pressure signal is a function of airspeed. This varies the trim rate between 0.1 and 0.25 degree/sec for one active STCM, and between 0.2 and 0.5 degree/sec for two active STCMs. Between high and low trim rates, the rate varies linearly as a function of elevator feel pressure. Actuation of the manual electric trim switches overrides autotrim mode, except when in Autoland mode. When one FCC is overridden by manual electric trim switches, a red A/P DISC light on panel P2, and the red master warning light on panel P7 are illuminated. The siren is also activated. Pressing the autopilot disengage button on the control wheel turns off all warnings.



Stabilizer Trim System Block Diagram
Figure 5 (Sheet 1)



Stabilizer Trim System Block Diagram
Figure 5 (Sheet 2)

(d) Each control column is coupled to a set of switches that enable electric trim commands to be interrupted (solenoid valve voltage removed) by column action. A column forward action opens the switch in the airplane nose-up stabilizer command circuit. A column aft action opens the switch in the airplane nose-down command circuit. Limit switches inside left and right Stabilizer Trim Limit Switch and position transmitter module (STLM), removes solenoid valve voltage when stabilizer position limits are reached.

H. Cutout Limits Schedule (Fig. 6)

(1) Cutout and Limit Switch Activation

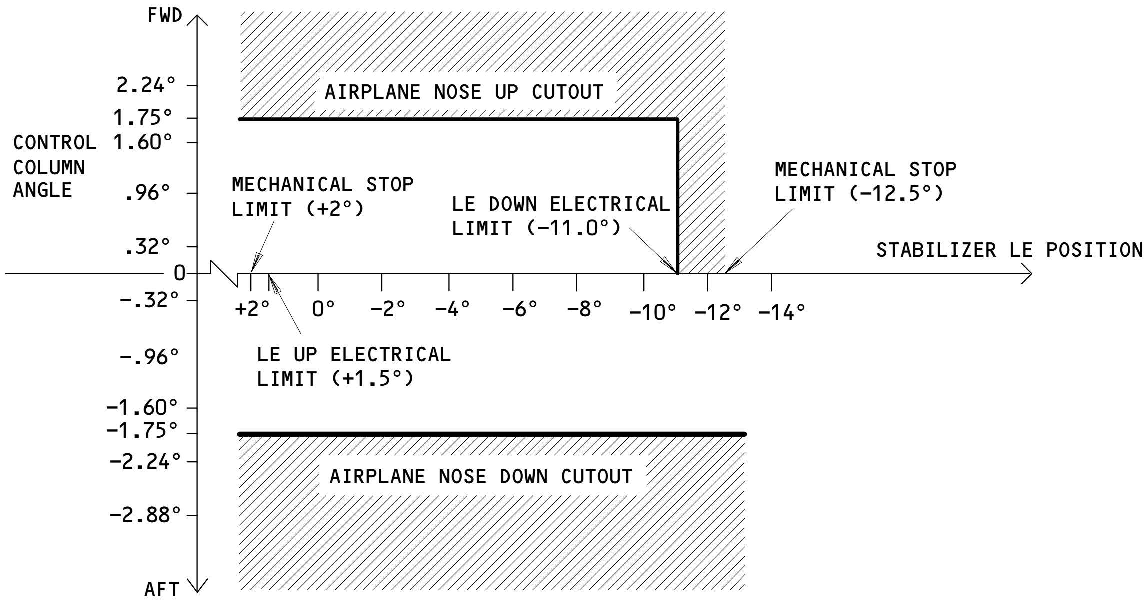
(a) The forward and aft control column cutout switches and the nose-up and nose-down limit switches operate as follows:

1) 767-200 AIRPLANES;
A control column forward movement greater than 1.5 degrees (approximately 7.5 degrees down elevator) opens the nose-up ARM inputs to the STCMs. If the stabilizer position is not within +2.0 to -11.5 degrees, the limit switch opens the STCM input.

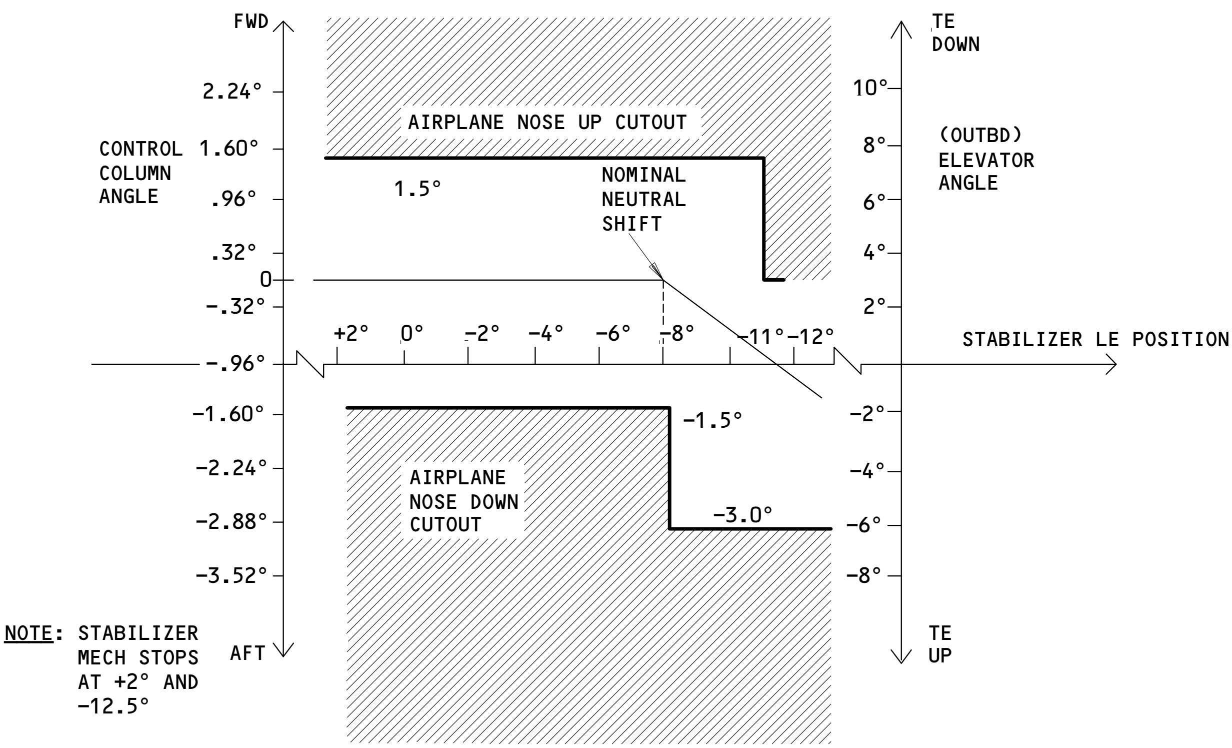
2) 767-300 AIRPLANES;
A control column forward movement greater than 1.75 degrees (approximately 7.5 degrees down elevator) opens the nose-up ARM inputs to the STCMs. If the stabilizer position is not within +2.0 to -11.5 degrees, the limit switch opens the STCM input.

3) 767-200 AIRPLANES;
A control column aft movement greater than 1.5 degrees (approximately 2 degrees up elevator) opens the nose-down ARM inputs to the STCMs. The limit switch opens the STCM input if the stabilizer position is not within +0.5 to -8.0 degrees.

4) 767-300 AIRPLANES;
A control column aft movement greater than 1.75 degrees (approximately 2 degrees up elevator) opens the nose-down ARM inputs to the STCMs. The limit switch opens the STCM input if the stabilizer position is not within +0.5 to -8.0 degrees.



767-300 AIRPLANE



767-200 AIRPLANE

Cutout Limits Schedule
Figure 6

5) 767-200 AIRPLANES;
a control column aft movement greater than 3.0 degrees (approximately 6 degrees up elevator) activates the second cutout switch. The limit switch opens if the stabilizer position is not within -8.0 to -11.5 degrees.

(b) 767-200 AIRPLANES;
stabilizer mechanical stops are at +2.0 and -12.5 degrees. Switch operating limits accommodate the neutral shift schedule (3 degrees elevator trailing-edge down until at -8 degrees of stabilizer position). These limits control the maximum range that the stabilizer may travel.

I. Configuration

(1) All airplanes have SAM -116/-117 installed.

NOTE: SAM -117 is optional to SAM -116.

2. Operation

A. Functional Description

(1) Automatic Stabilizer Trim Block Diagram (Fig. 5)

(a) The purpose of the Flight Control Computers (FCCs) in the stabilizer trim system is to provide trim commands when at least one channel of the autopilot is engaged. All three FCCs may be used; either one when in the cruise mode, or one or more when in the autoland mode. The left and right FCCs provide inputs to their respective SAMs. The center FCC provides inputs to both SAMs. Only one SAM is engaged at any time, with the first FCC engaged selected as the autotrim source. Priority for simultaneous FCC engagement is left, center, right.

(b) The FCCs generate trim commands in response to an elevator deflection exceeding a preset authority threshold. Trim commands applied to the SAM cause the stabilizer to assume the elevator trim load. Stabilizer position signals are fed back to the FCCs from the associated Flap Stabilizer Position Modules (FSPM). Trim commands stop when the elevator is within the elevator authority threshold limits.

(c) The FCC generates a digital trim engage discrete and a separate analog autotrim discrete. These cause the SAM to select the FCC as the active trim source. With only one SAM engaged at any time in autotrim mode, a single STCM hydraulic output drives the stabilizer motor at half-rate speed (0.10 degrees/sec to 0.25 degrees/sec). The elevator feel computer determines the pressure applied at the STCM, which determines stabilizer trim rate.

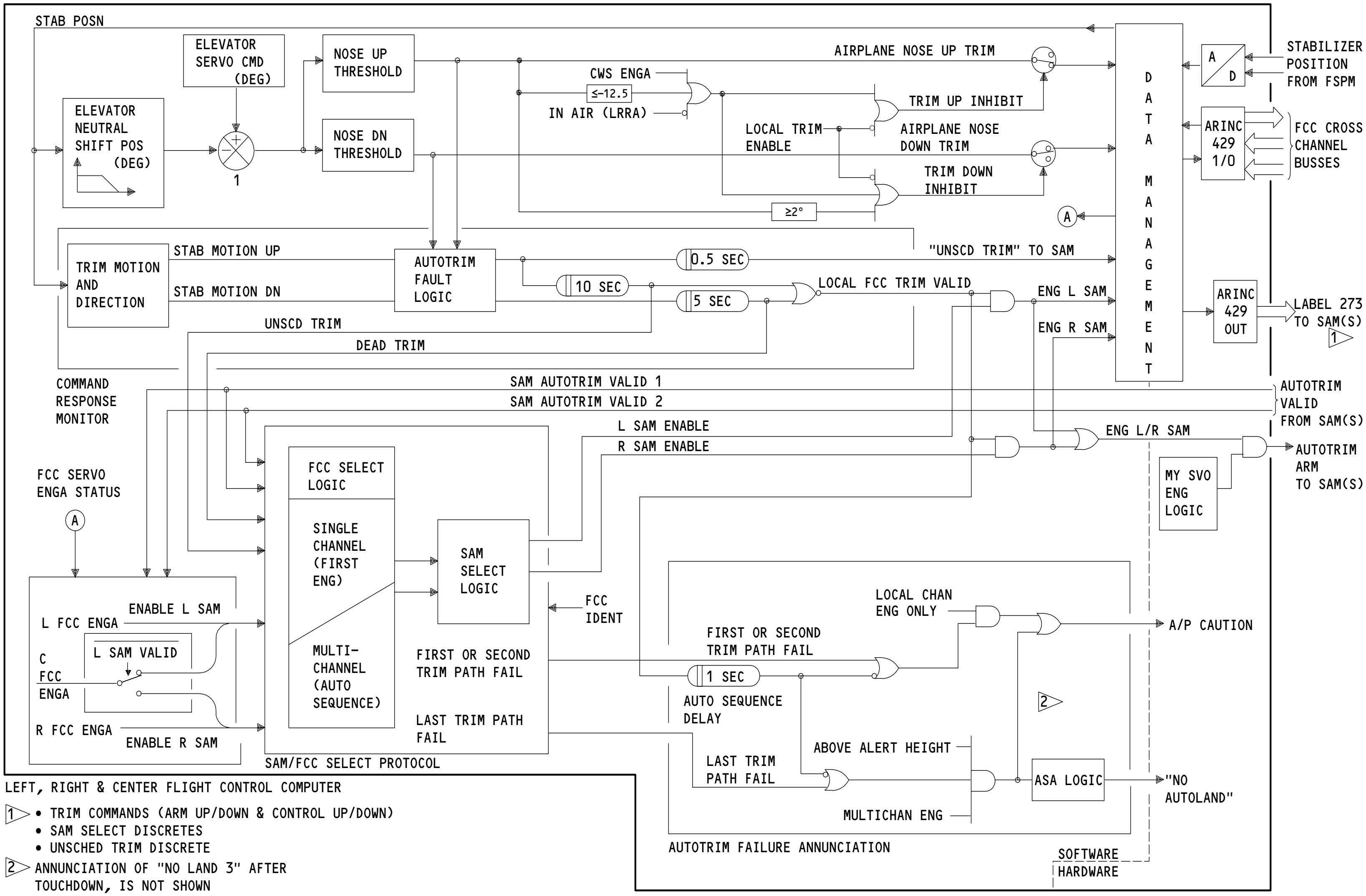
(d) 767-300 AIRPLANES;
Autotrim mode also provides full rate autotrim during multi channels approach. With one autopilot engaged, half-speed autotrim shall be provided when one SAM engaged. The engaged SAM will command the STCM to drive the stabilizer. The remaining SAM will be inhibited. Left and right SAM simultaneously command the stabilizer to a full rate trim when full rate trim request is received from an engaged Flight Control Computer during multi channels approach.

(2) Automatic Stabilizer Trim FCC Logic (Fig. 7)

(a) Autotrim FCC Operation

1) The FCC provides trim commands when the autopilot is engaged and manual trim mode is not selected. The engaged FCC generates a digital trim-engage discrete and a separate analog autotrim arm discrete. The ENG L SAM or ENG R SAM discretes are separate bits in ARINC 429 word 273. The AUTOTRIM ARM signal is the analog discrete. Both signals are provided to each channel of the SAM. When the SAM accepts the autotrim engage request, an AUTOTRIM VALID analog discrete signal is transmitted to the FCC.

2) The three FCCs communicate with the SAM via ARINC 429 buses. Protocol logic within each FCC determines which computer trims the stabilizer. The FCCs monitor each other. In the autotrim mode, the command response monitor in FCC performs loop monitoring to detect autotrim failure. Autotrim failure occurs when stabilizer does not respond to computed commands, stabilizer movement is inconsistent with computed commands or when analog AUTOTRIM VALID from SAM becomes invalid. When single autopilot is engaged, a detected autotrim failure which cannot be cleared by internal autosequencing (center FCC only) disables the active trim control path by disengaging the appropriate SAM. Sequencing to another trim channel following a trim failure can occur only if the center autopilot is engaged, since the autopilot interfaces with both SAMs. When the center autopilot is engaged, the left SAM is always selected. A subsequent trim failure requires the FCC to initiate sequencing to the right SAM. When more than one autopilot is engaged, sequencing to another trim channel is attempted when the current SAM - driving autopilot disconnects, or the active trim control path fails the local command-response monitor. The FCC autosequencing cycle is Left - Right - Center - Left.



1 • TRIM COMMANDS (ARM UP/DOWN & CONTROL UP/DOWN)
• SAM SELECT DISCRETES
• UNSCHED TRIM DISCRETE

2 ANNUNCIATION OF "NO LAND 3" AFTER TOUCHDOWN, IS NOT SHOWN

Automatic Stabilizer Trim FCC Logic
Figure 7

18430

3) Elevator commands greater than 0.25 degrees and stabilizer position determine trim commands. Trim up or trim down ARM and CONTROL signals are digitally outputed to the SAM. Trim motion is then monitored by each FCC to determine autotrim failure. If the TRIM VALID signal is not valid, AUTOTRIM ARM, and ENG R SAM or ENG L SAM signals are not transmitted.

(3) Automatic Stabilizer Trim Schematic (Fig. 8)

(a) Autotrim SAM Operation

1) The SAM contains ARM and CONTROL microprocessors to form a redundant system. Each processor receives inputs from its corresponding FCC and center FCC. The AUTOTRIM ARM input is an analog signal from the FCC. Digital inputs from the FCC are the following:

a) UP or DN CONTROL and ARM command inputs

b) Left or right SAM ENG inputs

2) Unscheduled stabilizer motion is detected by the corresponding SAM (stabilizer moves more than 0.55 degree in either direction without a valid FCC autotrim command), a ground is provided by the SAM to turn on the UNSCHED STAB TRIM light on panel P5. An UNSCHD STAB TRIM message is also displayed on EICAS display unit to alert the flight crews.

3) The FCC inputs are monitored to determine which FCCs are engaged. The input monitors also detect faults that set the FCC fault ball as follows:

a) If four or more parity errors occur within eight successive samples, a fault is set for the corresponding FCC. The new value is discarded and the previous valid value is used.

b) Autotrim command data which is identified as FAILED by the ARNIC 429 status matrix is discarded and no trim data substituted. If four or more autotrim command inputs within eight successive samples are indicated as FAILED, a fault is set for the corresponding FCC (detected by the ARINC 429 sign status matrix).

c) If an autotrim command word is not received for 125 milliseconds for two consecutive frames when required by the SAM, a fault is set for the corresponding FCC. Old data is used when new data is not received for one frame.

d) Input coincidence monitoring detects invalid autotrim commands. An invalid command is defined as non-coincidence of ARM and CONTROL of autotrim digital discretes, coincidence of opposite direction ARM or CONTROL autotrim digital discretes, or an ARM or CONTROL autotrim command without digital ENGAGE or analog ARM discretes. If an invalid command is received for two or more successive samples, a fault is set for the corresponding FCC.

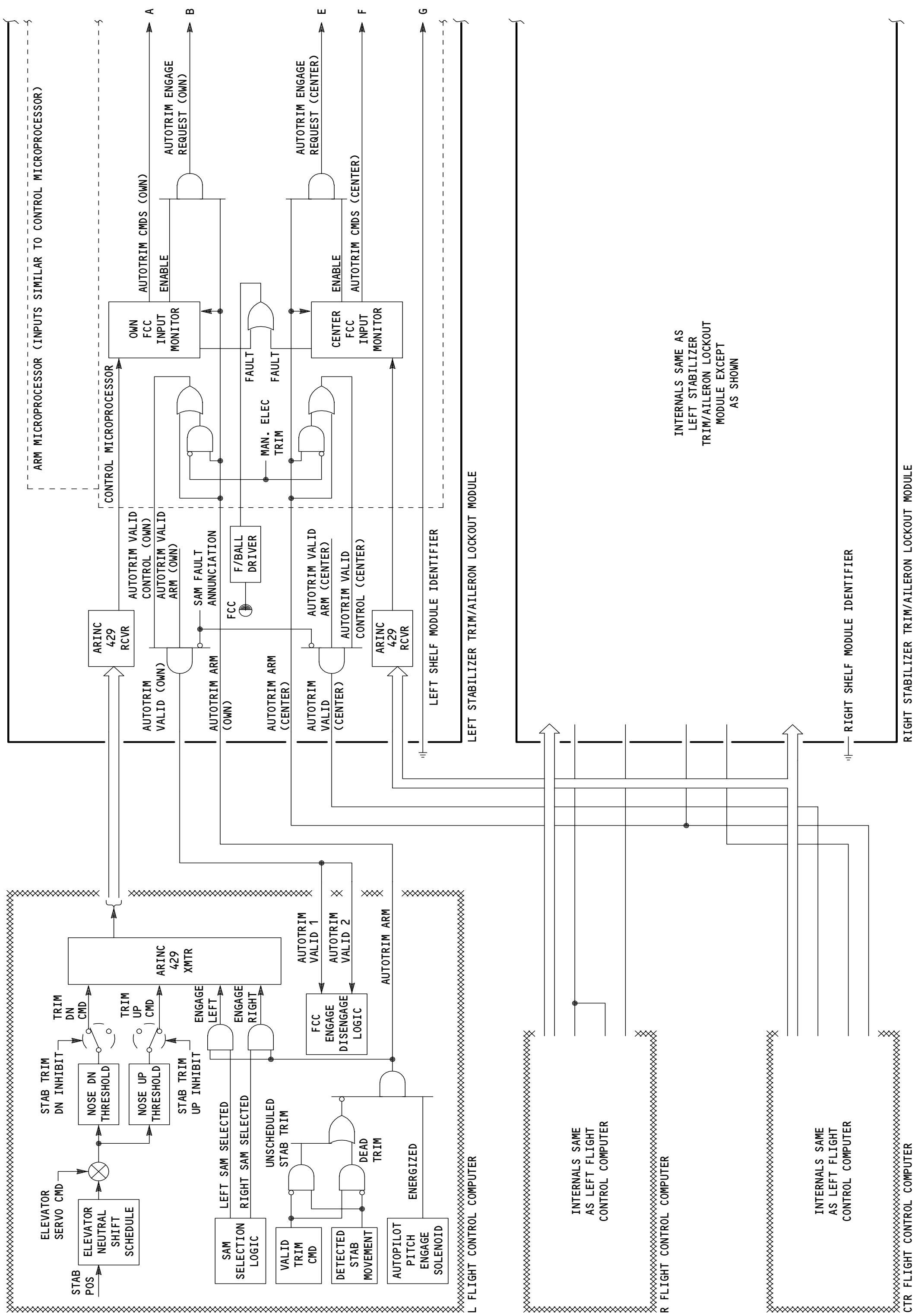
4) An FCC fault is cleared in-air if valid data is continuously received for two seconds from the corresponding FCC.

5) Autotrim inputs identified by the status matrix as TEST or NO COMPUTED DATA are ignored by the SAM. The corresponding autotrim and enable valid discretes are set invalid. The corresponding FCC is not considered as fail unless other monitors are tripped. If enable conditions are met, an AUTOTRIM VALID signal is transmitted to the engaged FCC.

6) Autotrim UP and DOWN (ARM and CONTROL) commands are delayed 3.5 seconds. This allows for mode switching and intermittent trim signals. Autotrim commands for the stabilizer trim control modules are output via the autotrim output coincidence monitor. The monitor compares CONTROL and ARM trim commands. The SAM fault ball is set if a difference is detected for longer than two seconds.

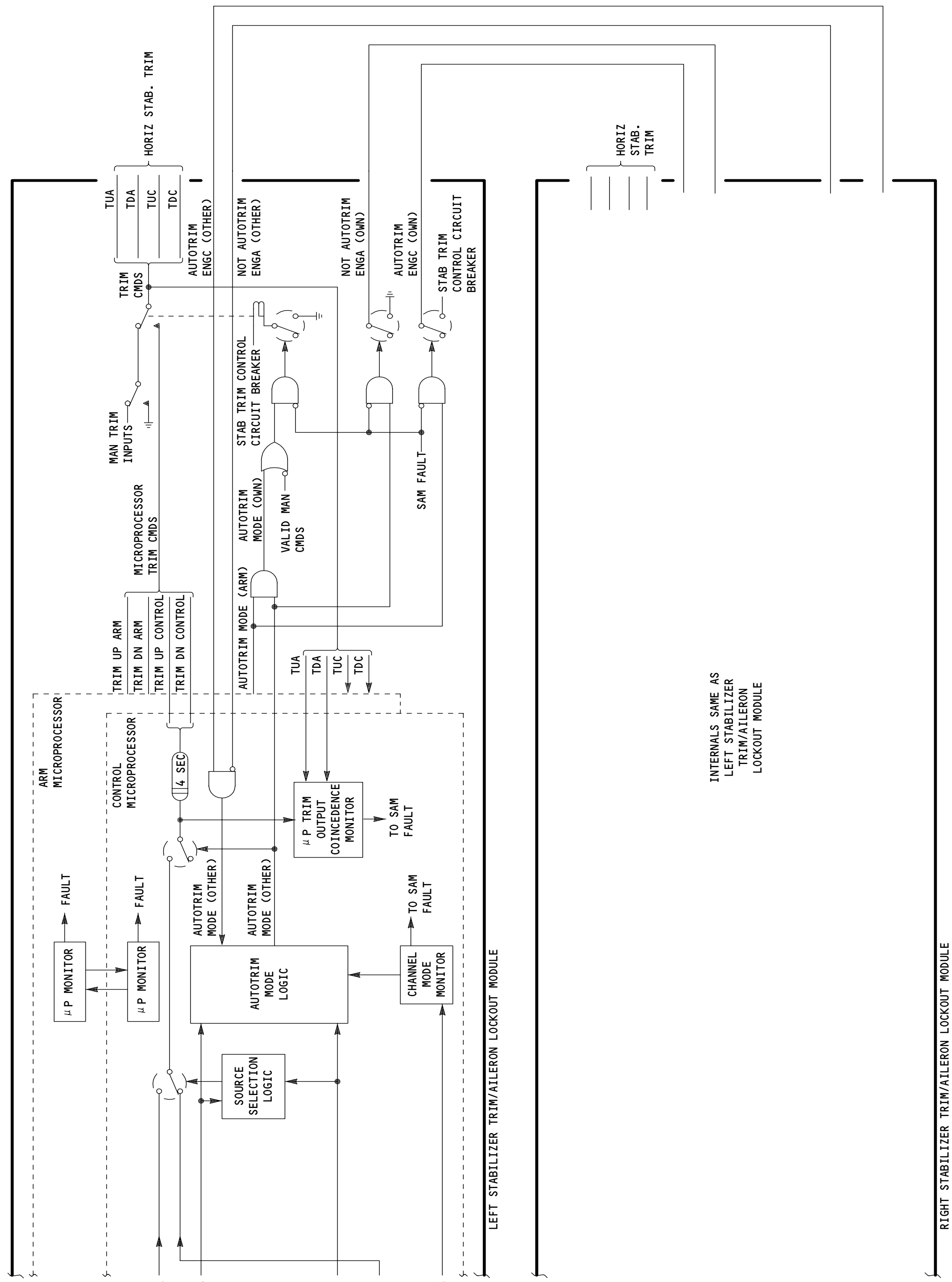
7) The SAM monitor sets the SAM fault ball when the following faults are detected:

a) Channel code discrete and airplane type code discrete faults, (If discrepancy persists for more than two seconds).



Automatic Stabilizer Trim Schematic
Figure 8 (Sheet 1)

211115



Automatic Stabilizer Trim Schematic
Figure 8 (Sheet 2)

211127

b) Mode coincidence, background, and ARM/CONTROL watchdog faults.
c) Aileron lockout actuator and power-up test faults.
d) Option Code Discretes faults that persist for more than two seconds will cause SAM fault ball to set.

AUTOMATIC STABILIZER TRIM SYSTEM - ADJUSTMENT/TEST

1. General
   A. This procedure gives the operational test and the system test of the Automatic Stabilizer Trim System. The two tests can be done without the use of external equipment. There are no adjustments for the Automatic Stabilizer Trim System.

TASK 22-22-00-715-001

2. Operational Test - Automatic Stabilizer Trim System

   A. General
      (1) The operational test examines the ability of the FCC to command the Stabilizer Trim and Elevator Asymmetry Limit Module (SAM) to drive the horizontal stabilizer up or down. It checks the FCC-to-SAM interface.
   B. References
      (1) AMM 22-10-00/501, Autopilot (Flight Control)
      (2) AMM 22-00-02/201, Autoflight BITE
      (3) AMM 24-22-00/201, Electrical Power - Control
      (4) AMM 27-41-00/501, Horizontal Stabilizer Trim Control System
      (5) AMM 27-51-00/201, Trailing Edge Flap System
      (6) AMM 27-61-00/201, Spoiler/Speedbrake Control System
      (7) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System
      (8) AMM 31-41-00/501, Engine Indication and Crew Alerting System
      (9) AMM 32-09-02/201, Air/Ground Relays
      (10) AMM 34-11-00/201, Pitot - Static System
      (11) AMM 34-12-00/501, Air Data Computing System
      (12) AMM 34-21-00/501, Inertial Reference System
   C. Access
      (1) Location Zones
          119/120 Main Equipment Center
          211/212 Flight Compartment

      (2) Access Panel
          119AL Main Equipment Center

   D. Prepare For Test

      S 865-002
      (1) Supply electrical power (AMM 24-22-00/001).

      S 285-003
      (2) Make sure these systems operate.
          (a) Horizontal Stabilizer Trim Control System (AMM 27-41-00/501).
          (b) Air Data Computing System (AMM 34-12-00/501).
          (c) Autopilot (Flight Control) (AMM 22-10-00/501).
          (d) Engine Indication and Crew Alerting System (AMM 31-41-00/501).

S 285-004

(3) Make sure that the control column is at its neutral position and not restrained.

S 865-005

(4) Make sure that these circuit breakers on the overhead circuit breaker panel, P11, are closed:
   (a) 11A35, IND LIGHT 3
   (b) 11C6, FLT CONT ELEC 1L AC
   (c) 11C7, FLT CONT ELEC 1L DC
   (d) 11C8, FLT CONT ELEC 2L AC
   (e) 11C9, FLT CONT ELEC 2L DC
   (f) 11C12, STAB TRIM SHUTOFF L
   (g) 11C13, STAB TRIM SHUTOFF CENTER
   (h) 11C14, FLAP/STAB POS SENSING
   (i) 11C17, ALIERON LOCKOUT L
   (j) 11C18, AILERON LOCKOUT R
   (k) 11C29, LANDING GEAR POSITION AIR/GND SYS 2 ALTN
   (l) 11C30, LANDING GEAR POSITION AIR/GND SYS 1
   (m) 11G17, FLT CONT ELEC 1R AC
   (n) 11G18, FLT CONT ELEC 1R DC
   (o) 11G26, FLT CONT ELEC 2R AC
   (p) 11G27, FLT CONT ELEC 2R DC
   (q) 11H10, STAB TRIM POS IND LEFT
   (r) 11H11, STAB TRIM CONT LEFT
   (s) 11H19, STAB TRIM POS IND R
   (t) 11H20, STAB TRIM CONT R
   (u) 11J17, FLAP/STAB POS SENSING L
   (v) 11J26, FLAP/STAB POS SENSING R
   (w) 11U23, POSITION AIR/GND SYS 2 (767-300 airplanes)
       11U24, POSITION AIR/GND SYS 2 (767-200 airplanes)

S 865-014

WARNING: KEEP PERSONS AND EQUIPMENT AWAY FROM ALL CONTROL SURFACES WHEN HYDRAULIC POWER IS SUPPLIED. AILERONS, ELEVATORS, RUDDER, FLAPS, SLATS, SPOILERS, AND STABILIZER ARE FULLY POWERED SURFACES. INJURY TO PERSONS OR DAMAGE TO EQUIPMENT CAN OCCUR WHEN HYDRAULIC POWER IS SUPPLIED.

(5) Pressurize the left, right, and center hydraulic systems (AMM 29-11-00/201).

S 865-173

(6) Put the L and C STAB TRIM cutoff switches in the NORM position.

E. Faultball Reset Procedure

S 865-212

(1) If any faultballs or overhead panel indicator lights are activated during a test, reset the SAM. Recycle CSEU AC circuit breakers and/or push the RESET button on each SAM.

F. Test Automatic Stabilizer Trim System

S 865-017

(1) If you use the MCDP remote control unit, do these steps.
   (a) Open the P6-5 door in the main power distribution panel, P6, and remove the dust cover from connector D1447.
   (b) Connect the remote MCDP to connector D1447.
   (c) Press the CONF/MCDP switch on EICAS maintenance panel (P61) to obtain the MCDP display on the lower EICAS display unit.

   NOTE: The same messages displayed on the MCDP in the main equipment center are displayed on the lower EICAS display. The message MCDP FLT FAULTS or MCDP GRD TEST is displayed to identify the MCDP mode.

   (d) Make sure the EICAS lower display unit shows the message MCDP OFF.

S 985-018

(2) Put the MCDP ON/OFF switch in the ON position.

S 285-019

(3) Make sure the MCDP displays LAST FLT FAULTS?.

S 985-020

(4) Operate the GRD TEST switch and make sure the GRD TEST mode is engaged.

S 285-021
(5) Use the GRD TEST SEL UP and DOWN switches or the NO/SKIP switch for test selection. Make sure the display indicates 65 STAB TRIM TEST?.

S 285-022
(6) To arm the test, push the YES/ADV switch and make sure the MCDP shows 65 IN PROGRESS.

S 285-025
(7) Make sure the MCDP shows 65 VFY HYD ON - ADV then push YES/ADV switch.

S 285-027
(8) Make sure the MCDP shows 65 A/P ENG L TO CMD then push the L CMD button on the mode control panel on the P55 panel.

S 285-162
(9) Make sure the MCDP shows 65 IN PROGRESS.

S 285-028
(10) Monitor the stabilizer indicator and the stabilizer motion.

S 285-029
(11) Make sure that these movements occur:
- (a) The stabilizer indicator is in the TO CG position.
- (b) The indicator and the stabilizer surface move in a nose down (trailing edge down) direction.
- (c) The indicator and the stabilizer surface move in a nose up (trailing edge up) direction.
- (d) The stabilizer indicator moves to the TO CG position.

S 985-030
(12) Push the A/P disconnect switch on the control column twice to remove the AUTOPILOT DISENGAGE warning.

S 285-031
(13) Make sure the MCDP shows 65 A/P ENG R TO CMD and push the R CMD button on the mode control panel.

S 285-032
(14) Make sure the MCDP shows 65 IN PROGESS.

S 285-033
(15) Monitor the stabilizer indicator and the stabilizer motion.

S 285-034
(16) Make sure that these movements occur:
  (a) The stabilizer indicator is in the TO CG position.
  (b) The indicator and the stabilizer surface move in a nose down (trailing edge down) position.
  (c) The indicator and the stabilizer surface move in a nose up (trailing edge up) direction.
  (d) The stabilizer indicator moves to the TO CG position.

S 985-035
(17) Push the A/P disconnect switch on the control column twice to remove the AUTOPILOT DISENGAGE warning.

S 285-036
(18) Make sure the MCDP shows 65 A/P ENG C TO CMD and press the C CMD button on the mode control panel.

S 285-037
(19) Make sure the MCDP shows 65 IN PROGRESS.

S 285-038
(20) Monitor the stabilizer indicator and the stabilizer motion.

S 285-039
(21) Make sure that these movements occur:
  (a) The stabilizer indicator is in the TO CG position.
  (b) The indicator and the stabilizer surface move in a nose down (trailing edge down) direction.
  (c) The indicator and the stabilizer surface move in a nose up (trailing edge up) direction.
  (d) The stabilizer indicator moves to the TO CG position.

S 985-040
(22) Push the A/P disconnect switch on the control column twice to remove the AUTOPILOT DISENGAGE warning.

S 285-041
(23) Make sure the MCDP shows TEST COMPLETE.

S 845-042

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(24) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 865-043
(25) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11C6, FLT CONT ELEC 1L AC
(b) 11C8, FLT CONT ELEC 2L AC
(c) 11E17, FLT CONT CMPTR PWR L
(d) 11E35, FLT CONT CMPTR PWR RIGHT

S 985-044
(26) Push the FLT FAULT mode switch and make sure the MCDP shows LAST FLT FAULTS?.

S 285-045
(27) Operate the GRD TEST switch and make sure the GRD TEST mode is engaged.

S 985-046
(28) Use the GRD TEST SEL UP and DOWN switches or the NO/SKIP switch for test selection.

S 285-047
(29) Make sure the display shows 65 STAB TRIM TEST?.

S 985-048
(30) Push the YES/ADV switch to arm the test.

S 285-049
(31) Make sure the MCDP shows 65 FCC L FAIL then push the YES/ADV switch.

S 285-050
(32) Make sure the MCDP shows 65 FCC R FAIL then push the YES/ADV switch.

S 285-051
(33) Make sure the MCDP only shows 65 NO INFC FCC C AUTO TRIM VLD 1 interface fault then push YES/ADV switch.

S 285-053
(34) Make sure the MCDP shows 65 VFY HYD ON - ADV then push the YES/ADV switch.

S 285-055
(35) Make sure the MCDP shows 65 A/P ENG C TO CMD then push the C CMD button on the mode control panel.

S 285-056
(36) Make sure the MCDP shows 65 IN PROGRESS.

S 285-057
(37) Monitor the stabilizer indicator and the stabilizer motion.

S 285-163
(38) Make sure that these movements occur:
- (a) The stabilizer indicator is in the TO CG position.
- (b) The indicator and the stabilizer surface move in a nose down (trailing edge down) direction.
- (c) The indicator and the stabilizer surface move in a nose up (trailing edge up) direction.

(d) The stabilizer indicator moves to the TO CG position.

S 985-058

(39) Push the A/P disconnect switch on the control column twice to remove the AUTOPILOT DISENGAGE warning.

S 285-059

(40) Make sure the MCDP shows TEST COMPLETED.

S 865-060

(41) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C6, FLT CONT ELEC 1L AC
(b) 11C8, FLT CONT ELEC 2L AC
(c) 11E17, FLT CONT CMPTR PWR L
(d) 11E35, FLT CONT CMPTR PWR RIGHT

S 845-061

(42) Do the Restore Airplane to normal procedure if Automatic Stabilizer System test is not required. The procedure is at the end of the system test.

TASK 22-22-00-735-062

3. System Test - Automatic Stabilizer Trim System

A. General
(1) The system test examines the Automatic Stabilizer Trim System interfaces to make sure the system operates correctly.

B. References
(1) AMM 22-10-00/501, Autopilot (Flight Control)
(2) AMM 22-00-02/201, Autoflight BITE
(3) AMM 24-22-00/201, Electrical Power - Control
(4) AMM 27-41-00/501, Horizontal Stabilizer Trim Control System
(5) AMM 27-51-00/201, Trailing Edge Flap System
(6) AMM 27-61-00/201, Spoiler/Speedbrake Control System
(7) AMM 29-11-00/201, Pressurize/Depressurize Main Hydraulic System

(8) AMM 31-41-00/501, Engine Indication and Crew Alerting System
(9) AMM 32-09-02/201, Air/Ground Relays
(10) AMM 34-11-00/201, Pitot - Static System
(11) AMM 34-12-00/501, Air Data Computing System
(12) AMM 34-21-00/501, Inertial Reference System

C. Access

(1) Location Zones

119/120	Main Equipment Center
211/212	Flight Compartment

(2) Access Panel

119BL	Main Equipment Center

D. Prepare For Test

S 715-063

(1) Do the Automatic Stabilizer Trim Operational Test.

S 985-064

(2) Put the stabilizer at approximately four units of trim with control wheel trim switches (either captain's or first officer's switch).

E. Test Stabilizer Trim/Aileron Lockout Module (SAM), Flight Control Computer ARINC 429 Bus Interface, and Stabilizer Trim Control Module (STCM) Interface.

S 285-065

(1) Make sure that the SAM, FCC, and STCM faultballs on both L and R SAMs are black.

F. AIRPLANES WITH STAB TRIM LEVERS ON THE CONTROL STAND; Test Maintenance Interface

S 985-178

(1) Push the ECS/MSG switch on the EICAS maintenance panel to show the ECS/MSG page.

S 285-091

(2) Make sure the lower EICAS display unit does not show the STAB TRIM message.

S 985-092
(3) Put the computer selector switch on the EICAS display select panel in the R position.

S 285-093
(4) Make sure that the lower EICAS display unit does not show the STAB TRIM message.

S 845-094

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(5) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 865-095
(6) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11C6, FLT CONT ELEC 1L AC
(b) 11C8, FLT CONT ELEC 2L AC
(c) 11G17, FLT CONT ELEC 1R AC
(d) 11G26, FLT CONT ELEC 2R AC

S 985-096
(7) Hold the STAB TRIM levers on the control stand, P10, out of the detent.

S 865-097
(8) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C6, FLT CONT ELEC 1L AC
(b) 11C8, FLT CONT ELEC 2L AC
(c) 11G17, FLT CONT ELEC 1R AC
(d) 11G26, FLT CONT ELEC 2R AC

S 285-098
(9) Make sure the STAB TRIM messages show on the lower EICAS display unit after 6 seconds.

S 985-099
(10) Put the computer selector switch on the EICAS display select panel in the L position.

S 285-100
(11) Make sure the STAB TRIM messages show on lower EICAS display unit.

S 985-101
(12) Move the STAB TRIM levers back to the detent position.

G. AIRPLANES WITH ALTERNATE STAB TRIM SWITCHES ON THE CONTROL STAND; Test Maintenance Interface

S 865-182
(1) Push the ECS/MSG switch on the EICAS maintenance panel to show the ECS/MSG page.

S 285-181
(2) Make sure the lower EICAS display unit does not show the STAB TRIM message.

S 865-180
(3) Put the computer selector switch on the EICAS display select panel in the R position.

S 285-179
(4) Make sure that the lower EICAS display unit does not show the STAB TRIM message.

S 865-183
(5) Open this circuit breaker on the P11 panel:
(a) 11E35, FLT CONT CMPT PWR R

S 285-184
(6) Make sure that the STAB TRIM message shows on the lower EICAS display unit after 6 seconds.

S 865-185
(7) Put the computer selector switch on the EICAS display select panel in the L position.

S 285-186
(8) Make sure that the STAB TRIM message shows on the lower EICAS display unit.

S 865-187
(9) Close this circuit breaker on the P11 panel:
(a) 11E35, FLT CONT CMPT PWR R

S 865-215
(10) Open this circuit breaker on the P11 panel:
(a) 11E17, FLT CONT CMPT PWR L

S 285-188
(11) Make sure that the STAB TRIM message shows on the lower EICAS display unit after 6 seconds.

S 865-189
(12) Put the computer selector switch on the EICAS display select panel in the R position.

S 285-190
(13) Make sure that the STAB TRIM message shows on the lower EICAS display unit.

S 865-191
(14) Close this circuit breaker on the P11 panel:
(a) 11E17, FLT CONT CMPT PWR L

S 285-192
(15) Make sure that the STAB TRIM message does not show on the lower EICAS display unit.

H. Test the STAB TRIM Fault Light Interface

S 985-102
(1) Put the stabilizer into approximately four units of trim with the control wheel trim switches.

S 845-103

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(2) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 865-104
(3) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11G17, FLT CONT ELEC 1R AC
(b) 11G26, FLT CONT ELEC 2R AC

S 985-105
(4) Put the C STAB TRIM switch in the CUTOUT position.

S 285-106
(5) Make sure that the yellow STAB TRIM light on the overhead panel, P5, is not lit.

S 285-213
(6) Make sure that the STAB TRIM message on the top EICAS display is not shown.

S 985-107
(7) Put the computer selector switch on the EICAS display select panel in the R position.

S 285-108
(8) Make sure the STAB TRIM message on the top EICAS display is not shown.

S 985-109
(9) Operate the control wheel trim switches in the up direction.

S 285-110
(10) Make sure that the yellow STAB TRIM light comes on and the STAB TRIM message is shown on the top EICAS display.

S 985-111
(11) Put the C STAB TRIM switch in the NORM position.

S 985-112
(12) Operate the control wheel trim switches in the up direction.

S 285-166
(13) Make sure that the yellow STAB TRIM light is out and the STAB TRIM message is not shown on the top EICAS display.

S 865-113
(14) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11G17, FLT CONT ELEC 1R AC
(b) 11G26, FLT CONT ELEC 2R AC

S 845-114

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

(15) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.

S 865-115
(16) Open these circuit breakers on the P11 panel and attach DO-NOT-CLOSE tags:
(a) 11C6, FLT CONT ELEC 1L AC

(b) 11C8, FLT CONT ELEC 2L AC

S 985-116
(17) Set the L STAB TRIM switch to CUTOUT.

S 985-117
(18) Put the computer selector switch on the EICAS display select panel in the L position.

S 985-118
(19) Operate the control wheel trim switches in the down direction.

S 285-167
(20) Make sure that the yellow STAB TRIM light comes on and the STAB TRIM message is shown on the top EICAS display.

S 985-119
(21) Put the L STAB TRIM switch in the NORM position.

S 985-120
(22) Operate the control wheel trim switches in the down direction.

S 285-168
(23) Make sure that the yellow STAB TRIM light is out and the STAB TRIM message is not shown on the top EICAS display.

S 865-121
(24) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11C6, FLT CONT ELEC 1L AC
(b) 11C8, FLT CONT ELEC 2L AC

I. Test Auto Trim Mode Crossfeed Interface

S 285-122
(1) Make sure the Inertial Reference System is initialized and operational (AMM 34-21-00/501).

NOTE: Initialization of IRS requires about 10 minutes.

S 985-123
(2) Put the stabilizer in approximately seven units of trim with the control wheel trim switches.

S 985-124
(3) Turn the flight director on.

S 985-125
(4) Push the V/S switchlight on the mode control panel.

S 985-126
(5) Turn the vertical speed knob until the digital display shows -2000 ft/min. then push the L CMD switch/light on the mode control panel.

S 285-127
(6) Make sure that stabilizer movement is shown on the stabilizer trim indicator.

S 285-169
(7) Make sure the yellow UNSCHED STAB TRIM light is not on and the UNSCHD STAB TRIM message is not shown on the top EICAS display.

S 985-128
(8) Push the autopilot disconnect switch to disengage the autopilot.

S 985-170
(9) Push it a second time to cancel the aural and visual alerts.

S 985-129
(10) Put the selector switch on the EICAS display select panel in the R position.

S 985-130
(11) Put the stabilizer in approximately two units of trim with the control wheel trim switches.

S 985-131
(12) Turn the vertical speed knob until the digital display on the mode control panel shows +2000 ft/min then press the R CMD switch/light.

S 285-132
(13) Make sure that stabilizer movement is shown on the stabilizer trim indicator.

S 285-171
(14) Make sure the yellow UNSCHD STAB TRIM light is not on and the UNSCHD STAB TRIM message is not shown on the top EICAS display.

S 985-133
(15) Push the autopilot wheel disconnect switch to disengage the autopilot.

S 985-172
(16) Push it a second time to cancel the aural and visual alerts.

S 985-134
(17) Turn the vertical speed knob until the digital display shows 0000.

J. Test Unscheduled Trim Interface

S 985-135
(1) Put the stabilizer in approximately seven units of trim with the control wheel trim switches.

S 285-193
(2) AIRPLANES WITH STAB TRIM LEVERS ON THE CONTROL STAND;
Make sure that the STAB TRIM levers on the control stand (P10) are in neutral.

S 285-137
(3) Make sure that the yellow UNSCHED STAB TRIM light is off.

S 985-138
(4) Push the L CMD switchlight on the mode control panel.

S 985-194
(5) AIRPLANES WITH STAB TRIM LEVERS ON THE CONTROL STAND;
Move the STAB TRIM lever forward.

S 985-195
(6) AIRPLANES WITH ALTERNATE STAB TRIM SWITCHES ON THE CONTROL STAND;
Set and hold the alternate stab trim switches in the APL NOSE DN position.

S 285-140
(7) Make sure that the yellow UNSCHD STAB TRIM light is on and the UNSCHD STAB TRIM message is shown on the top EICAS display.

S 985-197
(8) AIRPLANES WITH STAB TRIM LEVERS ON THE CONTROL STAND;
Set the STAB TRIM levers to neutral.

S 985-198
(9) AIRPLANES WITH ALTERNATE STAB TRIM SWITCHES ON THE CONTROL STAND;
Set the alternate stab trim switches to neutral.

S 285-142
(10) Make sure that the UNSCHED STAB TRIM light is out and the message is not shown.

S 985-143
(11) Push the autopilot wheel disconnect switch twice to disengage autopilot and cancel autopilot alerts.

S 985-144
(12) Put the computer selector switch on the EICAS display select panel in the L position.

S 985-145
(13) Push the R CMD switchlight on the mode control panel.

S 985-200
(14) AIRPLANES WITH STAB TRIM LEVERS ON THE CONTROL STAND;
Move the STAB TRIM levers aft.

S 985-201
(15) AIRPLANES WITH ALTERNATE STAB TRIM SWITCHES ON THE CONTROL STAND;
Set the alternate stab trim switches in the APL NOSE UP position and hold.

S 285-147
(16) Make sure that the yellow UNSCHD STAB TRIM light is on and the UNSCHD STAB TRIM message is shown on the top EICAS display.

S 985-203
(17) AIRPLANES WITH STAB TRIM LEVERS ON THE CONTROL STAND;
Set the STAB TRIM levers to neutral.

S 985-204
(18) AIRPLANES WITH ALTERNATE STAB TRIM SWITCHES ON THE CONTROL STAND;
Set the alternate stab trim switches to neutral.

S 285-149
(19) Make sure that the UNSCHED STAB TRIM light is out and the message is not shown.

S 985-150
(20) Push the autopilot wheel disconnect switch twice.

K. Test Standby Trim Switch Interface

S 985-206
(1) AIRPLANES WITH STAB TRIM LEVERS ON THE CONTROL STAND;
Move the STAB TRIM levers forward.

S 985-207
(2) AIRPLANES WITH ALTERNATE STAB TRIM SWITCHES ON THE CONTROL STAND;
Set the alternate stab trim switches in the APL NOSE DN position and hold.

S 285-152
(3) Make sure that the yellow UNSCHED STAB TRIM light stays out.

S 985-209
(4) AIRPLANES WITH STAB TRIM LEVERS ON THE CONTROL STAND;
Set the STAB TRIM levers to neutral.

S 985-210
(5) AIRPLANES WITH ALTERNATE STAB TRIM SWITCHES ON THE CONTROL STAND;
Set the alternate stab trim switches to neutral.

L. Restore Airplane to Normal

S 985-154
(1) Push the RESET button on the left and the right SAM.

S 285-155
(2) Make sure all faultballs on the two SAMs show black.

S 985-156
(3) Push the ECS/MSG switch on EICAS maintenance panel, P61.

S 985-158
(4) Push the EVENT READ - AUTO switch on the EICAS maintenance panel, P61.

S 985-159
(5) Push and hold the ERASE switch on the EICAS maintenance panel, P61, for 3 seconds then release.

S 865-160
(6) Remove pressure from the left, center and right hydraulic systems (AMM 29-11-00/201).

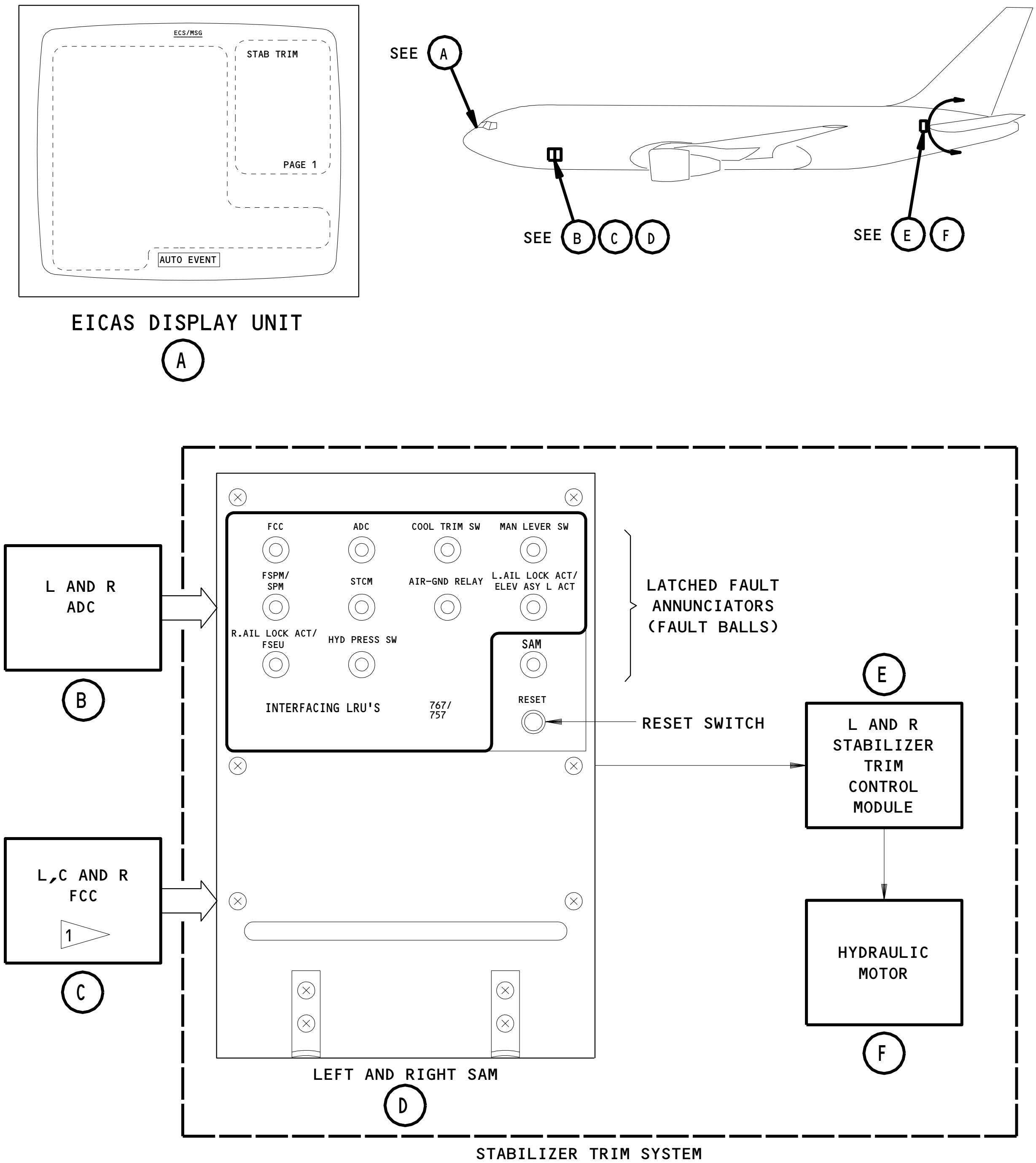
S 985-220
(7) Stop the MCDP if it is running.

S 865-219
(8) Disconnect the remote MCDP control panel from connector D1447 at the main distribution panel, P6, if it is connected.

S 865-161
(9) Remove electrical power if it is not necessary (AMM 24-22-00/201).

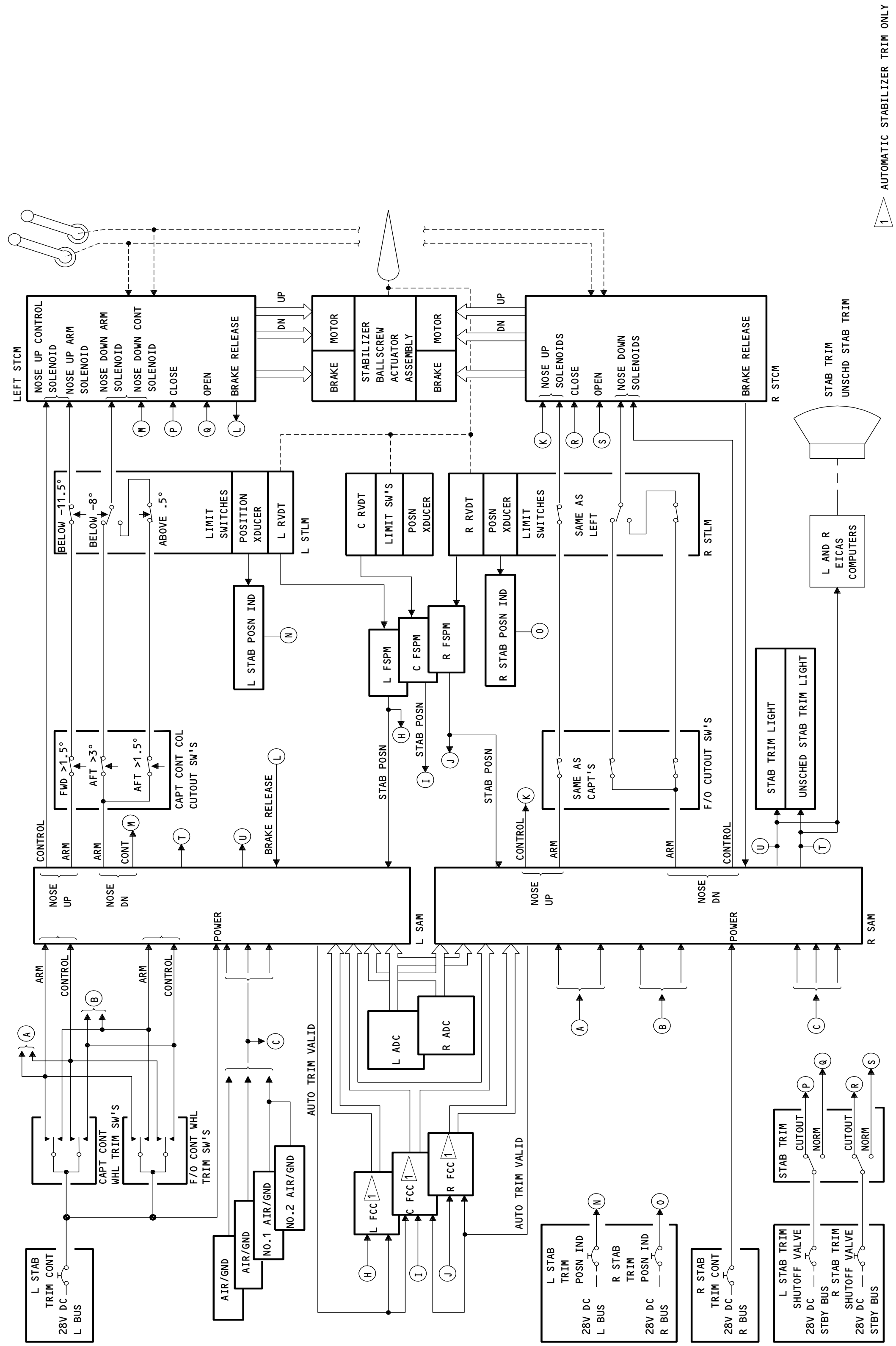
MACH TRIM SYSTEM - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The Mach trim mode of the stabilizer trim system provides incremental horizontal stabilizer position in response to changes of Mach. The mode is only engaged when airborne (after a 20 second delay) and the manual and autotrim functions are not engaged.
   B. This chapter contains Mach trim stability operation. Refer to 22-22-00/001 for a general description of the Automatic Stabilizer Trim System. Refer to 27-41-00/001 for the component details and system operation of the Horizontal Stabilizer Trim Control System.
2. Operation
   A. Functional Description
      (1) Stabilizer Trim System Block Diagram (Fig. 2)
         (a) The Stabilizer Trim Aileron Lockout Module (SAM) selects the Mach function as the active trim command source when:
            1) The flight control computer (autotrim) function is not used.
            2) The control wheel mounted manual electric trim switches are not activated.
         (b) When a Mach function is the trim command source, only one trim channel (left or right) is operated.
         (c) The controlling SAM is randomly selected at power-up. Priority for simultaneous engagement is given to right SAM. This provides half-rate trim (0.05 to 0.25 degrees/sec). During Mach operation, the stabilizer position is related Mach. The SAM determines the desired stabilizer position command that operates the Stabilizer Trim Control Module (STCM) and ball-screw actuator. The SAM uses the following inputs to determine ARM and CONTROL output commands to the STCM:
            1) Digital Mach data from the Air Data Computers (ADCs).
            2) Flap position (up or down) from the Flap/Stabilizer Position Modules (FSPMs).
            3) Stabilizer position from the Rotary Variable Differential Transformer Transducers (RVDTs) via the FSPMs.
         (d) If the Mach function is operational and the flaps are retracted, Mach trim control law is in control.
      (2) Mach Trim Control Law Schedule (Fig. 3)



1 AUTOMATIC STABILIZER TRIM SYSTEM ONLY

Mach Trim System
Figure 1



Stabilizer Trim System Block Diagram
Figure 2 (Sheet 1)

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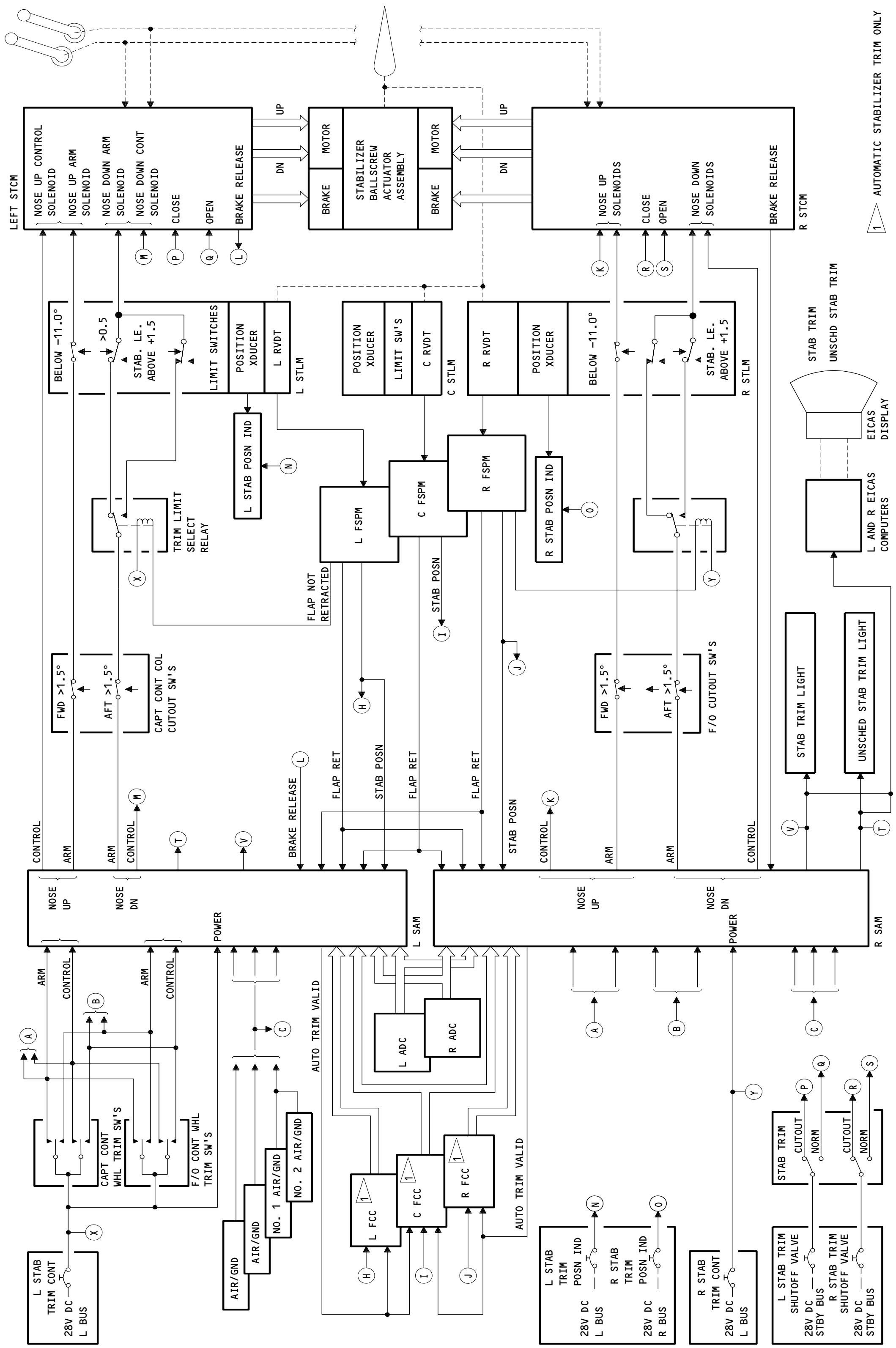


Stabilizer Trim System Block Diagram
Figure 2 (Sheet 2)

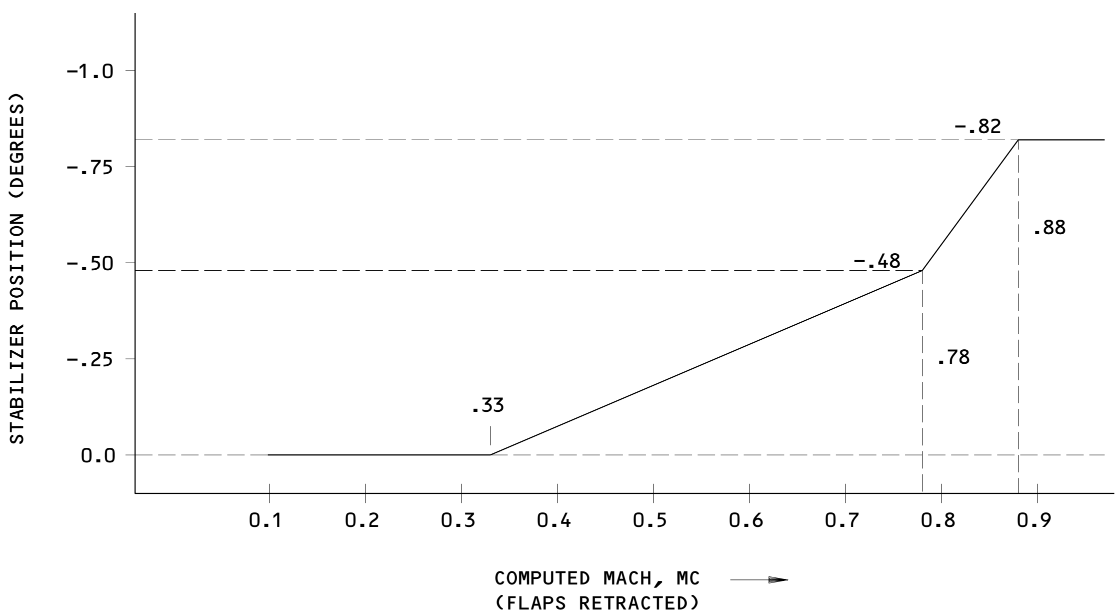
(a) Mach Trim Control Law

1) When the Mach trim mode engages, the Mach trim control law synchronizes to stabilizer position. When the flaps are retracted, the control law provides incremental stabilizer position changes dependent upon computed Mach (Mc). The allowable trim threshold is less than or equal to 0.01 Mach. After trimming, the final stabilizer position change produced is equal to the position change corresponding to the ideal characteristic in Fig. 3 within +0, -0.10 degrees for increase in Mach number and within -0, +0.10 degrees for decrease in Mach number. The ADCs provide digital Mc inputs. Half-speed trim (0.05 to 0.25 degrees/sec) commands are provided by the selected SAM. The Mach trim mode is disengaged on-ground and until 20 seconds into the in-air mode.

(3) Mach Trim Schematic (Fig. 4)

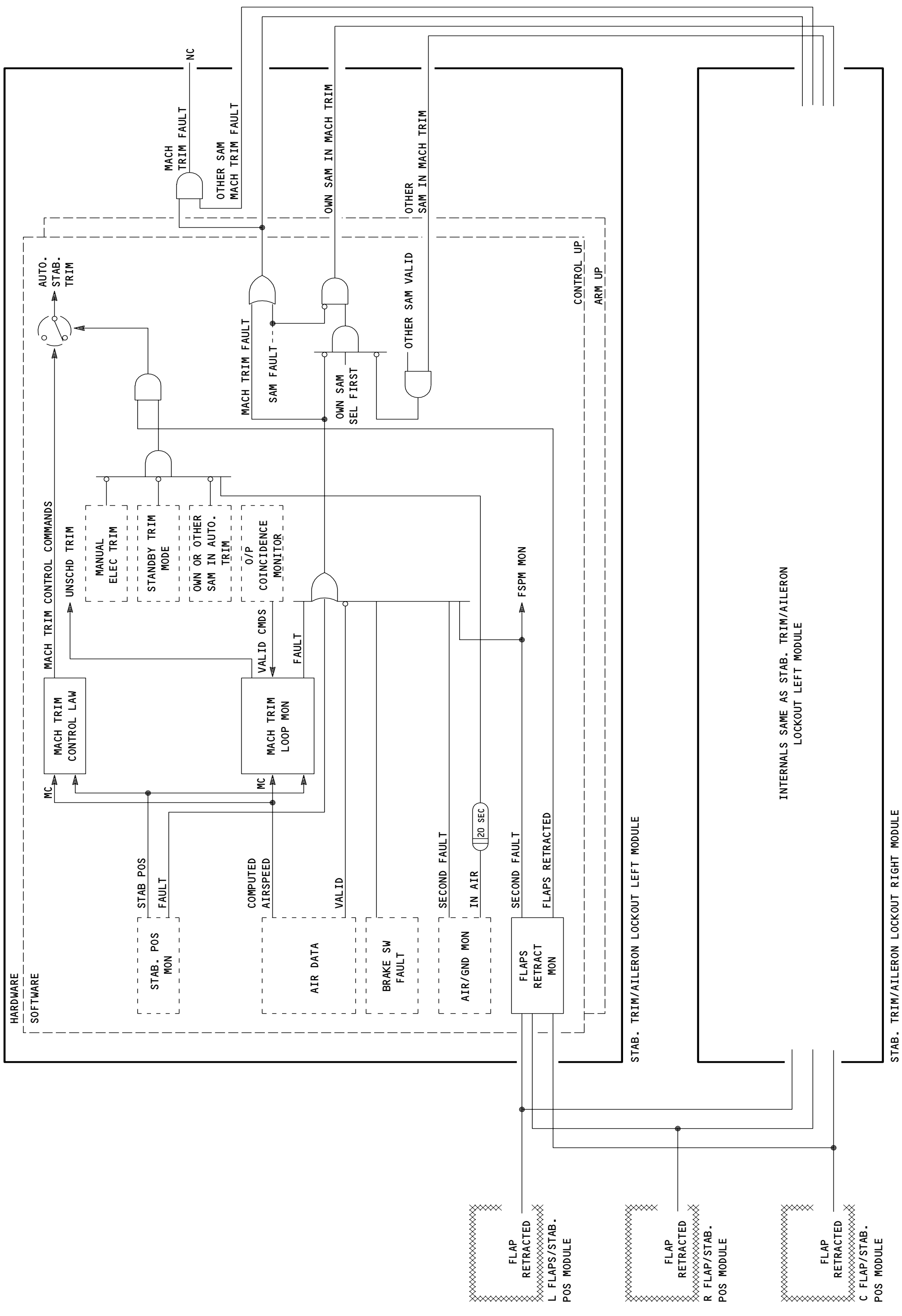
(a) The Stabilizer Trim Aileron Lockout Module (SAM) uses the following inputs to determine Mach trim commands.

1) The Air Data Computers (ADCs) provide digital Mach inputs to both SAMs (Ref 34-12-00/001, Air Data Computing System).



MACH TRIM CONTROL LAW

Mach Trim Control Law Schedule
Figure 3



Mach Trim Schematic
Figure 4

2) The three Flap/Stabilizer Position Modules (FSPMs) each input a flap retracted signal to both SAMs. The left and right FSPMs input stabilizer position to the corresponding SAM (Ref 27-58-00, TE Flap Position Indicating System).
3) The air/ground relays provide airplane air/ground status. Each SAM receives three air/ground inputs from four relays (Ref 32-09-00, Landing Gear Multiple Use System/Components).

(b) Mach Trim SAM Operation
1) Each SAM contains ARM and CONTROL microprocessors to form a dual system. Each processor receives digital Mach inputs from the left and right ADCs. Continuous monitoring is provided for both primary and secondary ADC inputs. The following input checks are provided:
    a) If a parity error is detected, the value is discarded and the previous valid value is used. Four or more parity errors within eight successive samples set an ADC fault.
    b) Data which is identified as FAILED by the ARINC 429 status matrix is discarded and the previous valid value is used. Four or more FAILED values of a parameter within eight successive samples sets an ADC fault.
    c) The Mc data identified as No Computed Data (NCD) by the status matrix is discarded. The value is replaced by 0.1 Mach.
    d) If Mc is not received within 150 milliseconds for 2 consecutive frames, when required by the SAM, an ADC fault is set. Old data is used when new data is not received for one frame.
2) Cross comparison between primary and secondary Mc is provided. An ADC comparison fault is set if:
    a) Primary and secondary Mc differ by more than 0.05 Mach for more than five seconds.
3) The cross comparison is inhibited if either ADC is invalid or in the TEST mode or the airplane is on the ground. An ADC fault is cleared if valid data is continuously received for 30 seconds and no more than four resets have occurred.
4) The flaps retract and air/ground monitor are majority vote (two out of three) monitors. Flaps-retract failures set the FSPM fault ball. Air/ground failures set the AIR/GND fault ball.

5) The stabilizer position is monitored and input to synchronize the control laws. An input level monitor checks for signal inputs less than +1.0 vdc or an open circuit. A detected failure sets the FSPM fault ball. A stabilizer fault is cleared in the air if a valid input is received for five seconds and no more than four resets have occurred.
6) Mach commands are generated in the control law. The command outputs depend on the following conditions:
   a) Flaps retracted for Mach command outputs.
   b) Neither SAM in autotrim or manual trim modes (Ref 27-41-00/001, Horizontal Stabilizer Trim Control System).
7) The unscheduled trim monitor is active when the Mach trim is selected. Unscheduled stabilizer motion is detected by the corresponding SAM when the stabilizer moves more than (0.55 degree in either direction without a valid trim command. An UNSCHD TRIM discrete is output from the controlling SAM.
8) The UNSCHD TRIM discrete turns on the UNSCHED STAB TRIM light on annunciator panel, and cause the message UNSCHD STAB TRIM to be generated on upper EICAS display unit.
9) The internal monitors in each SAM determine Mach loop faults. An unscheduled trim fault shall inhibit Mach loop faults from being set. The monitors detect the following conditions which set the Mach fault indications.
   a) The stabilizer does not move in the correct direction within 10 seconds of a valid SAM command. (The loop fault latch resets after 5 seconds of no fault conditions.)
   b) Input stabilizer signal level is outside preset limits.
   c) The Mach control loop error exceeds 0.3 degree in the CONTROL channel or 0.5 degree in the ARM channel for longer than 10 seconds with a valid SAM trim command. (The loop fault latch shall be reset after 5 seconds of no fault conditions).
   d) Release of a hydraulic brake without a valid SAM output trim command.
   e) A hydraulic brake not released within 2 seconds of a valid SAM output trim command.
   f) Faults within ADCs or wiring to SAMs that cause air data to be monitored as invalid.
   g) Following a second fault in either flaps-retracted or air/ground discrete signals majority vote.

10) Switching to the other SAM occurs when MACH fault or SAM fault is detected within controlling SAM. If a MACH fault or SAM fault is detected within both SAMs, the message STAB TRIM is displayed as a maintenance level message on the EICAS display unit to indicate complete loss of the SAM function (SAM faultball set).

MACH TRIM/SPEED STABILITY - DESCRIPTION AND OPERATION

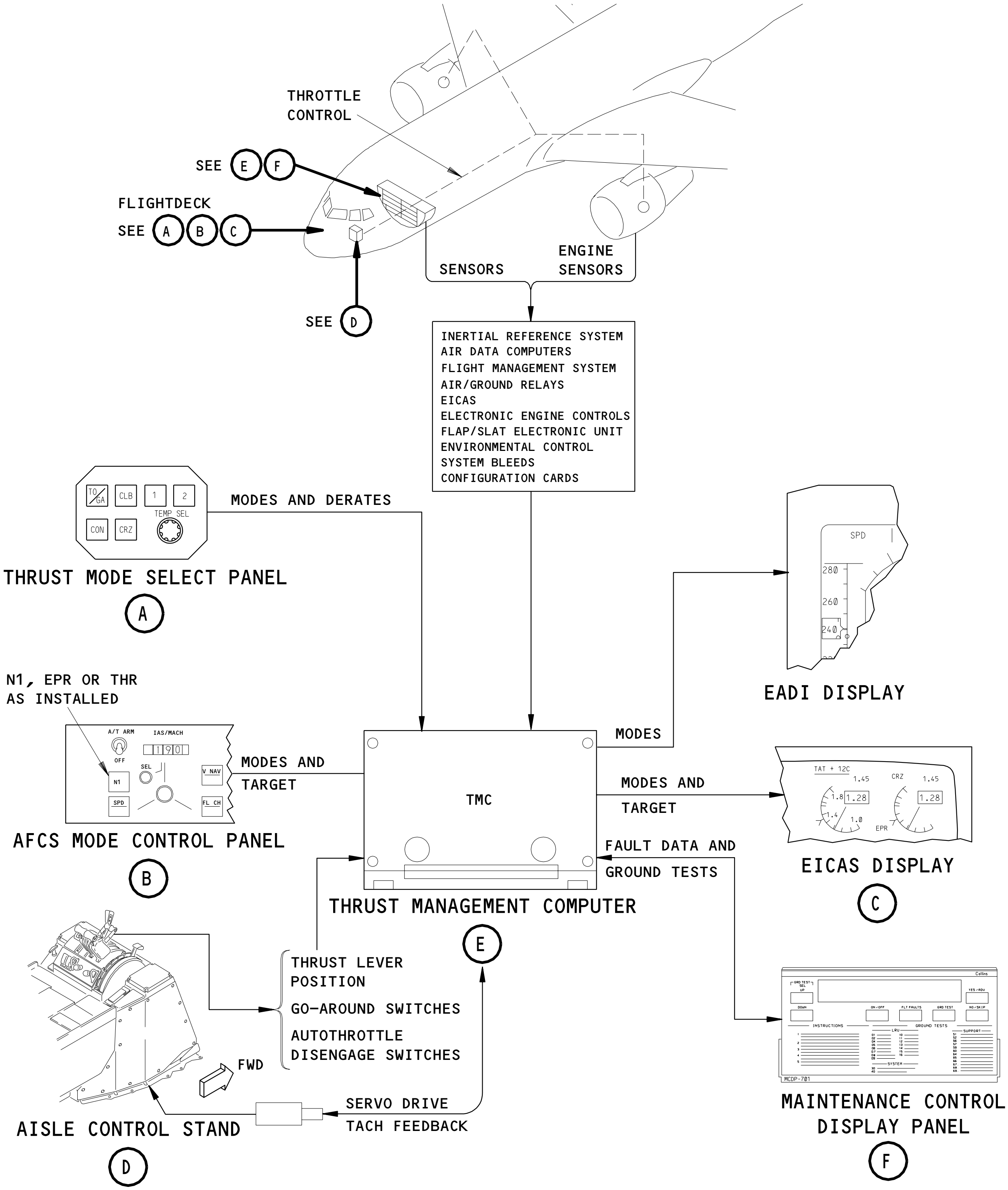
MACH TRIM SYSTEM - ADJUSTMENT/TEST

TASK 22-24-00-705-001

1. Mach Trim System Test
   A. General
      (1) The Mach Trim System is part of the SAM system. To do a test on the Mach Trim system, refer to AMM 22-22-00/501.

THRUST MANAGEMENT POWER - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. Thrust Management System (TMS) Overview (Fig. 1)
      (1) The Thrust Management System (TMS) is part of the autoflight control system (AFCS). The TMS controls engine thrust in response to flight mode requests and thrust limit selection. The TMS controls engine thrust through the full flight regime of operation. A comprehensive TMS description and operation is located in AMM 22-32-00/001.
      (2) TMS thrust limit calculations and autothrottle functions are performed by the thrust management computer (TMC). The TMC interfaces with the engines through the A/T servomotor generator, which drives the thrust levers through the A/T clutch/brake pack assemblies.
      (3) The operator interface to the TMS is through the thrust mode select panel (TMSP) and the AFCS mode control panel (MCP). The TMSP allow manual selection of thrust limit mode, fixed derate and assumed temperature. Autothrottle flight mode and speed target selection is made through the MCP.
      (4) TMS information show on the EADI and the EICAS display. The autothrottle flight mode and status data show on the EADI display. The EICAS display show the selected thrust limit mode or fixed derate, corresponding limits and assumed temperature.
      (5) Autothrottle disconnect causes visual and aural indications. TMS flight fault information is sent to the Maintenance Control Display Panel (MCDP) for post-flight analysis.
      (6) The TMC uses single phase, 400 Hz, 115v ac and 28v dc power from the left main bus. Separate 28v dc power from the standby bus is used for failure monitoring. The TMC converts the 115v ac to various ac and dc voltage levels for internal and external uses. The A/T servomotor generator uses 115v ac power.
   B. TMS Components
      (1) Thrust Management Computer (TMC)
         (a) One TMC is installed in main electrical-electronic equipment compartment. It performs these functions:
            1) Thrust limit calculations



Thrust Management System
Figure 1

2) Autothrottle control
3) Engine trim
4) BITE/Fault monitoring
5) TMS annunciation

(b) Thrust limits are continuously calculated by the TMC. Maximum and reference thrust limits are computed for these modes:
1) Takeoff (TO) thrust
2) Go-around (GA)
3) Climb (CLB)
4) Continuous (CON)
5) Cruise (CRZ)
6) Temperature derated TO
7) Fixed derated TO and CLB modes

(c) Thrust limits are functions of the following inputs:
1) Temperature
2) Barometric altitude
3) Barometric pressure
4) Engine/airframe data bases
5) Thrust model data base
6) Flight deck inputs
7) Air/ground state

(d) Thrust limit protection is active in all modes to prevent exceeding the limits for overboost, overspeed, or minimum speed.

(e) The autothrottle function sets the thrust levers to capture and hold a thrust setting or to hold a target airspeed. A/T function calculations use these TMC sub-functions:
1) Engage logic
2) Control Laws and Mode logic

(f) The TMC sends A/T drive commands to the A/T servomotor which moves the thrust levers.
1) Mode logic allows manual or automatic selection of the available autothrottle mode, thrust limit mode, and thrust limits.
2) Control laws calculate the A/T drive command. The A/T drive command is derived from the difference between the desired and the actual airplane condition.
3) Engage logic determines the validity of the control law data and permits the software control laws to operate. The engage logic also sends logic that connects excitation voltage to the A/T servomotor. This permits the servomotor to operate when it receives an A/T drive command.

(2) Thrust Mode Select Panel (TMSP)
 (a) One TMSP is located in F/O instrument panel (P3-1). The TMSP lets the crew select thrust limit computation modes. A thrust limit mode and corresponding derate is selected for each flight phase.
(3) Autothrottle Servomotor Generator
 (a) The autothrottle (A/T) servomotor generator is one component of the A/T assembly and is located under the thrust levers in flight deck control stand (P10).
 (b) The A/T servomotor generator provides the electrical interface between the TMC and the thrust levers. The A/T servomotor drives the thrust levers through the gearbox and clutch/brake pack assemblies. The generator provides a tachometer feedback signal to control the thrust lever movement rate.

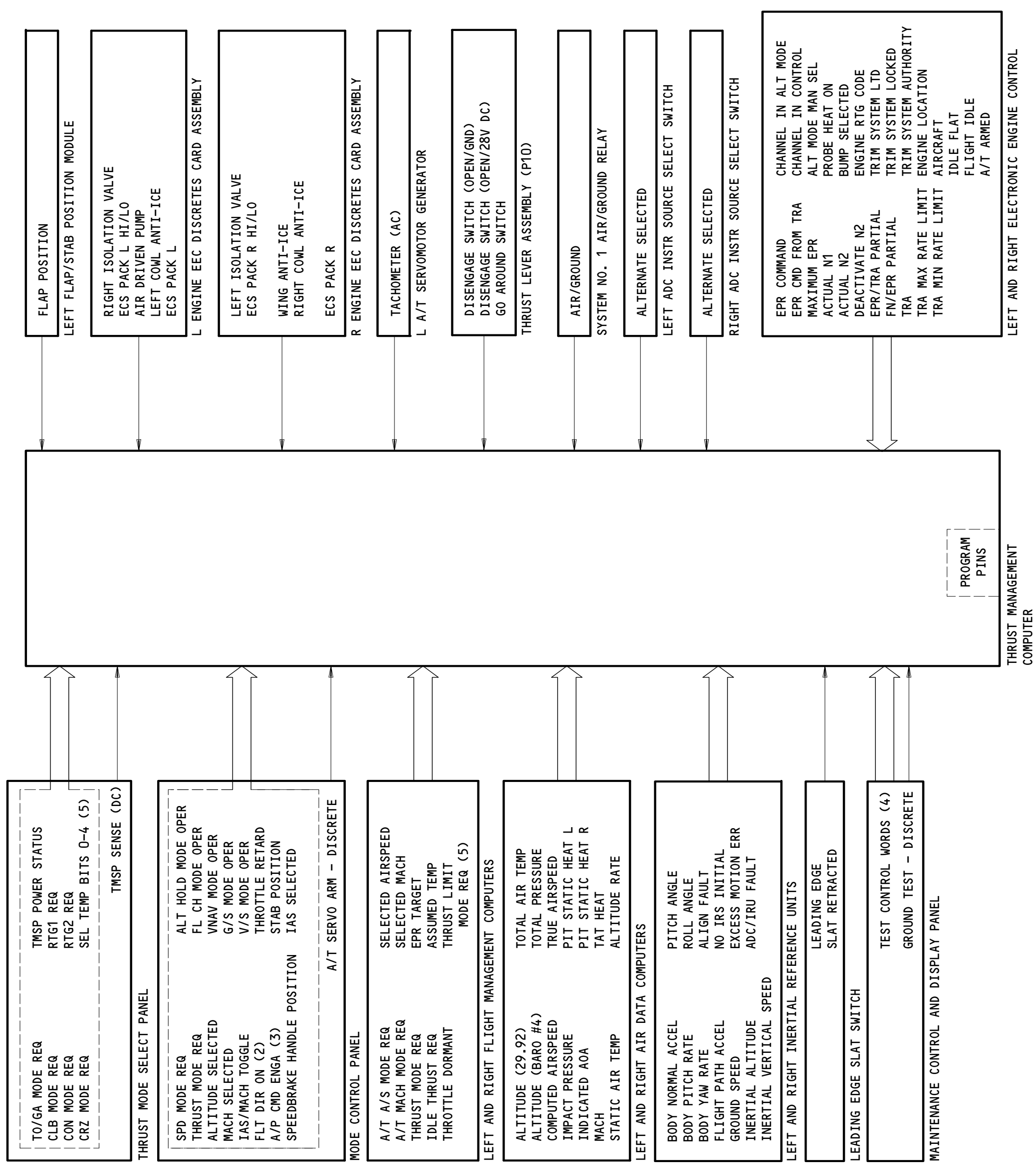
C. TMS Control Inputs
(1) AFCS Mode Control Panel (MCP)
 (a) MCP - A/T mode select switches
  1) The A/T mode select switches give manual selection for these A/T control modes: Thrust mode (EPR/N1/THR), Speed mode (SPD), Flight Level Change mode (FL CH) and Vertical Navigation mode (VNAV). A/T mode engagement occurs after arming. The switch dot-matrix display comes on when the mode request is accepted and operating.
 (b) MCP - A/T Arm switch
  1) The A/T arm switch is a two position toggle switch, located on the left side of the MCP, that supplies +28v dc power when positioned in the ARM position. The switch enables selected A/T control mode, energizes the engine enable trim relay and arms the A/T servomotor excitation relay.
 (c) MCP - IAS/MACH control knob
  1) The IAS/MACH is a rotary/push switch used to set the reference airspeed/Mach and transfers speed control between the A/T and FMS.
(2) Thrust Mode Select Panel (TMSP)
 (a) The TMSP has four thrust limit mode select switches that select Take off (TO), Go-around (GA), Climb (CLB), Continuous (CON), and Cruise (CRZ) thrust limit modes.

(b) The TMSP has two fixed derate select switches that give two optional fixed thrust limit derate values.
(c) The TMSP has an assumed temperature select knob that gives variable thrust limit derate values.
(3) Flight Management System (FMS)
(a) The FMS controls the A/T flight modes and corresponding target reference values when the A/T VNAV mode is engaged.
(b) The assumed temperature derate selection is made through the FMC CDUs.

D. Sensor Inputs
(1) The Air Data Computing System (ADCS) (AMM 34-12-00/001) consists of left and right Air Data Computers (ADCs), which provide environmental data to the TMC.
(2) The Inertial Reference System (IRS) (AMM 34-21-00/001) interface consists of left and right IRUs, which provide pitch and roll attitude; and longitudinal and vertical acceleration.
(3) The left and right electronic engine control (EEC) (AMM 73-21-00/001) interfaces give engine parameter data, thrust command data and thrust lever position data.

E. TMS Outputs
(1) Autothrottle thrust modes, limits and temperature derates shown on the EICAS display.
(2) Autothrottle flight modes, limits and status shown on the EADI display.
(3) Fault data output to the MCDP.
(4) A/T Servomotor drive command.

F. System Interface Inputs (Fig. 2)
(1) Digital Data Bus Inputs
(a) Thrust Mode Select Panel (TMSP)
(b) AFCS Mode Control Panel (MCP)
(c) Maintenance Control Display Panel (MCDP)
(d) Left and Right Air Data Computers (ADCs)
(e) Left and Right Flight Management Computers (FMCs)
(f) Left and Right Inertial Reference Units (IRUs)
(g) Left and Right Electronic Engine Controls (EECs)



TMC System Interface Inputs
Figure 2

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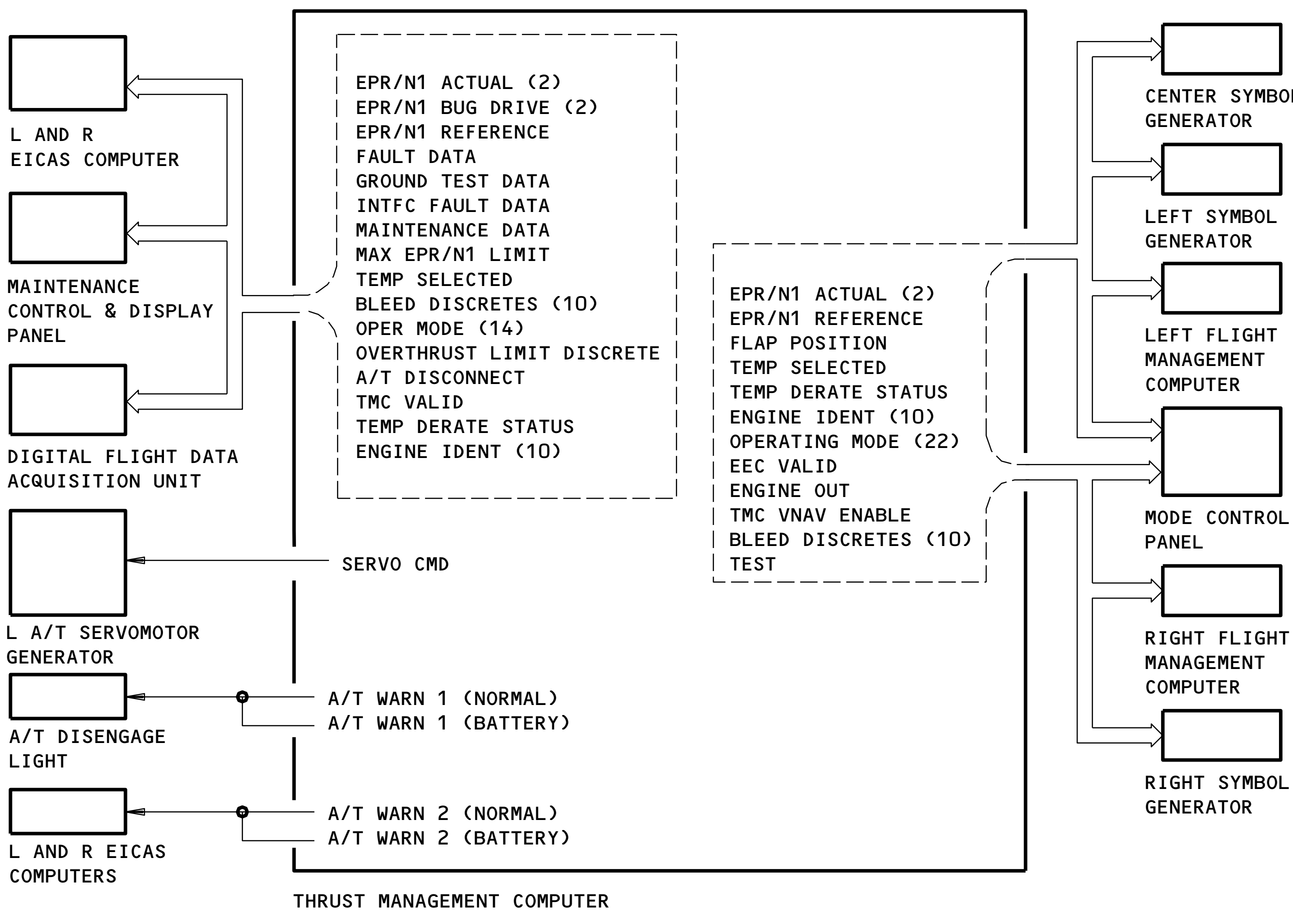
(2) Analog Inputs
(a) TMSP sense voltage for the TMC voltage regulator
(b) Flap position from the flap/stab position module
(c) Tachometer feedback from the servomotor generator
(3) Analog Discrete Inputs
(a) A/T servo ARM from the MCP
(b) In-air from air/ground relay
(c) Ground test from the MCDP
(d) Bleed discretes from the left and right EEC discrete card assemblies.
(e) A/T disconnect/reset from the A/T disconnect switches.
(f) Go-around command from the Go-around (GA) switches.
(g) Slat extended from the leading edge slat switch.

G. TMC Program Pin Data
(1) The TMC is designed to accomodate various airplane, engine, and customer selectable operational options. This is done by loading the computer with appropriate software and connecting designated external program pins. For specific pin connections on each airplane, refer to FIM 22-00-04/101, Figure 104. The hardwired pins establish the binary codes for software control.
(2) Engine Airframe and Thrust Limit Options
(a) Pin connections for the engine airframe options are open/ground. Thrust limit options are 28 Vdc/ground. A parity pin is provided.
(3) Autothrottle and Aircraft Options
(a) Pin connections for autothrottle options are 28 Vdc/ground. Aircraft options are open/ground. A parity pin is provided.
(4) Customer Option
(a) Pin connections are open/ground for customer options. There is no parity pin. If a ground is lost, that option is also lost.

H. TMC/System Interface Outputs (Fig. 4)

I. The TMC has five digital bus outputs.
(1) Digital data bus number one outputs:
(a) Thrust Mode Select Panel (TMSP)
(b) Left and Right Electronic Engine Controls (EECs)

Not Used
Figure 3



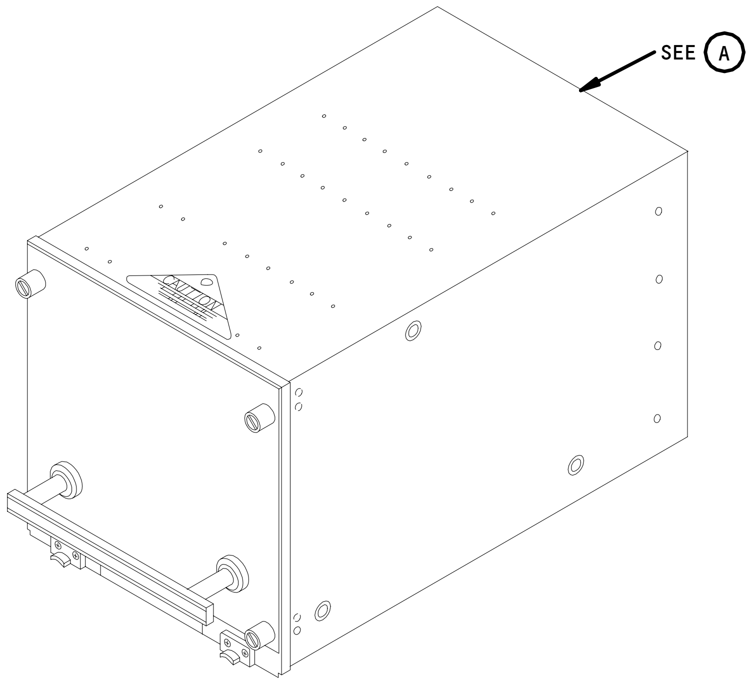
TMC Systems Interface Outputs
Figure 4

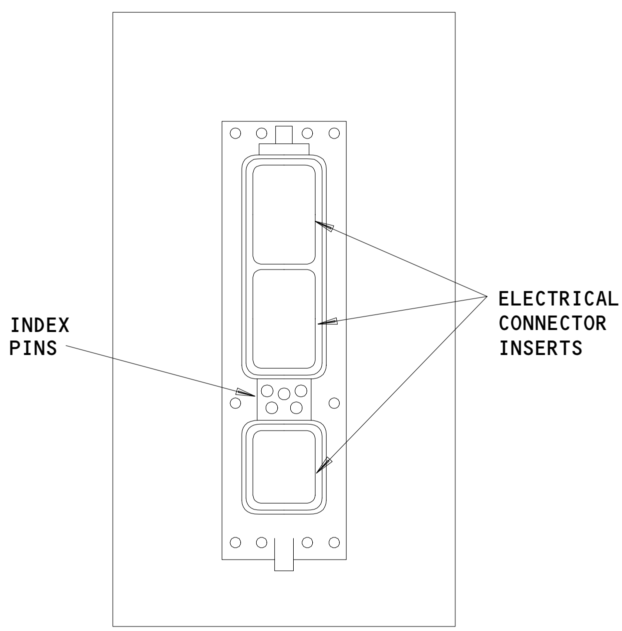
(2) Digital data bus number two outputs:
(a) AFCS Mode Control Panel (MCP)
(b) Left and Center EFIS Symbol Generators
(c) Left Flight Management Computer (FMCs)
(3) Digital data bus number three outputs:
(a) AFCS Mode Control Panel (MCP)
(b) Right EFIS Symbol Generators
(c) Right Flight Management Computer (FMCs)
(4) Digital data bus number four outputs:
(a) Maintenance Control Display Panel (MCDP)
(b) Left and Right EICAS computers
(5) Digital data bus number five outputs:
(a) Digital data bus test point.
(6) Analog Outputs
(a) There are four analog servomotor generator outputs:
1) Servomotor command L - FWD and REV
2) Servomotor command R - FWD and REV (not used)
(7) Analog Discrete Outputs
(a) A/T disconnect light discretes:
1) A/T WARN 1 (NORMAL)
2) A/T WARN 1 (BATTERY)
(b) EICAS "A/T DISC" message discretes:
1) A/T WARN 2 (NORMAL)
2) A/T WARN 2 (BATTERY)

J. Configuration
(1) SAS 050-051, 150-157, 162-167, 275-278
280-281 PRE SB 22A-97;
The -205 TMC is installed.
(2) SAS 050-051, 150-157, 162-164, 166, 275-278
280-281 POST SB 22A-97;
SAS 165, 167 POST 22A-97 AND PRE SB 71-0118;
The -1205 TMC is installed.
(3) SAS 165, 167 POST SB 71-0118;
The -1206 TMC is installed.

2. Component Details

A. Thrust Management Computer (Fig. 5)
(1) The TMC consists of 8 modules in a 6 MCU case. The weight is 18 pounds. The TMC uses 115 Vac and 28 Vdc supplied through circuit breakers on the P11 panel.





THRUST MANAGEMENT COMPUTER CONNECTIONS
A

Thrust Management Computer
Figure 5

(2) The TMC is a realtime, digital processor (MCP 701A). Functions are performed by modules which communicate by digital bus. A description of each module is given below:

(a) The CPU/Timing module includes the central processing unit, interrupt priority logic, real-time reference and watchdog monitor, memory parity generation and checking.

(b) The memory module contains program storage and variable memory which are functionally independent. A growth location memory module is also provided.

(c) The ARINC 429 DITS module interfaces all TMC digital inputs and outputs to the digital processor. The ARINC DITS expansion module contains the overflow and growth ARINC receivers.

(d) The analog I/O module interfaces TMS external AC inputs and outputs and TMC internal DC inputs and outputs to the digital processor.

(e) The System Interface Module (SIM) performs two separate operations. It receives and transmits airplane system information in the form of analog discretes. It also provides non-volatile memory for registering the results of specific internal self tests that generate a TMC INVALID digital discrete.

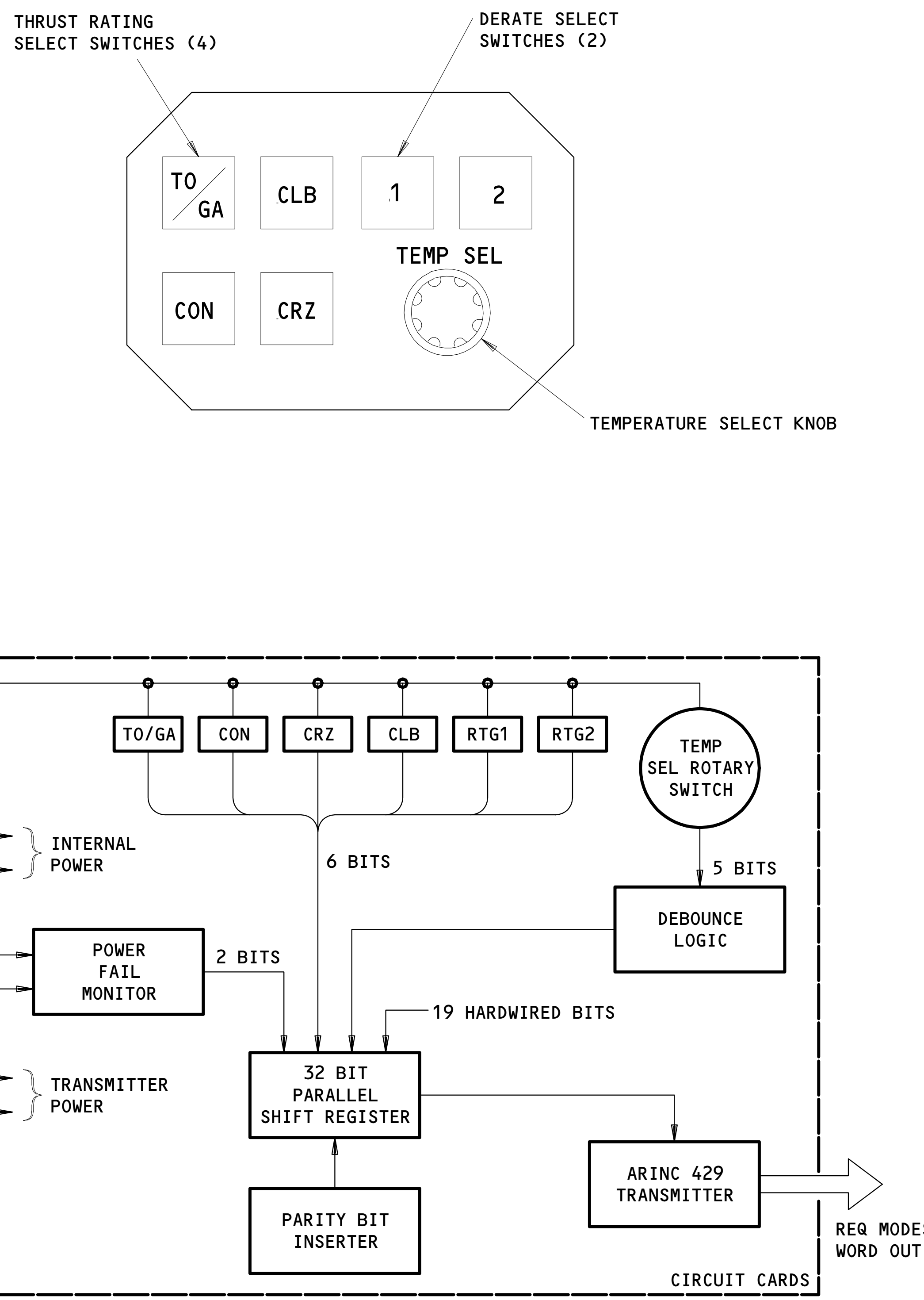
(f) The power supply module provides all power conversion for the TMC and TMSP circuits.

(3) Connections

(a) Three connector inserts are on the back side of the TMC. One insert contains TMC power pins only. The other two inserts contain the TMC signal pins and low voltage power pins. Index pins ensure correct TMC installation.

B. Thrust Mode Select Panel (TMSP) (Fig. 6)

(1) The TMSP consists of six pushbutton switches, one rotary switch, and two circuit cards mounted in a metal case. Panel switch illumination is provided by the edge-lit lightplate panel. The lightplate is powered by variable 0 to 5 Vac provided by the airplane master dim and test circuits. A connector at the rear of the unit mates the TMSP with airplane wiring.



THRUST MODE SELECT PANEL OPERATION

Thrust Mode Select Panel
Figure 6

(2) Select Switches

(a) Manual thrust limit mode selection is accomplished by pressing the TMSP pushbutton for the desired mode. The switches ground the appropriate mode discrete bit line when actuated. This informs the TMC that a particular mode has been selected. Pressing a pushbutton corresponding to the currently selected mode has no affect.

(b) The TMSP provides two thrust derating methods for reducing the thrust limit. Fixed derating is selected by pressing TMSP derate pushbutton 1 or 2. Assumed temperature derating is selected by using the TEMP SEL (temperature select) knob. The TEMP SEL knob controls a binary-coded 22-position continuous rotation switch. The five bits that represent the position of the switch are transmitted to the TMC as part of the REQ MODES word. The TMC uses the switch position information (assumed temperature) to compute a derated value of thrust limit. Rotation of the knob by one detent position causes the flat rated temperature to be displayed on the EICAS display unit. Continued clockwise rotation changes the display in 1°C increments of the selected temperature. An increase in assumed temperature derates the REF Thrust Limit value.

(3) Module Description

(a) The circuit components of the TMSP are on two cards. The ARINC transmitter circuit card assembly contains the ARINC 429 serial data communication circuits. The clock generator circuit card assembly generates the timing signals required by the data communications circuits.

(b) The TMSP transmitter circuits inform the TMC of mode, derating factor, and assumed temperature selection. Data is transmitted as a single 32-bit digital word called the REQ MODES (request modes) word. The REQ MODES word informs the TMC of TMSP power status, mode switch selection, and TMSP SEL switch position. The REQ MODES word is transmitted to the TMC every 16 milliseconds.

(c) The clock generator circuit card consists of a crystal-controlled oscillator that generates a 2 MHz signal. The 2 MHz is divided to form 16 millisecond clock pulses used by the TMSP. The 16 millisecond clock triggers the sequence of events required to transmit the 32-bit REQ MODES word.

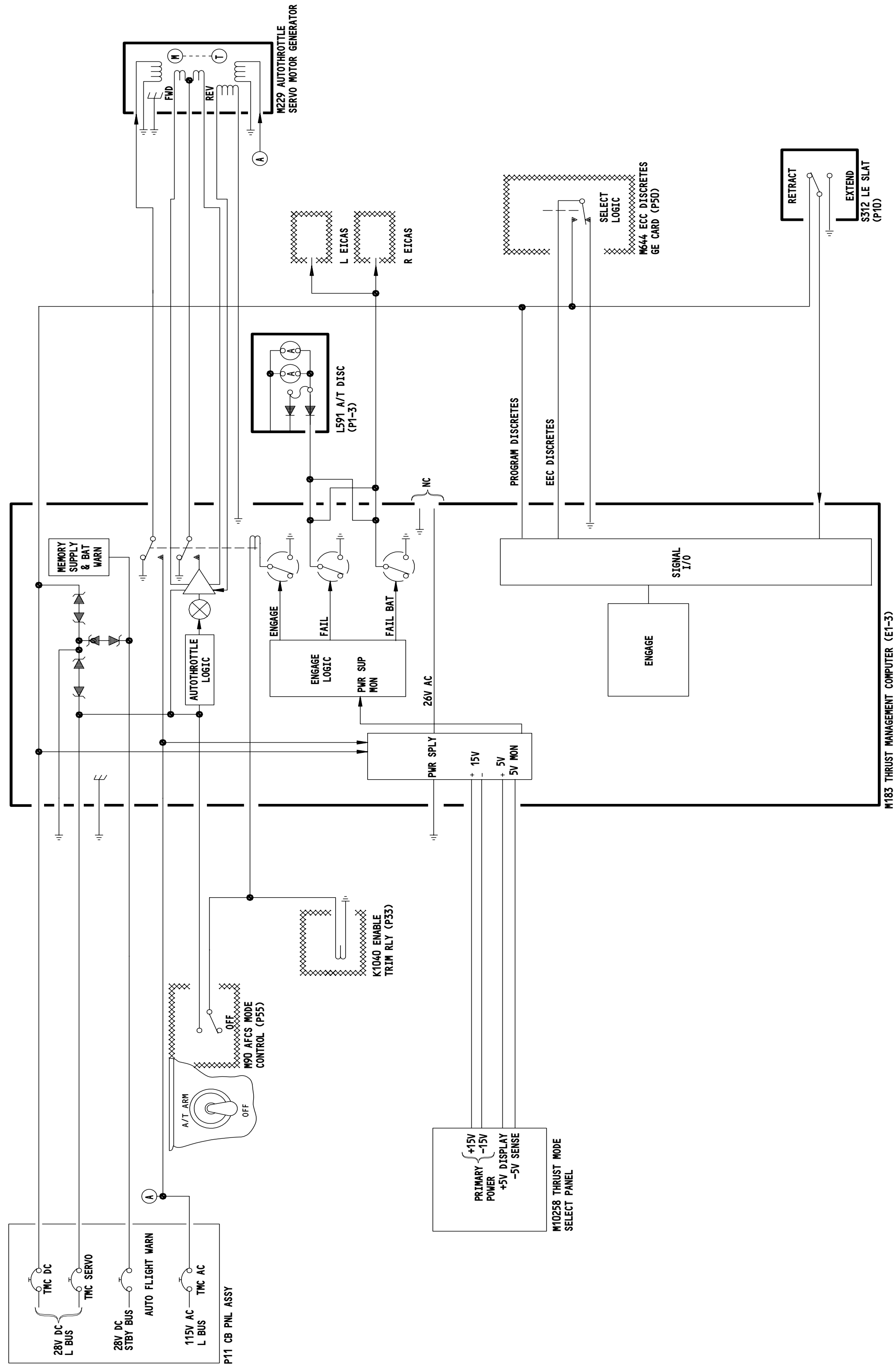
3. <u>Operation</u>

A. Functional Description

(1) Thrust Management System - Power Distribution (Fig. 7)

(a) TMC Power

1) The TMC power consists of 115v ac and 28v dc power. The 28v dc standby bus provides power for disconnect warning if TMC power fails. The excitation relay in the TMC connects 115v ac servomotor excitation voltage and servomotor command voltage when engage conditions are valid. The EMI filters remove transients from the line and internal switching noise from the power supply.



TMC Power Interface
Figure 7

L16474

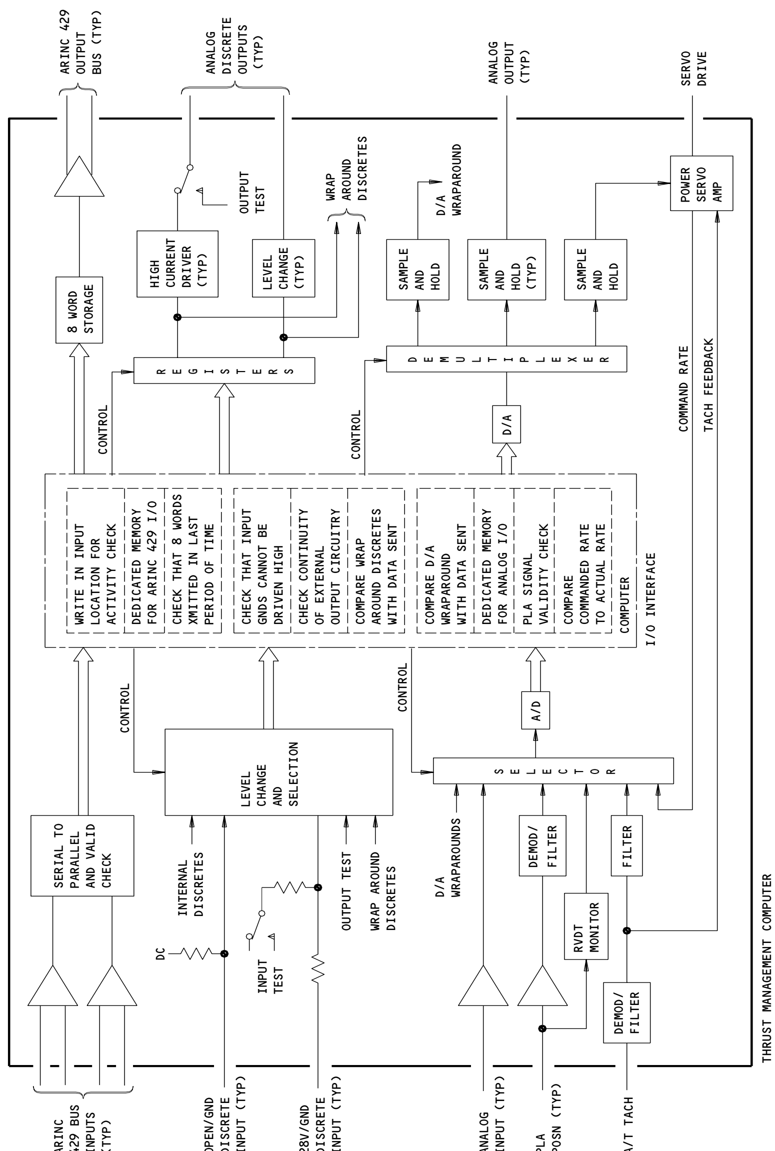
2) The 115v ac aircraft power is transformed to appropriate levels in the TMC. The 8v ac is used for TMC demodulation/modulation reference, and 26v ac is used for PLA transducer excitation. Full wave rectification provides ±15v dc and +5v dc for TMSP internal power. The TMC provides voltage regulation on all dc outputs.
3) The power supply is monitored to provide a primary power fail discrete that results in a TMC INVALID when input power is low. The abnormal steady-state ac limit is 97v ac and the dc lower limit is 18v dc. The TMC is also monitored to provide a power fail discrete if the ±15v dc or +5v dc logic supply falls below ±14v dc or +4.5v dc, respectively.

(b) TMSP Power
1) The TMSP receives +5v dc from the TMC. In order to maintain this voltage within ±0.25v dc under varying conditions, remote sensing lines are provided (+5 volts sense and ground sense). When +5 volts changes, the TMC power supply/regulator responds, thus ensuring that normal operating limits are maintained. When the voltage drops below +4.3 volts, the TMSP power loss monitor provides a TMSP status fail signal to the TMC. The TMC also provides ±15v dc power that operates the TMSP transmitter.

(c) A/T Servomotor Generator
1) The A/T servomotor generator uses 115v ac. The servo tachometer is an ac generator with a linear relationship between shaft rpm and ac voltage output. The tachometer is directly excited by 115v ac. Servomotor excitation and servomotor command voltage is controlled by the excitation relay in the TMC.

(d) A/T ARM
1) The A/T ARM switch is on the MCP. When positioned to ARM, the A/T switch connects 28v dc to arm the servomotor excitation relay and the engine trim enable relay.

(2) TMC Operation (Fig. 8)



TMC Input/Output Interface and Monitors
Figure 8

(a) Input Interface

1) Serial ARINC 429 bus inputs are converted to parallel and stored in dedicated memory. Analog discrete inputs consist of two types: open/ground and 28v dc/ground. The input level changers convert these signals into transistor to transistor logic compatible format. This is done by equating an open or 28v dc to a logical one and ground to a logical zero. The 48 level changed lines are formed into three 16-bit words. They are input along with a 16-bit internal discrete word into a one of four selector. The computer then selects the desired word for input by addressing the appropriate location. Analog inputs are selected one at a time. They are changed to a digital code in the hybrid Analog to Digital (A/D) converter for use in the digital processing section of the computer. When the conversion is complete, the digital value is stored in dedicated memory.

(b) Output Interface

1) The ARINC 429 transmitter receives 8 words at a time. These words are then transmitted serially on the 429 output bus. Analog discrete data is latched to the output register and then level changed to a voltage compatible to the airplane. These become the analog discrete outputs. Digital data is routed to the D/A converter and then multiplexed to one of eight Sample and Hold (S/H) circuits which provide the analog outputs. Two S/H provide A/T servo command output to the power servo amplifier; however, only one is used. Two S/H are used for BITE (D/A wrap around) and two are used for testing. Two S/H are spares.

(c) Failure Protection and Maintenance

1) The TMC provides failure protection to assure system safety, and maintenance functions that isolate faults to a specific LRU of the TMS or interfacing system. Failure protection and maintenance activities of the TMC are classified as fault data management, interface verification test, and maintenance functional tests.

2) The primary function of fault data management is to support system safety, prevent computing a false thrust limit, or issuing an invalid servo command. Fault data management runs continuously when the TMC is powered except during maintenance test. It interrogates all system monitors and runs self-test routines as required. When a fault is detected, fault data management responds to the type of fault and operational mode. It also supports maintenance activities by performing fault isolation and storing TMC fault information in nonvolatile memory. It continually transmits fault data to the Maintenance Control Display Panel (MCDP). When activated by touchdown, the MCDP stores the fault data for later display to maintenance personnel.

3) The interface verification test verifies that all input/output lines between the TMC and interfacing components are connected. It consists of passive and active tests. Passive tests are performed without interfering with the normal operation of the TMC. Active tests require the TMC to change the state of its outputs to verify connections. The active test are part of the maintenance functional tests.

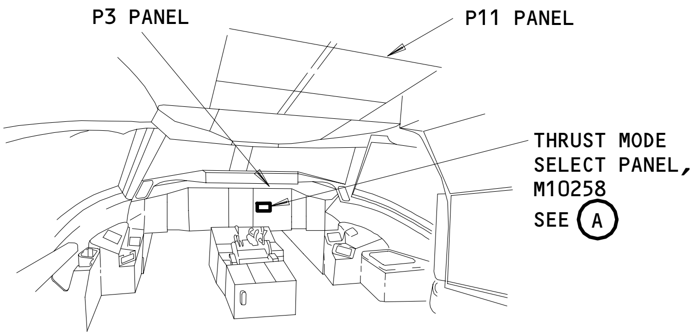
4) The maintenance functional tests provide the capability for performing TMS ground tests. The maintenance functional test mode is selected when the MCDP is placed in the GRD TEST mode. Maintenance personnel then select specific test(s) to be run via an MCDP input. The TMC then runs the test and transmits the results to the MCDP for display to the operator. The test may require operator action or verification of results. These tests can only be initiated when the airplane is on the ground.

THRUST MANAGEMENT POWER

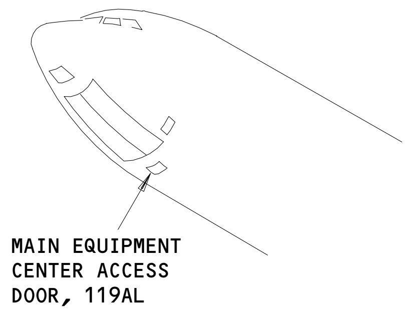
COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
CIRCUIT BREAKER -	--		FLT COMPT, P11	
AUTO FLIGHT WARN, C521		1	11A17	*
TMC AC, C501		1	11F14	*
TMC DC, C525		1	11F15	*
TMC SERVO, C512		1	11F16	*
COMPUTER - THRUST MANAGEMENT, M183		1	119AL, MAIN EQUIP CTR, E1-3	22-31-01
PANEL - THRUST MODE SELECT, M10258		1	FLT COMPT, P3	22-31-02

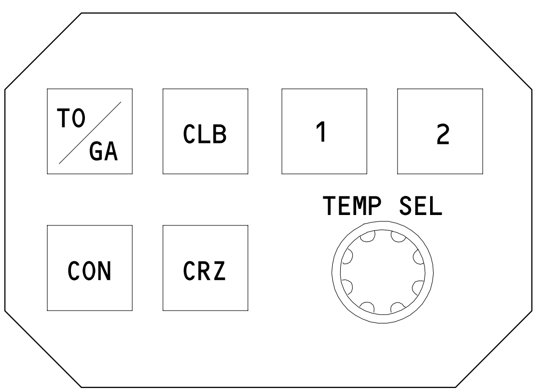
* SEE THE WDM EQUIPMENT LIST

Thrust Management Power - Component Index
Figure 101



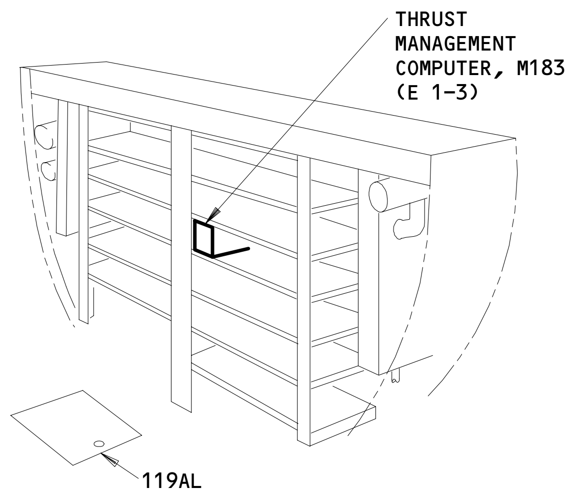
FLIGHT COMPARTMENT





THRUST MODE SELECT PANEL, M10258

A



MAIN EQUIPMENT CENTER

Thrust Management Power - Component Location
Figure 102

THRUST MANAGEMENT COMPUTER - REMOVAL/INSTALLATION

1. General
   A. The thrust management computer (TMC) is installed in the main equipment center.
      (1) The TMC is held in position by two hold-down hooks and a lower extraction lip.
      (2) Electrical connections are made through one multi-pin connector located on the back of the TMC. The connector has three separate inserts (A, B and C) and index pins. The C insert contains only the TMC power pins. The A and B inserts contain the TMC signal pins and low voltage power pins. Index pins provide connector keying which ensures correct TMC installation.

TASK 22-31-01-004-015

2. Remove the Thrust Management Computer
   A. References
      (1) AMM 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
      (2) AMM 20-10-01/401, E/E Rack-Mounted Components
      (3) AMM 20-41-01/201, Electrostatic Discharge Sensitive Devices
   B. Access
      (1) Location Zones
          119/120 Main Equipment Center
          211/212 Flight Compartment

      (2) Access Panel
          119AL Main Equipment Center

   C. Prepare for Removal

      S 864-003
      (1) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11A17, AUTO FLIGHT WARN
          (b) 11F14, TMC AC
          (c) 11F15, TMC DC
          (d) 11F16, TMC SERVO

      S 014-001
      (2) Open the access panel, 119AL, for the thrust management computer (AMM 06-41-00/201).

D. Remove the Thrust Management Computer

S 914-002

CAUTION: DO NOT TOUCH THE THRUST MANAGEMENT COMPUTER BEFORE YOU DO THE PROCEDURE FOR DEVICES THAT ARE SENSITIVE TO ELECTROSTATIC DISCHARGE. ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE THRUST MANAGEMENT COMPUTER.

(1) Do the procedure for devices that are sensitive to electrostatic discharge (AMM 20-41-01/201).

S 024-004

(2) Remove the thrust management computer (AMM 20-10-01/401).

TASK 22-31-01-404-005

3. Install the Thrust Management Computer

A. References

(1) AMM 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
(2) AMM 20-10-01/401, E/E Rack-Mounted Components
(3) AMM 20-41-01/201, Electrostatic Discharge Sensitive Devices
(4) FIM 22-00-03/101, Autoflight BITE MCDP Ground Tests
(5) AMM 24-22-00/201, Electrical Power - Control
(6) AMM 27-61-00/201, Spoiler/Speedbrake Control System

B. Access

(1) Location Zones

119/120 Main Equipment Center
211/212 Flight Compartment

(2) Access Panel

119AL Main Equipment Center

C. Install the Thrust Management Computer

S 914-016

CAUTION: DO NOT TOUCH THE THRUST MANAGEMENT COMPUTER BEFORE YOU DO THE PROCEDURE FOR DEVICES THAT ARE SENSITIVE TO ELECTROSTATIC DISCHARGE. ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE THRUST MANAGEMENT COMPUTER.

(1) Do the procedure for devices that are sensitive to electrostatic discharge (AMM 20-41-01/201).

S 424-006

(2) Install the thrust management computer (AMM 20-10-01/401).

D. Do a Test of the Thrust Management Computer

S 864-007

(1) Supply electrical power (AMM 24-22-00/201).

S 864-008

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
   (a) 11A17, AUTO FLIGHT WARN
   (b) 11F14, TMC AC
   (c) 11F15, TMC DC
   (d) 11F16, TMC SERVO

S 864-009

WARNING: THIS TEST USES MOVEMENT OF THRUST LEVERS AND CAN CAUSE AIRPLANE MOVEMENT IF ENGINES ARE ON OR SPOILER/SPEEDBRAKE MOVEMENT IF PRESSURE TO THE HYDRAULIC SYSTEMS IS ON. REFER TO THE SPOILER/SPEEDBRAKE DEACTIVATION PROCEDURE (AMM 27-61-00/201). ACCIDENTAL ENGINE OPERATION OR SPOILER/SPEEDBRAKE MOVEMENT CAN CAUSE BAD INJURY TO PERSONS.

(3) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoilers.

S 864-013

(4) Make sure the engines are not on.

S 864-010

(5) Set the MCDP to OFF or go into Flight Faults Mode to make sure that the MCDP is out of Ground Test Mode.

   NOTE: Before the MCDP Ground Test 02 TMC is done, the MCDP must be put out of Ground Test Mode to remove from the MCDP memory problems that have been corrected.

S 714-011

(6) Go into Ground Test Mode and do the MCDP Ground Test 02 TMC (FIM 22-00-03/101).

E. Put the Airplane Back to Its Initial Condition

S 864-017

(1) Do the activation procedure for the spoilers if you did the deactivation procedure (AMM 27-61-00/201).

S 414-014

(2) Close the access panel, 119AL (AMM 06-41-00/201).

S 864-012

(3) Remove electrical power if it is not necessary (AMM 24-22-00/201).

THRUST MODE SELECT PANEL - REMOVAL/INSTALLATION

1. General
   A. The thrust mode select panel (TMSP) is installed on the center instrument panel P3-1. The TMSP is held in position by a mounting clamp. One multi-pin connector is at the rear of the unit.

TASK 22-31-02-004-002

2. Remove the Thrust Mode Select Panel
   A. Access
      (1) Location Zone
          211/212 Flight Compartment

   B. Remove the Thrust Mode Select Panel

      S 864-001
      (1) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
          (a) 11A17, AUTO FLIGHT WARN
          (b) 11F14, TMC AC
          (c) 11F15, TMC DC
          (d) 11F16, TMC SERVO

      S 034-003
      (2) Loosen the mounting clamp screws adjacent to the TMSP front panel.

      S 024-004

      CAUTION: YOU MUST CARFEULLY REMOVE THE TMSP FROM THE INSTRUMENT PANEL TO PREVENT DAMAGE TO THE ELECTRICAL CABLE.

      (3) Remove the TMSP from the instrument panel.

      S 034-005
      (4) Disconnect the electrical connector.

TASK 22-31-02-404-006

3. Install the Thrust Mode Select Panel
   A. References
      (1) AMM 22-00-02/201, Autoflight BITE

(2) AMM 24-22-00/201, Electrical Power - Control

B. Access

(1) Location Zone

211/212 Flight Compartment

C. Install the Thrust Mode Select Panel

S 434-007

(1) Connect the electrical connector to the TMSP.

S 424-008

CAUTION: YOU MUST CAREFULLY INSTALL THE TMSP IN THE INSTRUMENT PANEL TO PREVENT DAMAGE TO THE ELECTRICAL CABLE.

(2) Install the TMSP in the instrument panel.

S 434-009

(3) Tighten the mounting clamp screws.

D. Do a Test of the Thrust Mode Select Panel

S 864-010

(1) Supply electrical power (AMM 24-22-00/201).

S 864-011

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11A17, AUTO FLIGHT WARN
(b) 11F14, TMC AC
(c) 11F15, TMC DC
(d) 11F16, TMC SERVO

S 714-012

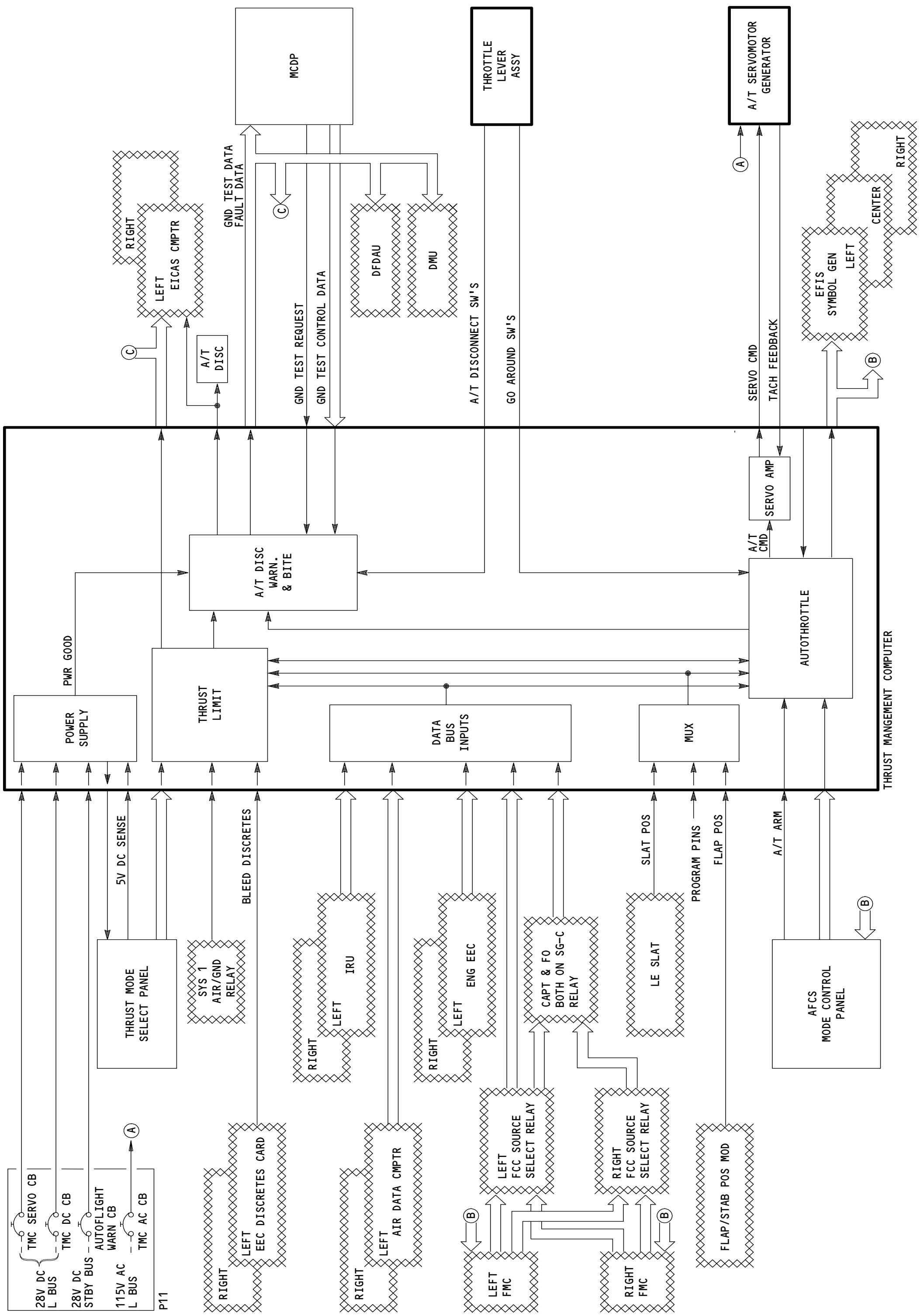
(3) Do the MCDP Ground Test 05 TMSP (AMM 22-00-02/201).

S 864-013

(4) Remove electrical power if it is not necessary (AMM 24-22-00/201).

THRUST MANAGEMENT SYSTEM - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The Thrust Management System (TMS) consists of a Thrust Management Computer (TMC) and Thrust Mode Select Panel (TMSP). These control a servomotor generator that drives the thrust levers. The TMS is a digital control system receiving inputs from the following primary interfaces:
      (1) Air Data Computers (ADCs)
      (2) Inertial Reference Units (IRUs)
      (3) Flight Management Computers (FMCs)
   B. The TMS functions include thrust limit computation, thrust lever control, and data display. Engine thrust limits are computed for the selected mode on the TMSP, then displayed on the upper EICAS display unit. The flight crew uses this data to monitor autothrottle (A/T) operations or to assist in manually setting engine thrust. The A/T servomotor generator is connected to thrust levers through a brake pack assembly. The brake pack allows manual thrust commands without backdriving the servo. The autothrottle provides thrust lever control to selected inputs throughout the full flight regime. The control laws governing A/T operation are based on airspeed, Mach, vertical speed and thrust target limits.
   C. The autothrottle may operate independent of, or together with the Flight Management Computer System (FMCS) and the Autopilot/Flight Director System (AFDS). The FMCS commands both the AFDS and TMS to follow flight path and speed commands as defined by FMC performance and vertical navigation functions (AMM 34-61-00, Flight Management Computer System, and AMM 22-10-00/001, Autopilot (Flight Control)).
   D. Thrust Management Computer (TMC)
      (1) The TMC is on shelf E1-3 of the main electrical/electronic equipment bay. It provides thrust limit computation and thrust lever control to Engine Pressure Ratio (EPR), vertical speed, airspeed, and Mach references. Additional functions provided are: minimum and maximum speed protection, thrust limit protection, and retarding the thrust levers during autoland.
   E. Controls



Thrust Management System - General
Figure 1 (Sheet 1)

586731

TABLE 1 INPUT PARAMETERS

SOURCE	SIGNAL
MODE CONT PNL	ALT SEL ELEV SPD CMD MACH SEL SPD BRK HDL POS SPD SEL STAB POS DISCRETES:- IAS/MACH SEL F/D ON-F/O F/D ON-CAPT SPD MODE REQ THRUST MODE REQ ALT HOLD REQ FL CH MODE OPER V/NAV MODE G/S MODE OPER THROT RETARD PITCH SPD CONT A/P CMD C ENGA A/P CMD L ENGA A/P CMD R ENGA V/S MODE OPER AFDS G/A OPER
IRU	NORM ACCEL PITCH RATE ROLL RATE YAW RATE XTK HRZ ACCEL FPAC GS INS ALT INS V/S PITCH ANGLE ROLL ANGLE TKA TRUE TRUE HDG DISCRETES:- ALIGN FAULT NO IRS INITIAL EXCESS MOTION ADC/IRU FAULT
FMC	ASSUMED TEMP EPR TARGET SEL MACH TARGET AS DISCRETES: A/T MACH MODE REQ A/T A/S MODE REQ CLB MODE REQ CON MODE REQ CRZ MODE REQ G/A MODE REQ T/O MODE REQ A/T LOW GAIN ARM IDLE THRUST REQ THROTTLE DORMANT THRUST MODE REQ
ADC	ALT (BARO 4) ALT (29,92) COMPUTED AS IMPACT PRESS IND AOA MACH MAX OPRTG SCHED SAT TAT TOTAL PRESS TAS DISCRETES:- P/S HT ON-L P/S HT ON-R TAT PROBE HT ON
EEC	EPR CMD FROM TRA EPR COMMAND EPR MAX N1 ACTUAL N2 ACTUAL N2 DERIVATIVE PARTIAL EPR/TRA PARTIAL FN/EPR TRA TRA MAX RATE LIM TRA MIN RATE LIM DISCRETES:- CHAN IN ALT MODE CHANNEL IN CONTROL ALT MODE MAN SEL PROBE HEAT ON BUMP SELECTED ENGINE RTG CODE 1 ENGINE RTG CODE 2 ENGINE RTG CODE 3 ENGINE RTG CODE 4 TRIM SYSTEM LTD TRIM SYSTEM LOCKED TRIM SYS AUTHORITY ENGINE LOCATION A ENGINE LOCATION B AIRCRAFT IDLE FLAT FLIGHT IDLE A/T ARMED
TMSP	DISCRETES:- TAT STAT FAIL TAT MWI FAIL TMS STATUS FAIL LIM/TEMP MWI FAIL MODE/REF MWI FAIL SEL TEMP BIT 0 SEL TEMP BIT 1 SEL TEMP BIT 2 SEL TEMP BIT 3 SEL TEMP BIT 4 RATING 1 REQ RATING 2 REQ CON MODE REQ TO/GA MODE REQ CRZ MODE REQ CLB MODE REQ
MCDP	TEST CONT 13 TEST CONT 14 TEST CONT 15 TEST CONT 16

TABLE 2 OUTPUT PARAMETERS

DESTINATION	SIGNAL
EFIS SYMBOL GENERATOR	TMS FLT MODE ANN
FMC	TMS MODE STATUS ENG BLEED STATUS EPR ACTUAL - L EPR ACTUAL - R EPR REFERENCE FLAP POSITION TEMP SELECTED DISCRETE PARAMETERS 3
MODE CONT PNL (L & R BUS)	TMS MODE STATUS VERT SPD CMD
TMSP (NOT USED)	MAX LIMIT DISPLAY MODE DISPLAY REFERENCE DISPLAY TAT DISPLAY TEMP SELECTED TEST WORD
EICAS CMPTR	EPR ACTUAL - L EPR ACTUAL - R EPR BUG DRIVE - L EPR BUG DRIVE - R EPR REF MAX EPR LIMIT TAT TEMP SELECTED DISCRETE PARAMETERS 2 DISCRETE PARAMETERS 3
MCDP	FAULT DATA GND TEST DATA INTERFACE FAULT DATA
DFDAU, DMU	EPR BUG DRIVE - L EPR BUG DRIVE - R MAX EPR LIMIT TEMP SELECTED DISCRETE PARAMETERS 1 DISCRETE PARAMETERS 2

Thrust Management System - General
Figure 1 (Sheet 2)

(1) The A/T ARM switch is on the left side of the AFCS Mode Control Panel (MCP). When armed, the MCP allows crew selection of autothrottle control modes. Mode engagement is accomplished directly through the N1/EPR/THR, SPD, and FLCH mode select switches or indirectly through the VNAV mode select switch. When in the VNAV mode, the FMCS selects autothrottle modes and thrust limits. The Thrust Mode Select Panel (TMSP) allows crew selection of thrust limit modes, alternate thrust limit ratings, and assumed temperature.

F. Switches

(1) Disengagement of the autothrottle occurs if: either of two autothrottle disconnect switches (one on each thrust lever) is pressed, the A/T ARM switch on the MCP is positioned to OFF, or either thrust lever is set in the reverse thrust position. Pressing either of two go-around switches on the thrust levers causes the TMS to adjust thrust for a go-around.

G. Sensors

(1) The Inertial Reference Units (IRUs) and Air Data Computers (ADCs) provide attitude and environmental data to the TMS. The Electronic Engine Control (EEC) provides EPR.

H. Outputs

(1) TMS outputs consist of:

(a) The Thrust Management Computer outputs to the servomotor generator which drives the thrust levers. The thrust lever position is transmitted to the Thrust Management Computer by the Electonic Engine Control using the thrust lever resolver position data.

(b) Fault data output to the MCDP.

(c) Display of the thrust limit mode and calculated thrust limit on the upper EICAS display unit.

(d) Reference "bug" on EPR indicator driven to the calculated limit or thrust target.

I. Annunciation

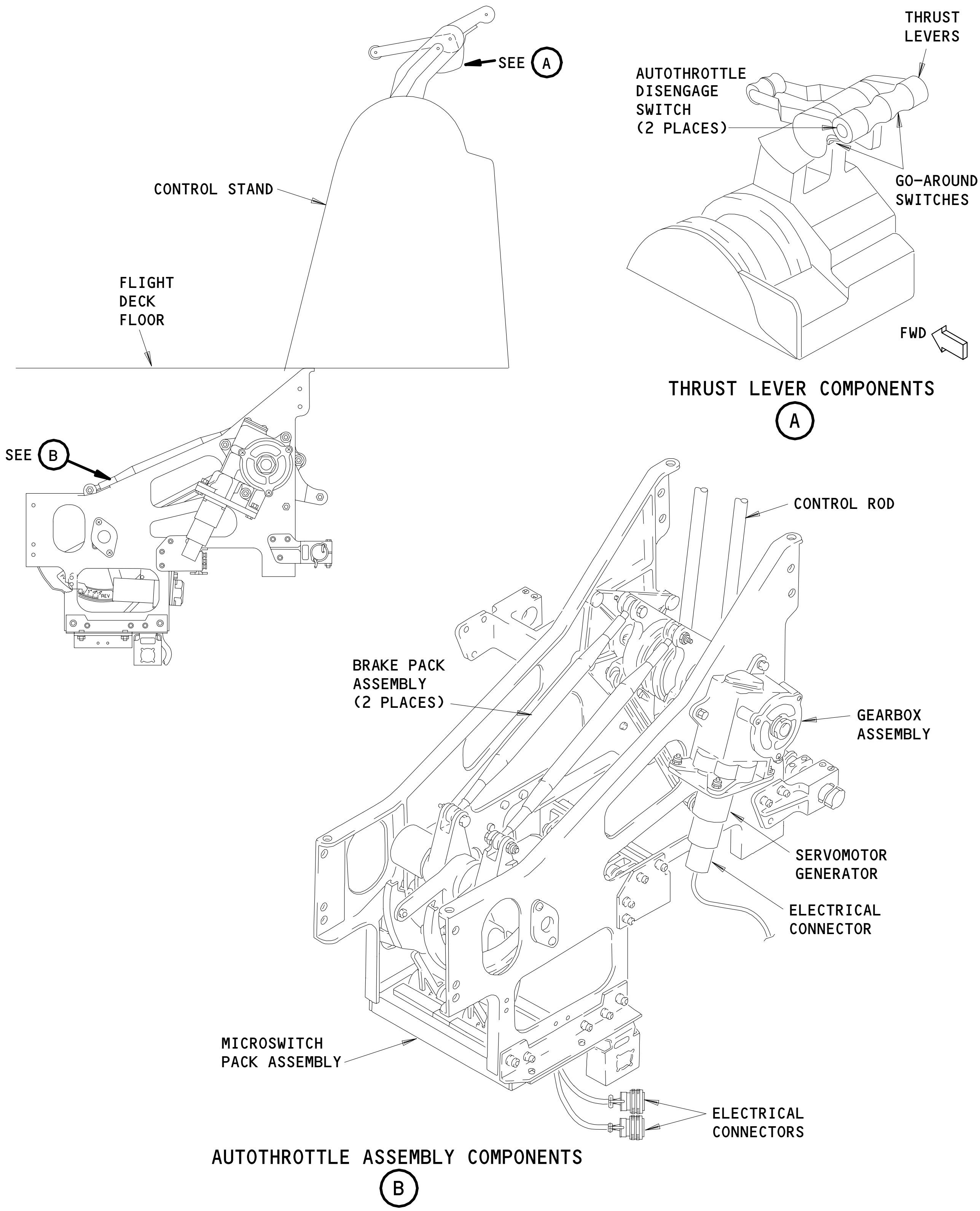
(1) Annunciation of selected autothrottle modes appears on the left side of the Electronic Attitude Director Indicator (EADI). Thrust limit modes are selected on the TMSP and displayed on the EICAS display unit. Upon autothrottle disengage, the A/T DISC light on the center instrument panel P1-3 and the Master Caution light on the glare shield illuminate. The A/T DISC light will also illuminate when the TMC is in ground test mode. The EICAS will display 'A/T DISC' and an aural caution will sound.

J. Test

(1) The Maintenance Control Display Panel (MCDP) records in-flight failures and initiates ground tests for the TMC. It identifies failures to the LRU level. In-flight failures are recorded for display during ground interrogation (AMM 22-41-00, Maintenance Monitor).

2. <u>Component Details</u>

A. Thrust Management System Components (Fig. 2)



Thrust Management System Components
Figure 2

(1) Autothrottle Assembly
   (a) The autothrottle assembly has a gearbox/servo assembly, two resolver assemblies, a brake pack assembly, a switch pack assembly and two interlock actuators.
   (b) The autothrottle assembly gives full range FLY-BY-WIRE thrust control during takeoff, climb, cruise approach and landing either by manual input from the thrust levers or autothrottle input from the TMS. When the thrust levers are manually moved, the resolver sends an electrical signal to the ECC on the engine to increase or decrease thrust. When the autothrottle is engaged, the thrust management computer compares the difference between the set speed and actual airplane speed and commands the servomotor to rotate clockwise and counterclockwise.
(2) Autothrottle Servomotor Generator
   (a) The autothrottle servomotor generator is below the flight deck floor under the thrust lever control stand. This device uses TMC commands to automatically control thrust lever position. It has a two-phase motor and single-phase generator (tachometer). The motor has a fixed field winding which uses 115 vac excitation when the autothrottle system is engaged. The motor also has two control windings controlled by the servo power amplifier in the TMC. The generator uses 115 vac excitation on one winding with a second winding providing a tachometer output for servo loop damping. The servomotor drives at a rate proportional to the amplitude of the control voltage.
   (b) The splined output shaft of the servomotor generator mates with the gearbox assembly. Three bolts hold the servomotor generator to the gearbox.
(3) A/T Disengage Switch
   (a) An autothrottle disengage switch (double-push reset) is on the end of each thrust lever. The switch is normally closed. When depressed, it disengages the autothrottle and provides an autothrottle disengage warning. A second press of the switch resets the warning circuits.
(4) Go-Around Switch
   (a) One palm-operated go-around switch is at the base of each thrust lever. The switch has dual contacts. One contact is normally grounded and the other is normally open. Actuating the switch engages the go-around mode if the go-around mode is armed (AMM 22-11-00/001 Autopilot/Flight Director Power for additional description).
(5) Brake Pack Assembly
   (a) The autothrottle brake pack assembly is directly under the thrust lever control stand. The brake pack assembly consists of two brake assemblies mounted on a splined shaft. The brake pack assembly provides feel friction to the thrust levers and allows the pilots to manually override the Thrust Management System.

(b) The servomotor drives the resolver assemblies and the brake pack assembly during thrust management system operation. The resolver provides an electrical input to the EEC (Electronic Engine Control) on the engine.

(6) Microswitch Pack Assembly and Leading Edge Slat Switches (Fig. 3)

(a) The microswitch pack assembly is attached to the bottom of the autothrottle drive assembly. It is held in place by four bolts. The assembly is capable of containing 16 sets of switches. The number of switches installed depends on airplane configuration. The switches contained in the assembly are shown in figure 3.

(b) The leading edge slat switch senses when the slats are extended. It is a Single Pole Double Throw (SPDT) switch that connects 28v dc when slats are retracted. The switch is actuated when the flap handle is moved to the 1 position. Once actuated, the TMC estimates the slats moving linearly to the flaps 1 position in a 5-second time period.

(c) The arm assembly on the flap handle is moved by the flap lever quadrant in the flaps 1 or greater position. The switch assembly contains four sets of switches which are actuated by the arm assembly movement. Three sets are used by the flight control computers. One set is used by the TMC.

3. Operation

A. Functional Description

(1) Thrust Management System - Block Diagram (Fig. 4)

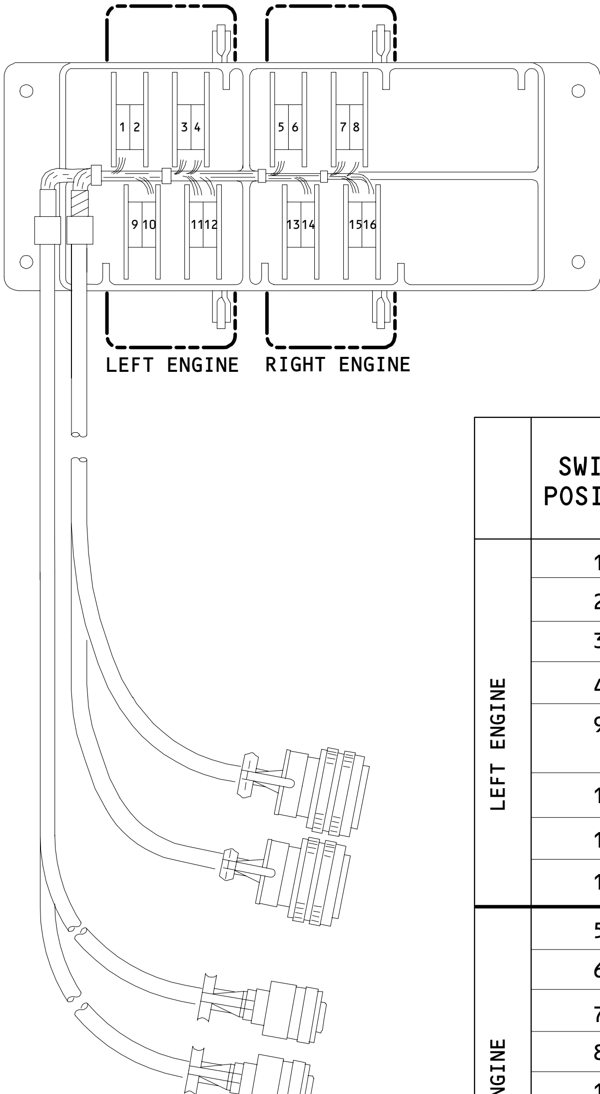
(a) Thrust Control

1) Control of thrust is provided by the Thrust Resolver Assembly (TRA) transducer which provides a signal to the EPCS. The TRAs are controlled manually by the thrust levers or automatically by the A/T servomotor generator. The EPCS provides the TMC with thrust lever angle information to close the servoloop. The EPR transmitters provide actual EPR to the Electronic Engine Control (EEC) and EICAS computers. The EEC uses this information along with throttle lever angle, air data and bleed status to:

a) Control and Limit thrust

b) Linearize the throttle-to-thrust-rating relationship

c) Drive the engine thrust indicator command sector on the upper EICAS display unit

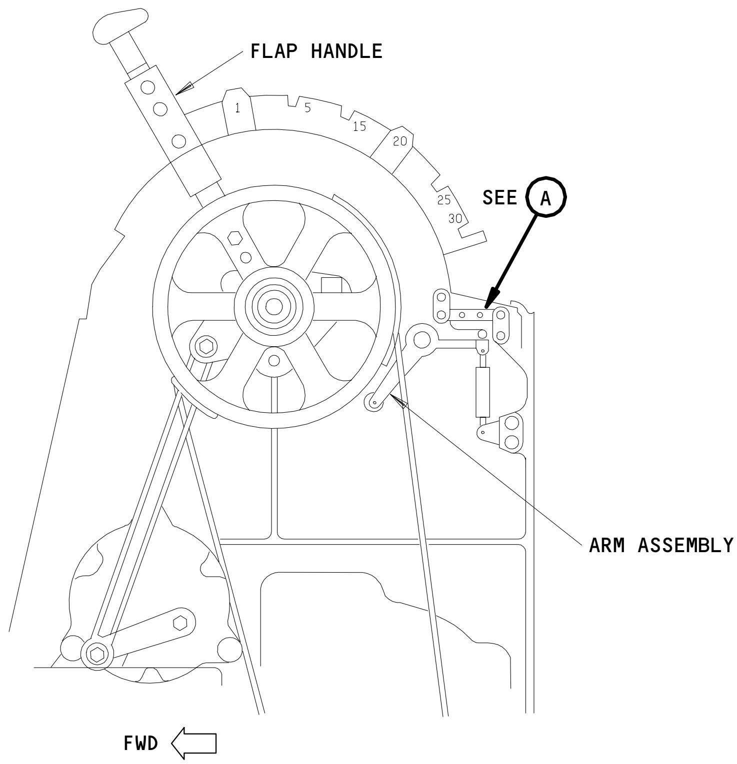


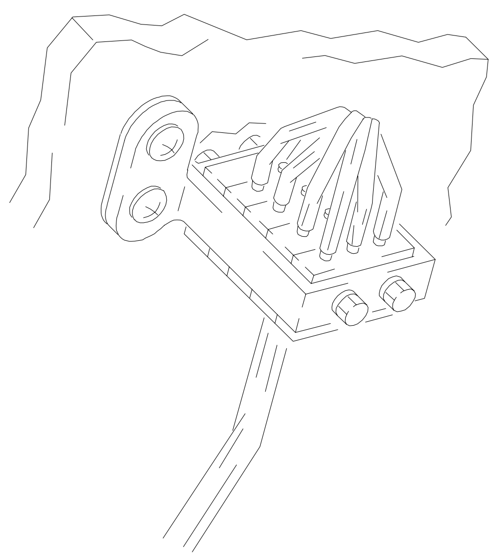
	SWITCH POSITION	SWITCH OR SPACER	FUNCTION
LEFT ENGINE	1	S3	LEFT AUTOBRAKE/AUTOBRAKE RTO
	2	SPACER	NONE
	3	S1	LANDING WARNING LEFT
	4	S2	LEFT AUTOBRAKE/AUTOBRAKE RTO
	9	S11	REVERSE THRUST DIRECTIONAL VALVE LEFT
	10	S21	LEFT THRUST REVERSE LOCK
	11	S17	LOAD SHED/PRESSURE CONTROL LEFT
	12	S10	SPEED BRAKE RETRACT LEFT
RIGHT ENGINE	5	S7	RIGHT AUTOBRAKE/AUTOBRAKE RTO
	6	SPACER	NONE
	7	S6	RIGHT AUTOBRAKE/AUTOBRAKE RTO
	8	S5	LANDING WARNING RIGHT
	13	S8	REVERSE THRUST DIRECTIONAL VALVE RIGHT
	14	S22	RIGHT THRUST REVERSE LOCK
	15	S18	LOAD SHED/PRESSURE CONTROL RIGHT
	16	S14	SPEED BRAKE RETRACT RIGHT

TMS: THRUST MANAGEMENT SYSTEM
RTO: REJECT TAKE-OFF

MICROSWITCH PACK ASSEMBLY

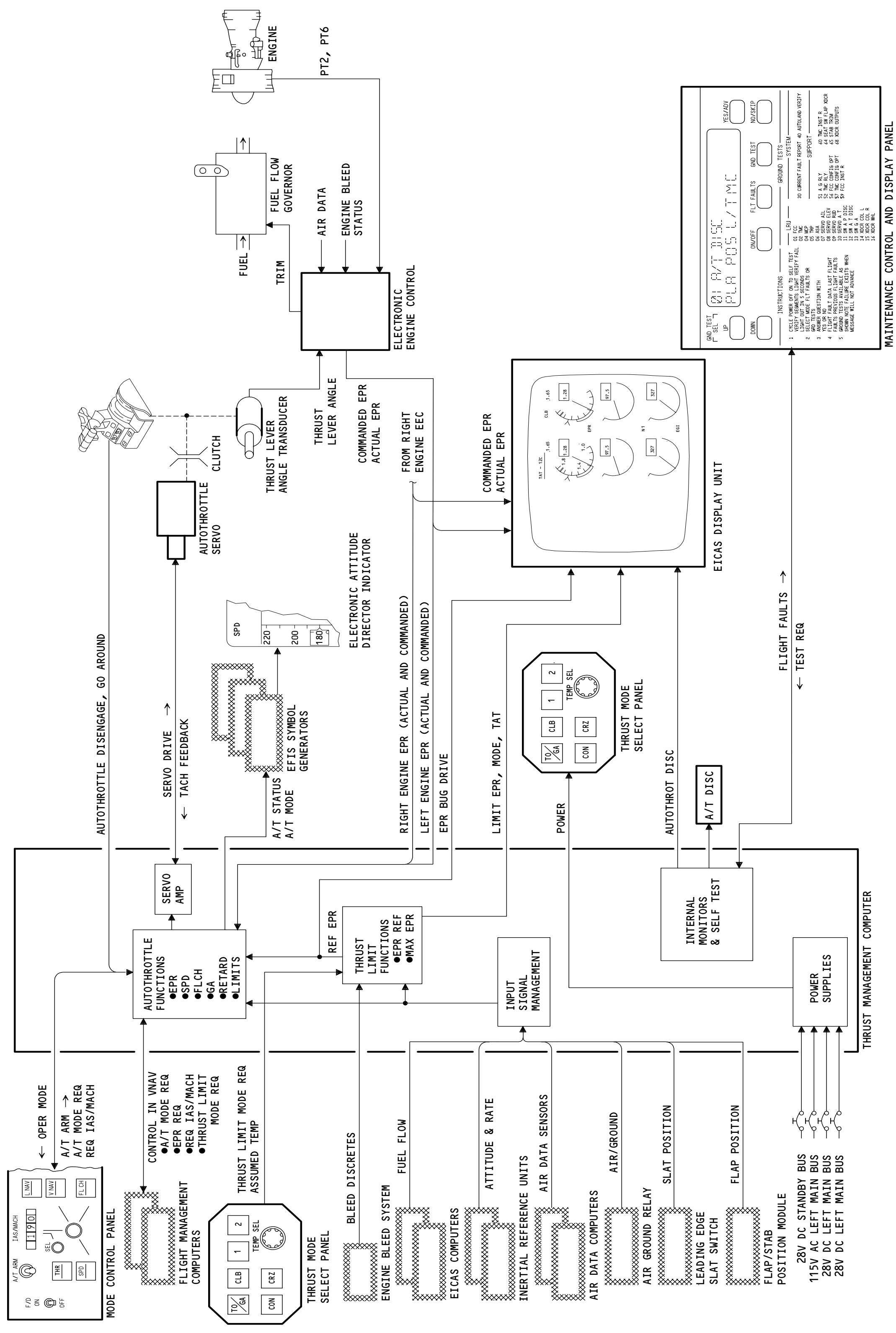
Microswitch Pack Assembly and Leading Edge Slat Switches
Figure 3 (Sheet 1)





SLAT SWITCH ASSEMBLY
A

Microswitch Pack Assembly and Leading Edge Slat Switches
Figure 3 (Sheet 2)



Thrust Management System - Block Diagram
Figure 4

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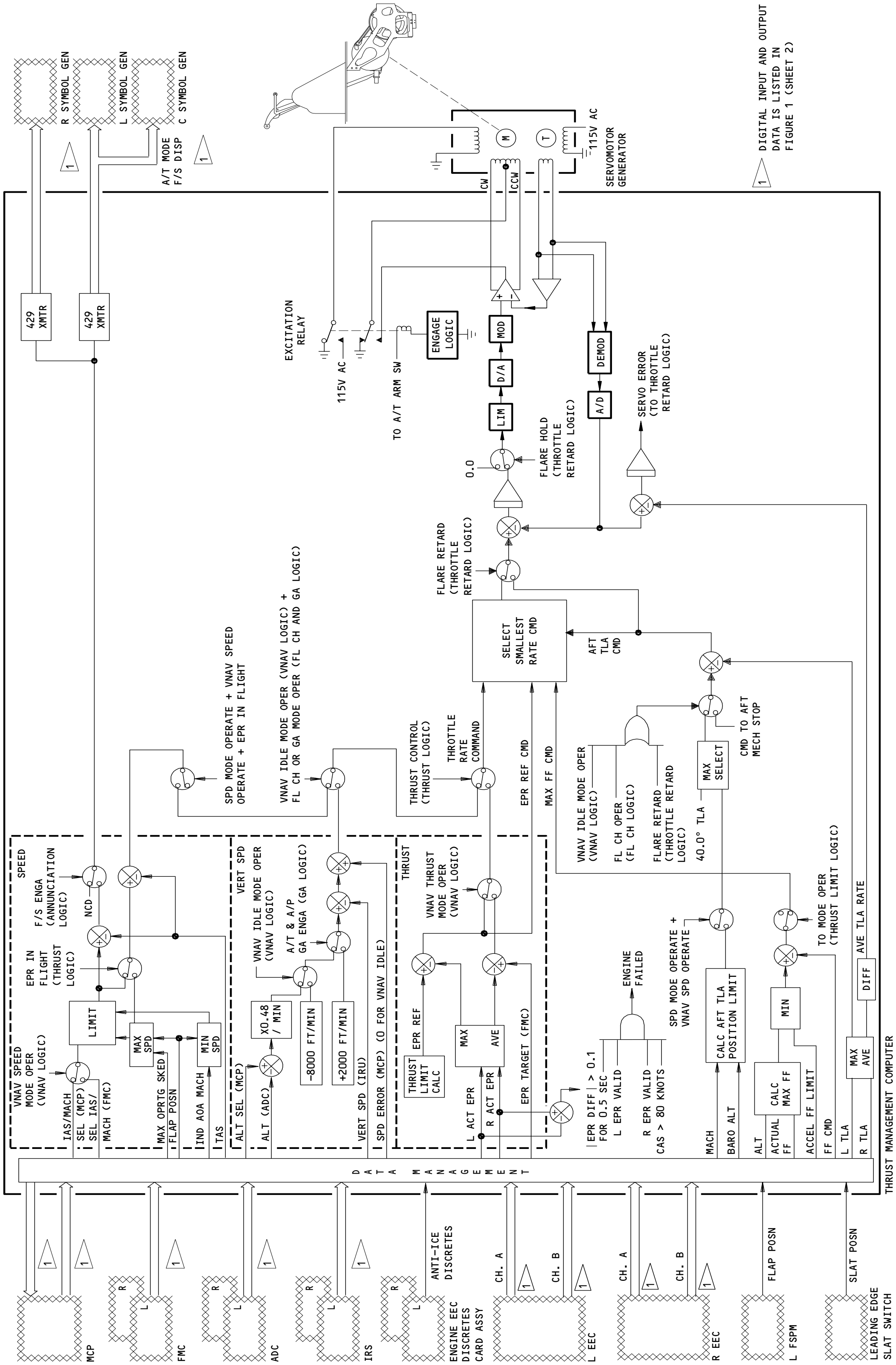
(b) Flight Deck Indications
  1) The primary engine format is shown on the upper EICAS display unit. Engine parameters shown on this format are required to set and monitor engine thrust. Digital displays are provided for reference EPR, thrust limit mode, total air temperature, and selected temperature. The EPR indicators on the EICAS display unit provide an analog-type round-dial display for reference EPR, commanded EPR, and actual EPR. Caution advisory messages relating to the TMS are also displayed on the EICAS display unit.
  2) Engaged autothrottle mode and status is shown on the left side of the EADI display.
  3) The A/T DISC light on panel P1-3 turns on when the autothrottle disengages.

(c) Sensor Inputs
  1) Air data from ADCs
  2) Attitude and rate from IRS
  3) Aircraft configuration discretes
  4) EEC data, including engine parameters, engine commands and throttle lever position
  5) Flap/slat position
    a) Flap/slat position includes trailing-edge flap position from the left flap/stabilizer position module and leading-edge slat extend discrete from the aisle stand flap handle activated microswitch. The slat discrete gives a signal that the slat is to extend when the flap handle is set to the flap 1 or greater position. The TMC uses the trailing-edge flap position and the leading-slat extend state to find the equivalent flap handle position, which is then used in the autothrottle and thrust limit functions. A check on the precision of the leading-edge slat discrete is also done by checking that it gives an extended slat signal if the equivalent flap handle position is greater than 10 degrees, as derived from the trailing-edge flap position. In the event that the "alternate flap extension" is used, the flap handle should be put in position so that the leading-edge slat status discrete is set correctly.

(d) Control Inputs
  1) Control inputs are received from three sources. The MCP provides A/T ARM, A/T mode select, and IAS/Mach select. The TMSP provides thrust limit mode requests and assumed temperature. Autothrottle mode requests; EPR, IAS, and Mach targets; and thrust limit mode requests are supplied by the FMC when VNAV is engaged.

(e) TMC Internal Functions
  1) Autothrottle functions (Fig. 5):



Autothrottle Block Diagram
Figure 5

a) Thrust is controlled in the EPR mode to the computed EPR REF target. When VNAV is engaged, the FMC selects EPR. If the SPD mode is selected on the MCP, speed is controlled to the selected target on the IAS/MACH display. The FMC selects the target speed in the VNAV mode. In the Flight Level Change (FLCH) mode, the TMC controls the vertical speed so that the transition to a new altitude takes about 2 minutes subject to the EPR REF for climb, and aft limit for descent. The TMC also sets thrust to the EPR REF for an autothrottle only Go-Around (G/A). If both the autopilot and autothrottle are engaged the TMC controls rate of climb during G/A and rate of retard during flare. All of these functions are subject to EPR, TLA, and speed limits.

b) For the autothrottle control laws, selection of signal source is automatic. No external switching is required by the crew, although manual switching of the ADC inputs is allowed through the ADC instrument source select switches. Also the FMC input data is switched on the same buses as the FCCs and therefore the buses can be switched using the FMC NAV source select switch. The TMC will select data from the left channel of the ADC, the FMC and the IRU if the autopilot is not engaged, or if only the left or center channel is engaged. If only the right channel is engaged, data will be taken from the right channel sensors. During multichannel operation; sensor selection corresponds to the autopilot first engaged. If the selected sensor is faulty the TMC will switch to data on the equivalent sensor from the other channel if the other sensor is valid or disengage the autothrottle if the other input channel is invalid. The autothrottle will also disengage if the autopilot is in VNAV mode and the corresponding FMC channel transmits faulty data.

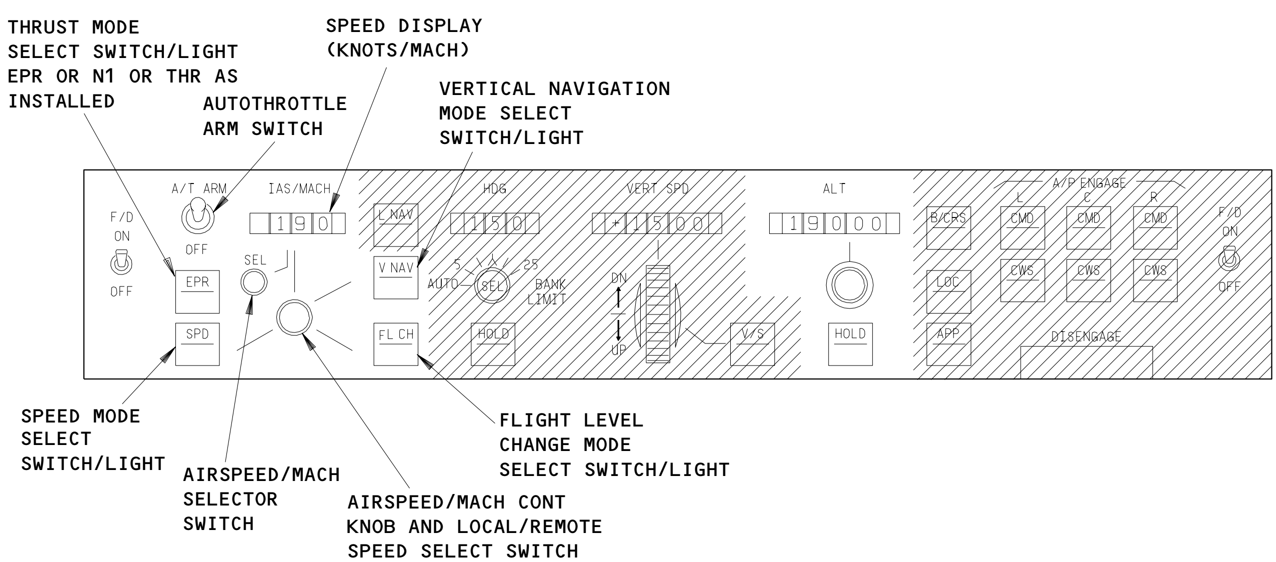
2) Thrust Limit Functions:

a) The REF and MAX EPR thrust limits are computed by the TMC. The limits are based on engine type, thrust limit mode selected, flight conditions, and airplane configuration.

b) For the thrust limit calculations, selection of signal source is automatic. No external switching is required by the crew, although manual switching of the ADC inputs is allowed through the ADC Normal/Alternate switches. The TMC will select data from the left channel of the ADC. If the left channel is faulty the TMC will switch to data from the right sensor. If the right input channel is also invalid, the EICAS displays a blank TMC mode and EPR/N1 REF.

3) Internal Monitors and Self Tests:
   a) The internal monitoring system of the TMC performs tests requested by the MCDP and senses autothrottle disconnect. All faults are recorded and transmitted to the MCDP.

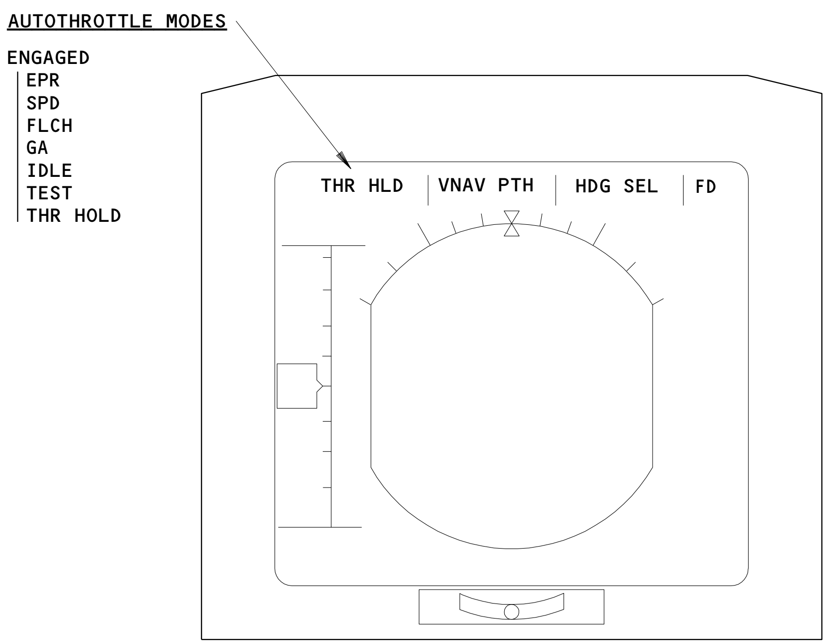
(2) Mode Control Panel - Autothrottle Functions (Fig. 6)
   (a) Autothrottle Arm Switch
      1) The A/T arm switch is a two-position toggle switch that sends a 28 vdc analog discrete signal to the TMC. When positioned to A/T ARM, it arms the excitation relay. The excitation relay connects 115 vac to the servomotor generator only after a mode is engaged. The tachometer/generator excitation is connected directly to 115 VAC. The A/T ARM switch is also used to re-engage the A/T into FLCH or VNAV after it has been disconnected by cycling the switch to 'OFF' then to 'A/T ARM'.
   (b) Mode Select Switches (EPR/N1/THR as installed, SPD, FLCH, VNAV)
      1) The mode select switches provide mode engagement after arming has occurred. The switch dot-bar matrix illuminates when the mode request is accepted and operable.
   (c) Speed Switches and Display
      1) Speed select is displayed in two forms:
         a) 100-399 knots in increments of 1 knot.
         b) 0.400-0.950 Mach in increments of 0.010 Mach.



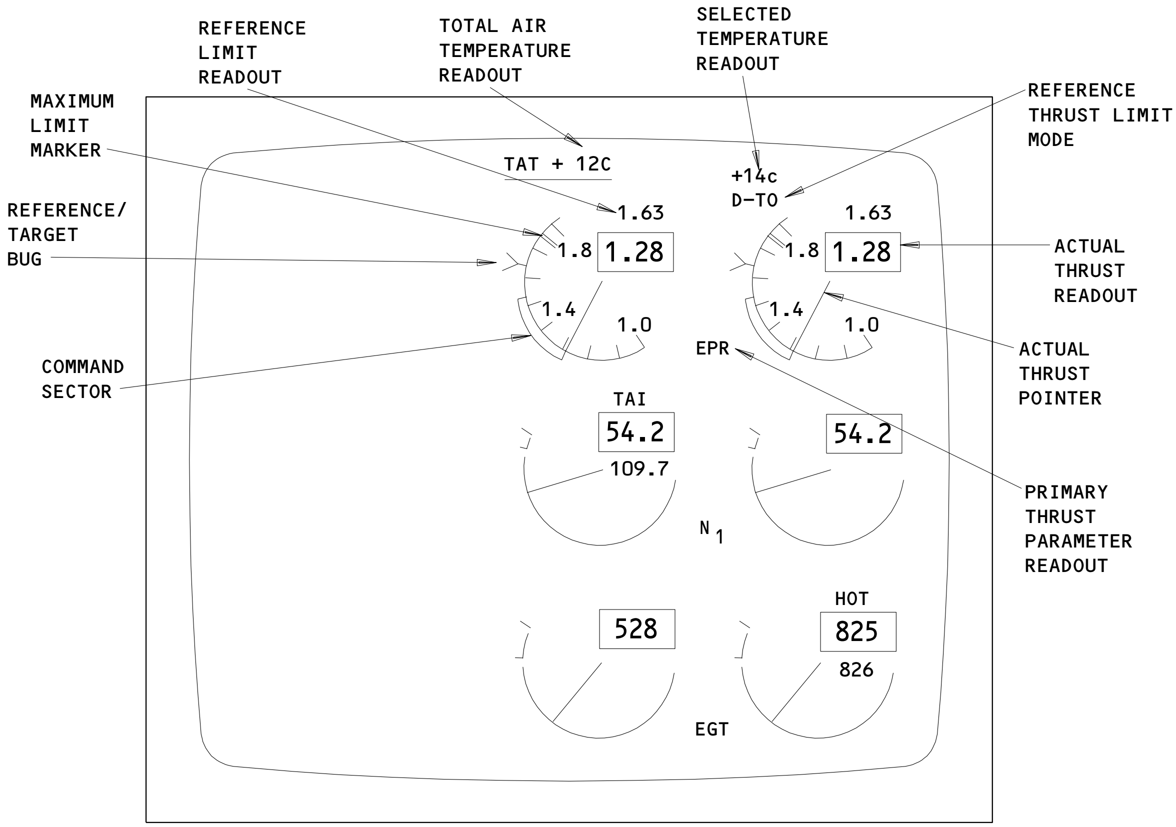
Mode Control Panel - Autothrottle Functions
Figure 6

2) An airspeed/Mach selector (SEL) switch transfers the speed display between airspeed and Mach. Airplane Mach must be at least 0.400 before transfer to the Mach display can be accomplished. Rotating the IAS/MACH control knob adjusts the speed/Mach display during local command.

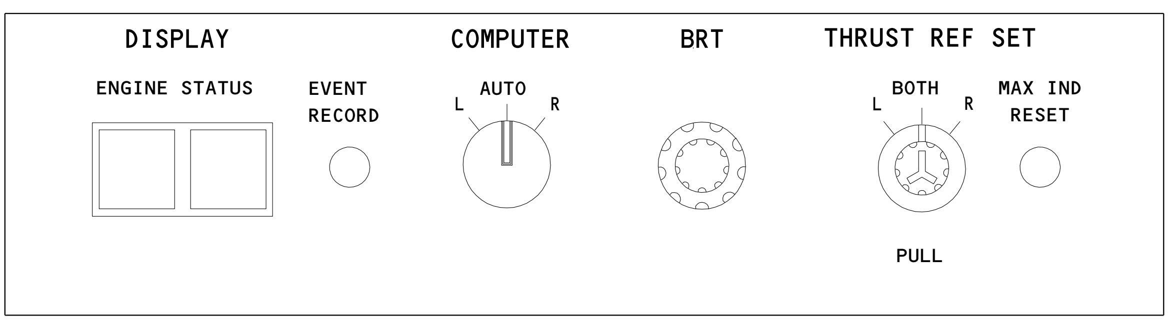
(3) Electronic Attitude Director Indicator (EADI) - Autothrottle Displays (Fig. 7)
  (a) Autothrottle modes (two types):
    1) Operating modes: EPR, SPD, FLCH, GA, IDLE, TEST, and THR HOLD.
    2) Limit modes: FLAP LIM, ALPHA and SPD LIM.
(4) EICAS Display Unit - Autothrottle Functions (Fig. 8)
  (a) The upper EICAS display unit provides the primary engine parameters and TMC computed data required to set and monitor thrusts. These parameters are displayed continuously in both analog type round dial and digital readout for the left and right engines.



EADI - Autothrottle Displays
Figure 7

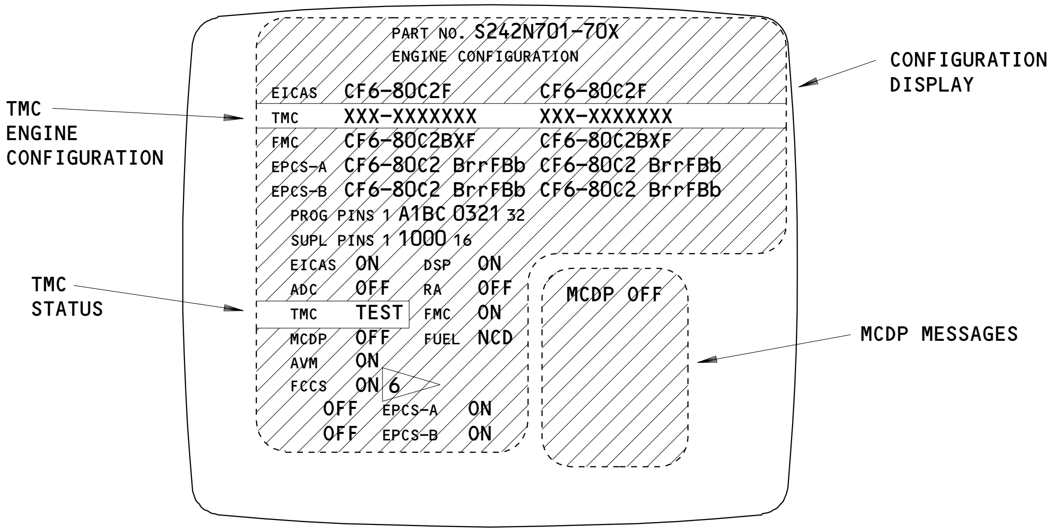


UPPER EICAS DISPLAY UNIT

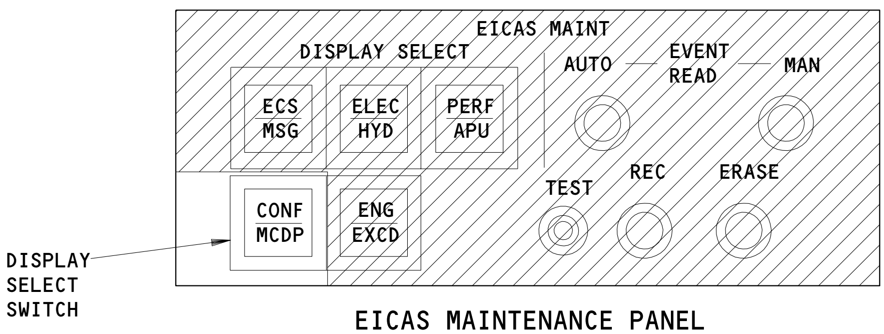


EICAS CONTROL PANEL
(EXAMPLE)

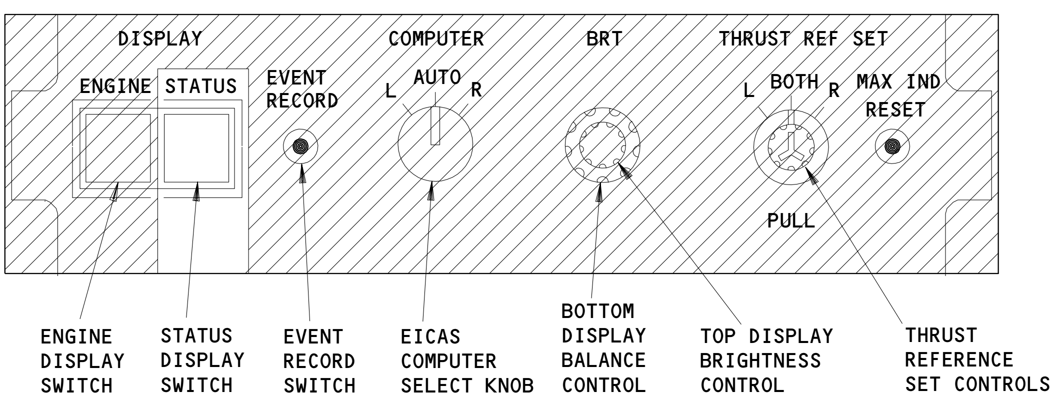
EICAS Display - Autothrottle Functions
Figure 8 (Sheet 1)



EICAS CONF/MCDP PAGE
LOWER EICAS DISPLAY
(EXAMPLE)



EICAS MAINTENANCE PANEL
(EXAMPLE)



EICAS DISPLAY SELECT PANEL
(EXAMPLE)

EICAS Display - Autothrottle Maintenance Information
Figure 8 (Sheet 2)

(b) Actual engine thrust is indicated by both actual thrust pointers and digital readout. Any momentary difference between actual thrust and commanded thrust is shown by the command sector. The maximum thrust limit as computed by the TMC or EEC is shown by an amber marker on the analog scale. The EEC maximum thrust limit data is normally used. If the EEC maximum thrust limit is not available or is not valid, the TMC maximum thrust limit data is used.

(c) Target thrust (reference EPR) is shown by a reference cursor on the analog scale and a digital readout above the actual thrust readout. The thrust limit mode and assumed temperature (if selected) is shown above the EPR indicators. When an assumed temperature is selected, a D appears in front of the thrust limit mode. The display of a D is only valid when the Takeoff Thrust Limit is active. The selection of the Assumed Temperature is allowed only when the airplane is on the ground, Mach is less than .12 and the Autothrottle is not engaged.

(d) Digital displays are also provided for total air temperature (TAT) and thrust reverser status. If a thrust reverser is fully deployed, the readout (REV) is green. The readout is yellow while the reverser is in transit. For normal in-flight conditions, the readout is blank.

(e) The manual THRUST REF SET control on the EICAS control panel consists of a dual rotary switch combined with a push-pull switch. The inner control is a continuously rotating control with 12 detents per revolution. When initially set (pulled out), the readout corresponding to the engine selector switch (L, R, or BOTH) defaults to 1.55 EPR. Clockwise rotation of the control increases the readout and counterclockwise rotation decreases the readout. The rotation sensitivity is 0.01 EPR per detent. Once a readout is modified manually its last input data is retained, regardless of engine selector switch position, until the inner control is set (pushed in). The outer control is a three-position rotary switch that selects the L (left), BOTH, or R (right) EPR readout.

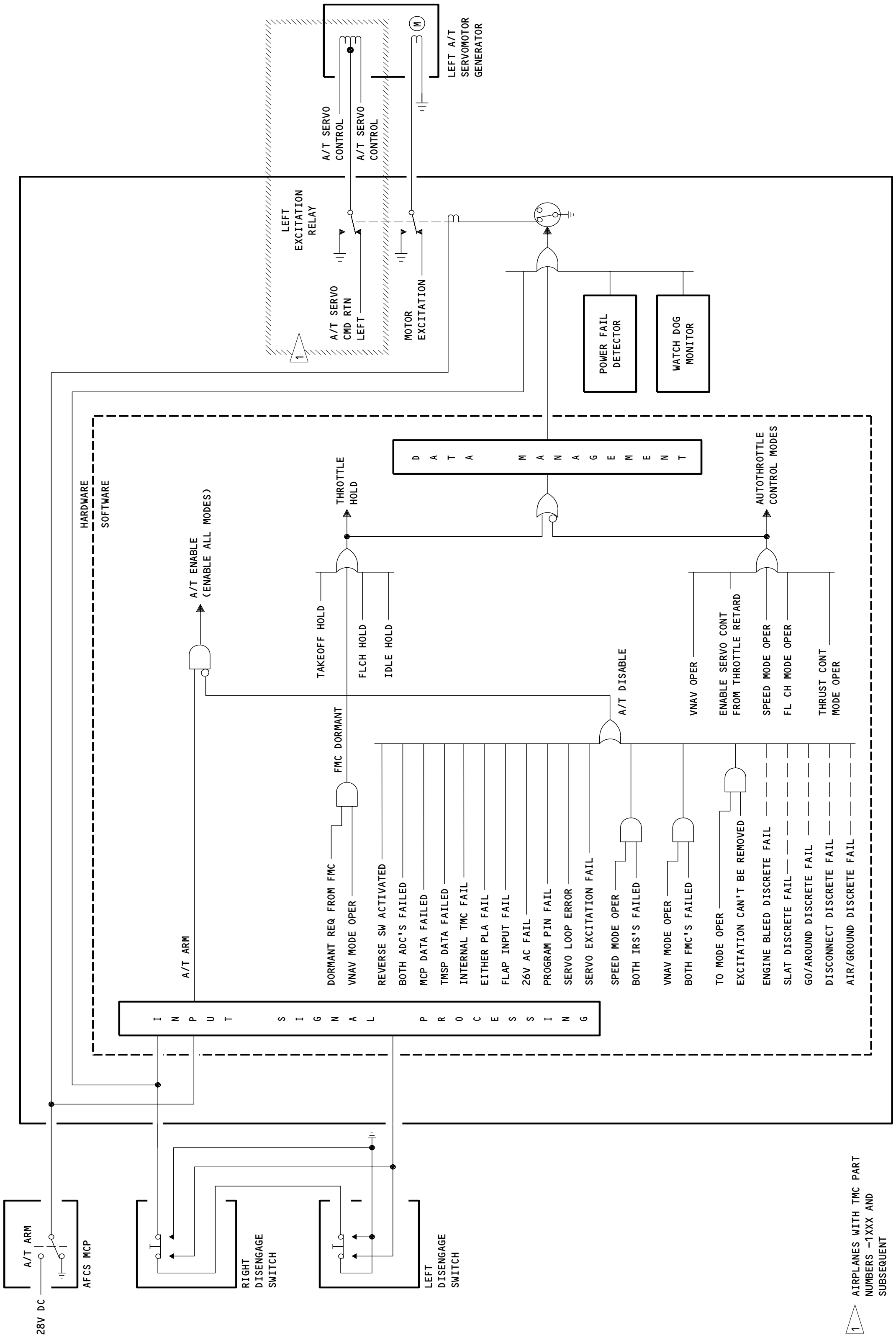
(5) EICAS Display Unit - Maintenance Displays (fig. 8).

(a) The lower EICAS display unit shows the TMC engine configuration and status information. The information is displayed on the EICAS CONF/MCDP page.

(6) Thrust Management System - Engage/Disengage Logic (Fig. 9)

(a) The engage-disengage logic consists of both engage and mode logic. Major input signal sources supplying information for engage/disengage control laws are the FMC, MCP, EECs, and ADCs. All inputs are used to control the motor excitation relay, which connects excitation power to the A/T servomotor generator.

(b) Input signals converted by software include digital and analog discretes and digital variable signals. Analog discrete signals used for logic functions are sent to A/D converters before software usage.



Thrust Management System - Engage/Disengage Logic
Figure 9

(c) Analog discrete inputs:
1) A/T disconnect
2) Air/ground
3) Autothrottle arm

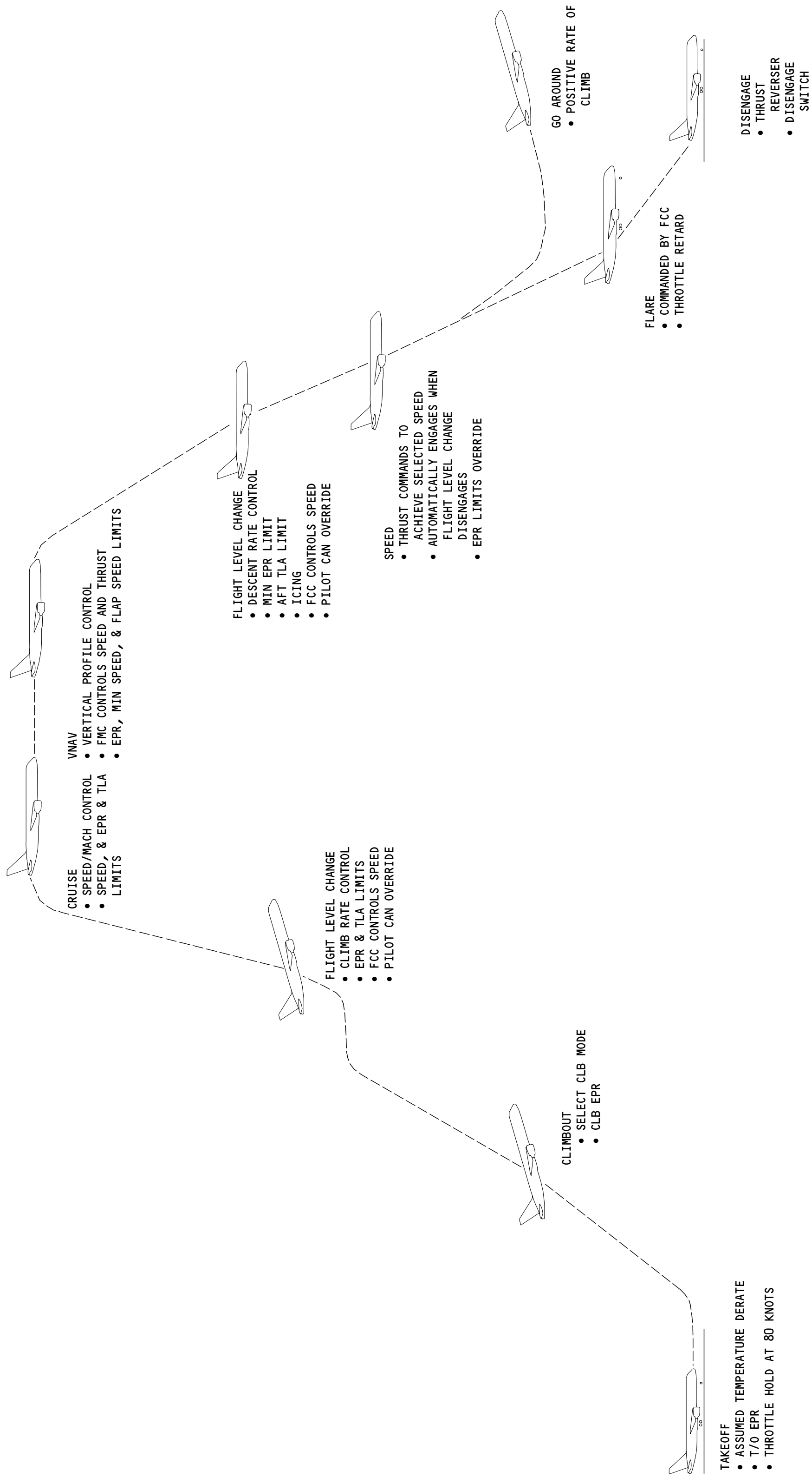
(d) Digital variable inputs (from ADCs):
1) Calibrated airspeed
2) Filtered standard barometric altitude

(e) Digital discrete inputs:
1) MCP Inputs: VNAV mode operational, throttle retard, and thrust mode request
2) EEC Inputs: EEC off and EEC valid
3) FMC Inputs: A/T dormant, thrust request, and idle request

(f) When the A/T switch is in the ARM position, the TMC is enabled to engage the autothrottle servo. The servo controls thrust according to manually or automatically selected operating modes. Detected faults affecting autothrottle operation in the selected mode results in servo disengagement and annunciation on the EICAS display unit. Manual engagement of EPR/N1 or SPD modes is obtained by pressing the THR/EPR/N1 switch or SPD switch located on the MCP. Automatic engagement occurs when FLCH or VNAV mode is selected and engaged in the AFDS. When the autothrottle is engaged, the operating mode is displayed on the EADI. Disengagement of the autothrottle occurs if:
1) A fault affecting autothrottle operation in the selected mode is detected.
2) An autothrottle disengage switch on the thrust levers is pressed.
3) The A/T ARM switch is positioned to OFF.
4) The reverse thrust levers are set in the reverse thrust position.

(7) Typical Autothrottle Flight Profile (Fig. 10)

(a) Takeoff (T/O)
1) When T/O is selected on the TMSP, a reference EPR is calculated based on temperature and altitude. A provision to insert artifically high temperatures for derated takeoff is included. When the EPR mode on the MCP is engaged, the thrust levers are driven until the reference EPR is reached. The T/O throttle hold (THR HOLD) mode deactivates the autothrottle servo by removing 115v ac excitation and zeroing the command when the airplane reaches 80 knots CAS. This prevents the thrust levers from retarding after T/O thrust is set.
2) When the airplane is more than 400 feet above ground level and the climb mode is selected on the TMSP, a new EPR REF is calculated by the TMC. The engines are retarded to the new EPR REF.



Typical Autothrottle Flight Profile
Figure 10

(b) Flight Level Change (FLCH)
1) In the flight level change (FLCH) mode, the TMC controls the thrust to attain a vertical speed so that the transition to a new altitude takes about 2 minutes, subject to the EPR REF for forward direction and the aft limit for the retard direction. When the throttles are driven to the aft throttle stop or manually overridden by the crew, the autothrottle mode transitions to 'THR HOLD'. Manual override of the thrust levers by the pilot causes the autothrottle servo to hold until a new mode is selected, capture of the selected altitude is achieved, or the EPR REF is exceeded.

(c) Cruise (CRZ)
1) When CRZ is selected on the TMSP, the thrust levers control to the speed/Mach reference on the MCP. The maximum/minimum speeds, EPR, and TLA limits are still observed.

(d) Vertical Navigation (VNAV)
1) The VNAV mode transfers both pitch and thrust control to the Flight Management Computer System (FMCS). The FMCS selects autothrottle modes (airspeed, Mach, and thrust) and provides the corresponding target reference values. The FMCS also commands throttle hold and retard to idle. All commands from the FMCS are subject to the thrust and speed limits computed by the TMS.
2) Selecting FLCH causes thrust lever retard to the computed aft limit until the selected altitude is captured. Manual override of the thrust levers by a pilot causes autothrottle servo hold.

(e) Speed (SPD)
1) The speed mode is selected manually by pressing the SPD pushbutton on the MCP. The autothrottle engages to control airspeed (or Mach) to the reference command on the IAS/MACH display on the MCP. The speed mode is automatically engaged when the FLCH mode is exited.

(f) Go-Around (G/A)
1) The thrust levers are automatically advanced for a positive rate-of-climb when a go-around switch on the thrust levers is pressed.

(g) Flare
1) During an autopilot multi-channel approach, the thrust levers retard to the aft limit on command from the FCCs.

(8) Thrust Management System - Mode Select Logic
The input signals shown are used for both mode and annunciation logic. With the autothrottle switch in the ARM position, selection of alternate modes is accomplished manually by actuating the proper controls, or automatically if proper conditions are met. The annunciation logic provides output signals for display on the EADI.
(a) Digital data bus inputs:
1) Autothrottle inputs from the MCP (discrete)

2) Control inputs from the FMCs: thrust mode valid, thrust mode REQ, speed mode REQ, and Mach mode REQ are discrete signals; Mach selected and airspeed selected are variable signals.
3) Environmental data from the ADCs (variable)

(b) Analog inputs are:
1) Flap position from the flap/stab position module

(c) Analog discrete inputs:
1) Autothrottle arm from the MCP.
2) Ground Test from the MCDP.
3) Go-around engage from the go-around switch.
4) Air/Ground discrete from the air/ground relay.

(d) Logic Modes
1) The autothrottle has three basic modes of operation: speed, thrust, and vertical speed. The speed (SPD) mode allows the autothrottle to control either airspeed or Mach depending on the control parameter selection. The thrust control mode provides control to a target EPR command, EPR reference limit, or MAX or MIN EPR aft limit depending on the given flight condition. The Flight Level Change (FLCH) mode provides automatic control of thrust to the EPR REF for climb and to the computed aft limit for descent. The Vertical Navigation (VNAV) mode transfers control of pitch and thrust to the FMCS. When the autothrottle is engaged, the operating mode is annunciated on the EADI.

(e) Output signals to the EADI (for annunciation) are digital discretes.
1) EPR, IAS, and MACH mode
2) THR-HOLD, IDLE, and TEST mode
3) FLAP LIM, ALPHA, IAS LIM, and MACH LIM mode
4) FLCH and GA mode

NOTE: IAS and MACH are annunciated as 'SPD'.

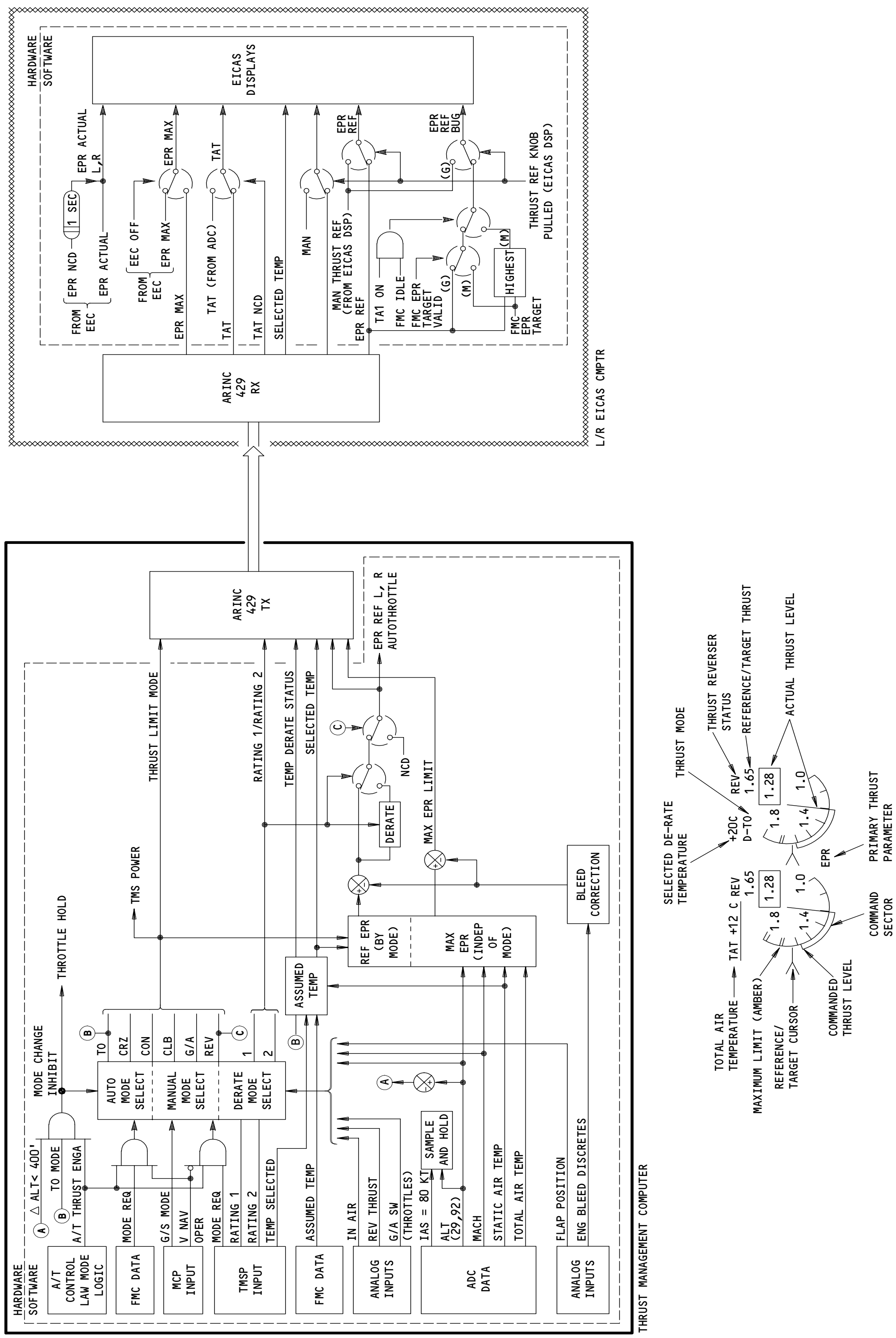
(9) Thrust Management System - Thrust Limit Mode Logic (Fig. 11)

(a) The thrust limit mode logic sets the mode for thrust limit calculations. Thrust modes are selected by TMSP switch engagement, FMC request when the VNAV mode is engaged, or upon power-up based on the flight condition.

(b) Selection of any thrust limit mode disengages the mode that is in operation unless the conditions that follow occur:
1) Reverse thrust or the landing flap is selected, or
2) the glideslope is active from an Autopilot or a Flight Director, and
3) the T/O thrust limit mode is not selected.

(c) The thrust limit mode that is engaged continuously shows on the upper EICAS display unit.

(d) When VNAV is engaged with the Autothrottle on, the TMSP switches are locked out unless the VNAV mode override conditions occur.



TMC Thrust Limit Schematic
Figure 11

202616

(e) T/O Mode
  1) The T/O thrust limit mode is entered four ways as follow:
    a) A TO/GA request from the TMSP when the airplane is on ground. This can only be used if the reverse thrust modes are not engaged.
    b) Automatically when flap position is less than 23 degrees and Mach is less than 0.12. This logics sets the T/O thrust limit mode after a landing without a use of the reverse thrust.
    c) Automatically upon power-up on ground unless reverse thrust is engaged.
    d) Automatically when reverse thrust is switched from on to off while the airplane is on the ground.

(f) G/A Mode
  1) The G/A thrust limit mode is entered six ways as follow:
    a) A manual TO/GA request from the TMSP while in air and the VNAV mode is disengaged.
    b) An FMC G/A request when VNAV is engaged.
    c) Automatically at the TMC power-up initialization in air and the airplane altitude is less than 18,000 ft and Mach is less than 0.46 to 0.55 depending on the altitude.
    d) Automatically when the airplane is in the air, and flap position is greater than 23 degrees or the glideslope is captured, and the T/O thrust limit mode is not selected.
    e) Automatically when flaps transition from 0 to 1.
    f) Automatically when the Autothrottle G/A mode is engaged.

(g) CON Mode
  1) The CON thrust limit mode is entered as follows:
    a) A manual CON request from the TMSP when VNAV and reverse thrust modes are disengaged.
    b) An FMC CON request when VNAV is engaged.
    c) Automatically at the TMC power-up initialization in air and the airplane altitude is greater than 18,000 ft and Mach is greater than 0.46 to 0.55 depending on the altitude.
    d) A manual CON override request when in VNAV CLB thrust limit mode.
    e) A manual CON override request when in VNAV CRZ thrust limit mode.

(h) CLB Modes
  1) The CLB mode is entered seven ways as follows:
    a) A manual CLB request from the TMSP when VNAV and reverse thrust modes are disengaged. Also, the landing flaps are not set or the glideslope is not active.
    b) A FMC CLB request when VNAV is engaged.
    c) A manual CLB override request when in VNAV CRZ thrust limit mode.

d) A FLCH mode request when in T/O thrust limit mode and above 400 ft AGL and the engine is in operation.
e) A VNAV mode request when in T/O thrust limit mode and above 400 ft AGL and the engine is in operation.
f) An A/T SPD mode request when in T/O thrust limit mode and above 400 ft AGL and the engine is in operation.
g) An A/T EPR mode request when in T/O thrust limit mode and above 400 ft AGL and the engine is in operation.

(i) CRZ Mode
1) The CRZ mode is entered two ways as follow:
a) A manual CRZ request from the TMSP when VNAV and reverse thrust are disengaged and the aircraft is above 400 ft AGL. Also, the landing flaps are not set and glideslope is not active from an Autopilot or a Flight Director.
b) A FMC CRZ request when VNAV is engaged.

(j) REV Mode
1) The REV mode is engaged when the airplane is on the ground and the reverse thrust levers are moved to the reverse thrust position.

(10) Thrust Limit Derate Logic
(a) Fixed Percentage Derates are available for the T/O and CLB modes. If the 1 or 2 on the TMSP is pushed when in T/O or CLB mode, the derate will activate and the mode will show on the EICAS in green. When not in the T/O or CLB mode and you push 1 or 2 on the TMSP, a CLB derate is preselected and the mode will show in white on the EICAS.
(b) When the airplane is on the ground and in the T/O mode, the TMC allows the assumed temperature entry from either the TMSP or the FMC CDU for a reduced T/O thrust. The TMC normally computes a thrust limit based on ambient temperature or flat rated temperature, whichever is greater. This value is used as the computation temperature. A further manual increase in computation temperature is referred to as assumed temperature and decrease thrust.
(c) Assumed temperature data is sent to the TMC for thrust limit calculation. If the assumed temperature is higher than flat rated and ambient temperature, the derated mode is displayed by a "D" adjacent to the T/O mode display on the upper EICAS display unit. Assumed temperature is set as follows:
1) Engage in a different limit mode to leave the T/O mode.
2) Push the GA switch on the throttle levers after lift off.
3) Cycle the TMC power.
4) Push the TO/GA button on the TMSP before lift off.
5) Push the 1 or 2 button on the TMSP.
(d) When the CLB mode is engaged, the CLB derated thrust can be selected on the TMSP. The 1 or 2 button on the TMSP gives a fixed decrease in thrust.
(e) When a CLB derate is preselected and you engage the CLB thrust limit mode, it will gives the CLB derated thrust.

(11) Thrust Limit Calculation
   (a) The TMC calculates EPR REF and MAX EPR based on:
      1) Selected mode (affects EPR REF only)
      2) Environmental data
      3) Engine bleed status
   (b) The TMC performs these calculations regardless of A/T engagement status. Calculated limits are displayed on the upper EICAS display unit.
   (c) However, the TMC calculated MAX EPR limit is only displayed if the corresponding EPR MAX limit from the EEC is not available.
   (d) EPR Calculation
      1) Two limit calculations are performed, MAX EPR and EPR REF. The REF calculations are based on thrust limit mode, Mach, temperature, altitude and bleed. The MAX calculations are based on Mach, temperature, altitude, and bleed only. The temperature used for T/O thrust limit calculation is the highest value of flat rated, ambient, or assumed temperature. The temperature used for all the other thrust limit modes is the Total Air Temperature (TAT). Standard barometric altitude and Mach are used for altitude correction.
   (e) Bleed Correction
      1) If bleed air is being extracted, EPR limits are reduced. Discrete inputs select the amount of correction needed. No correction for cowl anti-ice is applied to the thrust limits if the altitude is less than 8000 feet and the static temperature is less than 10.0°C for T/O and G/A modes. If cowl anti-icing is operating under these conditions, the EPR correction is latched until an altitude of 14,000 feet or a speed of 0.45 Mach is obtained. Once the cowl anti-ice is set in an envelope above 14,000 feet and Mach is greater than 0.45, the anti-ice debit stays ON as long as the bleed is ON and either T/O or G/A mode is ON.
   (f) MAX EPR
      1) The MAX EPR limit is calculated, stored, and output to EICAS. The calculated value is displayed on the EICAS display unit when the MAX EPR from EEC is either unavailable or NCD SSM.

(g) REF EPR

1) The REF EPR limit is calculated and stored on the computer pass opposite of the MAX EPR pass. When the T/O mode is engaged, the MAX select function chooses the highest value for REF EPR between REF calculated and MAX calculated minus 0.1. Thrust limit calculations are displayed on the EICAS display unit, and the EPR indicator bug is driven to the REF EPR. If the reverse mode switch is engaged, the REF EPR goes to No Computed Data (NCD) and the EICAS display is blank. During VNAV operation, the EPR indicator is driven to the FMC target EPR (except during A/T Speed or Mach control) and the color is magenta.

(12) Autothrottle - Takeoff Mode (Fig. 12)

(a) Upon T/O engagement, the EPR REF is computed by the thrust limit function of the TMC. Assumed-temperature derating of a takeoff thrust limit is achieved by inserting a temperature that is higher than both ambient and flat rated temperature. This is used as a reference in computing the T/O thrust limit. The assumed temperature is inserted to the TMC either from the TMSP or from the FMC CDU. The T/O mode deactivates the autothrottle servo when 80 knots CAS is obtained. This prevents the thrust levers from retarding after T/O thrust has been set. The EADI displays EPR mode and the THR switch/light is illuminated on the MCP. The REF bug on the EPR indicator is driven by the TMC to the EPR REF value. When 80 knots CAS is reached, THR HOLD is displayed on the EADI. The flap placard protection is provided above 400 feet AGL without transitioning from TO to CLB thrust limits. When controlling to the flap protection speed,the A/T mode display changes to 'FLAP LIM' from 'THR HOLD' on the EADI.

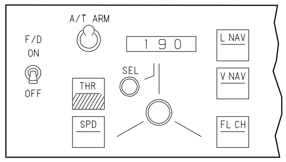
(13) Autothrottle - EPR Mode (Excluding Takeoff) (Fig. 13)

(14) Autothrottle - THR Mode (Excluding Takeoff) (Fig. 13)

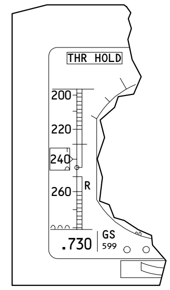
(a) The A/T may be engaged in the EPR mode to control to the selected EPR REF limit (CRZ, CLB, CON, or GA) subject to flap placard protection. The thrust limit mode may be changed as required, except when flaps are in the landing position at which time GA TLC mode is locked in, or when the altitude is less than 400 feet AGL in A/T takeoff mode.

(15) Autothrottle - VNAV Mode (Fig. 14)

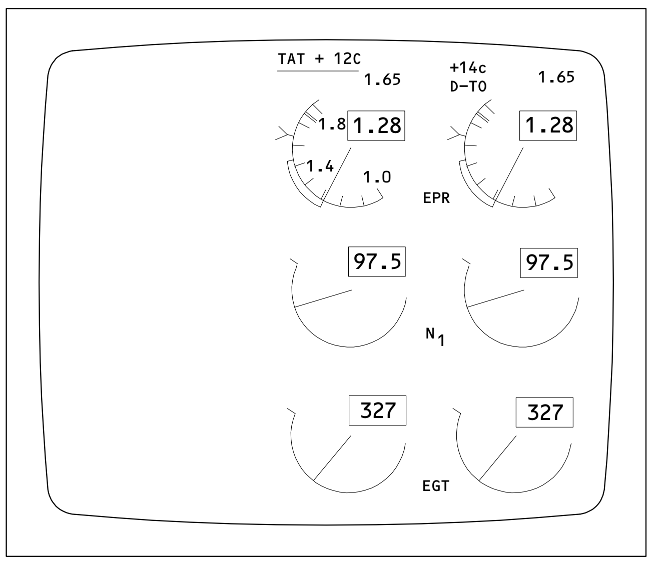
(a) The VNAV mode engages when the VNAV switch is pressed and the FCC accepts the VNAV request. The MCP VNAV switch/light dot-bar matrix then illuminates.



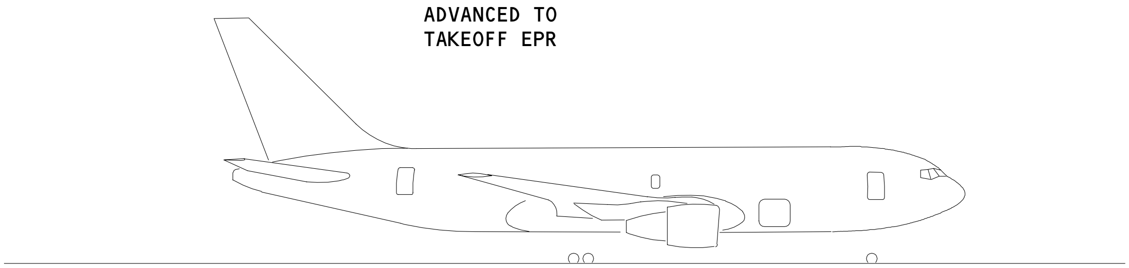
MODE CONTROL PANEL
- A/T ARM
- EPR SELECTED ON THE GROUND WITH T/O THRUST LIMIT MODE



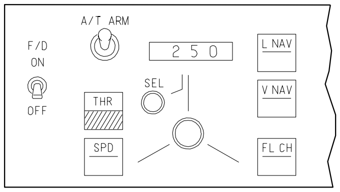
ATTITUDE DIRECTOR INDICATOR
- THROTTLE HOLD AT 80 KNOTS



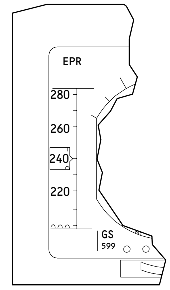
UPPER EICAS DISPLAY
- DERATED TAKEOFF MODE
- REFERENCE EPR DISPLAYED
- ASSUMED TEMPERATURE DERATE
- REF BUG DRIVEN BY TMC
- THROTTLE ADVANCED TO TAKEOFF EPR



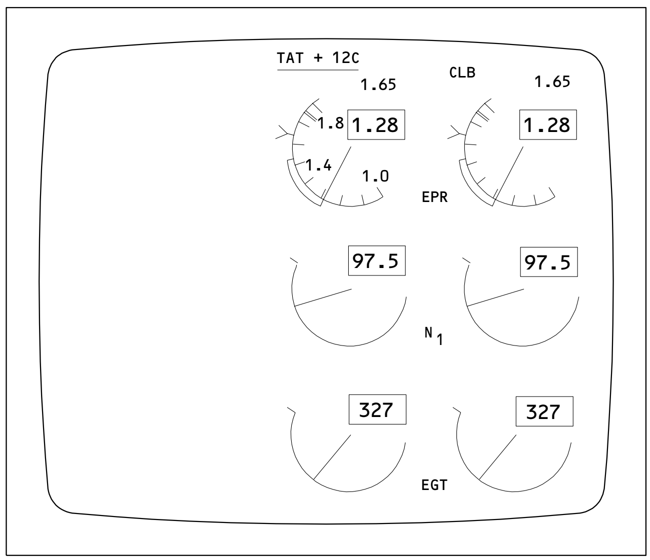
Autothrottle - Takeoff Mode
Figure 12



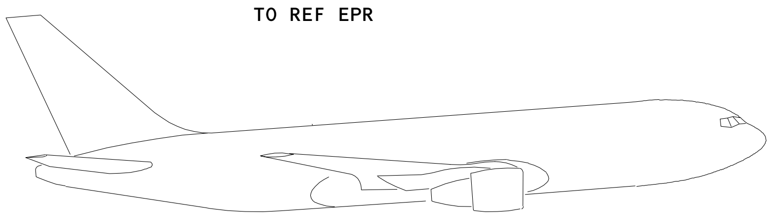
MODE CONTROL PANEL
- A/T ARM ON
- THR SELECTED



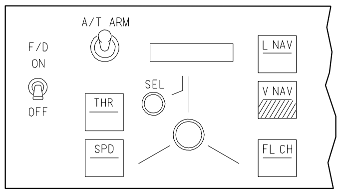
ATTITUDE DIRECTOR INDICATION
- EPR MODE DISPLAYED



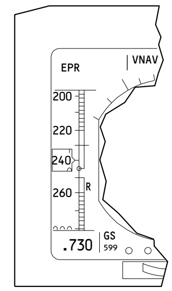
UPPER EICAS DISPLAY
- CLB MODE SELECTED
- REF EPR DISPLAYED
- MAX EPR DISPLAYED
- REF BUG DRIVEN BY TMC (GREEN)
- THROTTLES CONTROL
  TO REF EPR



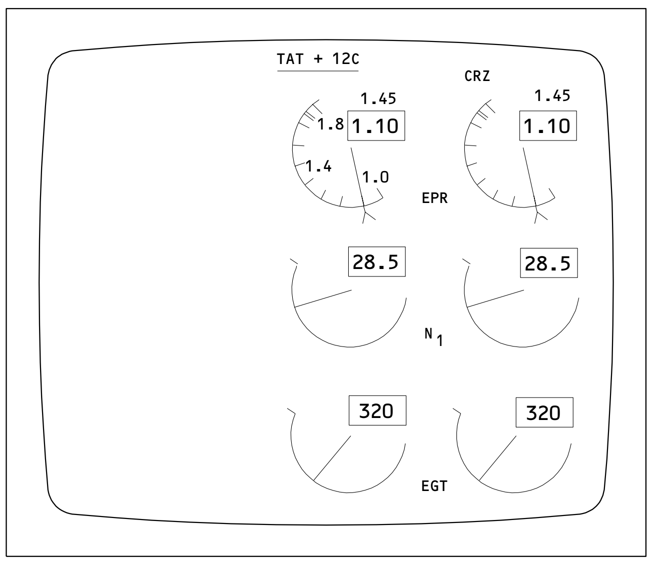
Autothrottle - EPR Mode
Figure 13



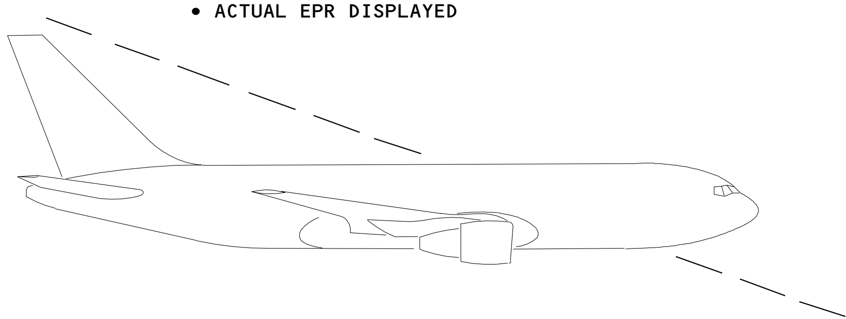
MODE CONTROL PANEL
- A/T ARM
- VNAV SELECTED
- SPEED BLANK



ATTITUDE DIRECTOR INDICATION
- EPR MODE DISPLAYED
  (FMC REQUESTED)



UPPER EICAS DISPLAY
- CRZ SELECTED
- REF EPR DISPLAYED
- MAX EPR DISPLAYED
- REF BUG CONTROLLED BY FMC (MAGENTA)
- ACTUAL EPR DISPLAYED



Autothrottle - VNAV Mode
Figure 14

(b) The VNAV mode transfers control of pitch and thrust to the FMC. The TMC receives the VNAV mode engage status from the FCC. When VNAV is engaged, the TMC accepts mode requests along with target commands (EPR, Mach and airspeed) from the FMC. The FMC request modes are EPR, SPD, mach, idle, or dormant. The THR and SPD pushbuttons on the MCP do not illuminate during VNAV, and the IAS/MACH display is blank. The EICAS display unit displays the TMC computed EPR REF. The EPR indicator bug on the EICAS display is driven to the FMC generated target EPR and colored magenta if the A/T is not engaged and controlling speed (Mach). When A/T is engaged in VNAV SPD mode, the bug is green at EPR REF. During VNAV DES mode with A/T engaged, theFMC controls thebug to a target EPR and the color is magenta. During VNAV operation, the thrust levers are automatically driven to the FMC generated target EPR, Mach or airspeed. The active autothrottle mode status is displayed on the EADI.

(16) Autothrottle - Flight Level Change Mode (Fig. 15)

(a) The Flight Level Change (FLCH) mode is engaged when the A/T switch is in the ARM position and the MCP accepts the FLCH mode request (in-air only).

1) In the flight level change (FLCH) mode, the TMC controls the thrust to attain a vertical speed so that the transition to a new altitude takes about 2 minutes, subject to the EPR REF for forward direction and the aft limit for the retard direction. When the throttles are driven to the aft throttle stop or manually overridden by the crew, the autothrottle mode transitions to 'THR HOLD'.

(b) When the FLCH mode is operational, annunciation of FLCH appears on the EADI. In addition, the reference bug on the EPR indicator is driven by the TMC to the EPR REF value for the selected EPR thrust limit mode.

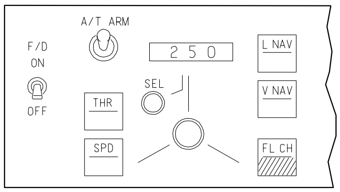
(c) Manual thrust lever movement by the pilot places the autothrottle servo in a hold state until a new mode is selected, the selected altitude is reached, or the EPR REF is exceeded.

(17) Autothrottle - Go-Around Mode (Fig. 16)

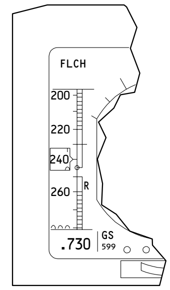
(a) The Go-Around (G/A) mode is automatically armed when the A/T switch is in the ARM position and; the glideslope is captured (from the FCC) or the flap position is greater than zero. Actual go-around engagement occurs when either go-around switch on the thrust levers is pressed.

(b) When the A/T G/A mode and an autopilot channel are engaged, the thrust levers advance to achieve a 2000 fpm rate of climb as limited by the EPR REF. When the A/T G/A mode is engaged with no autopilot engaged, the thrust levers advance to achieve the EPR REF. When the go around mode is operational, annunciation of GA appears on the EADI. The reference bug on the EPR indicator is driven by the TMC to the EPR REF value.

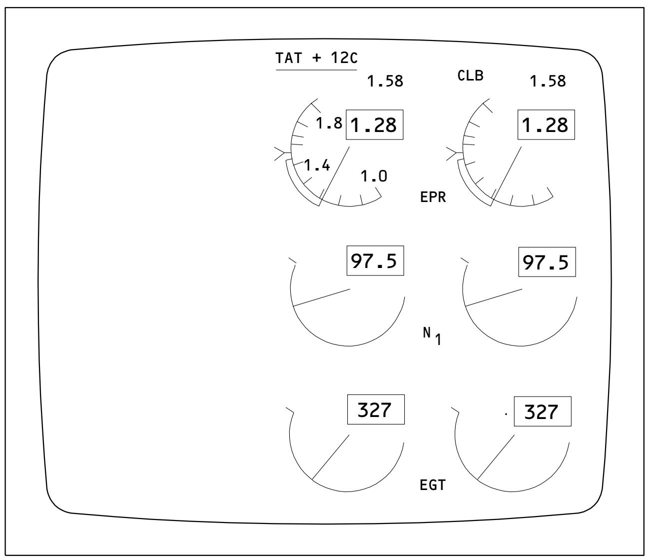
(18) Autothrottle - Flare Retard (Fig. 17)



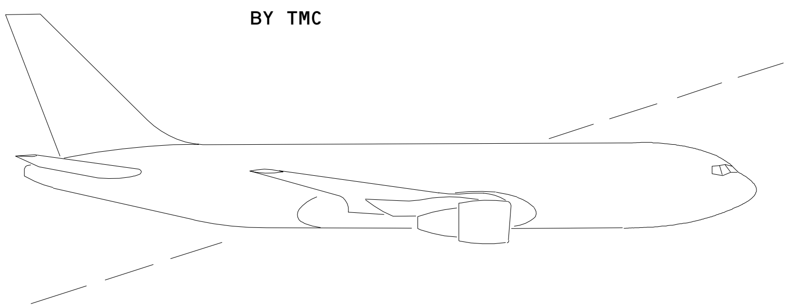
MODE CONTROL PANEL
- A/T ARM
- FL CH SELECTED
- AUTOTHROTTLE SETTING IS A CLIMB OR DESCENT RATE PROPORTIONAL TO ALTITUDE CHANGE
- IAS/MACH IS TARGET FOR FCC PITCH CHANNEL



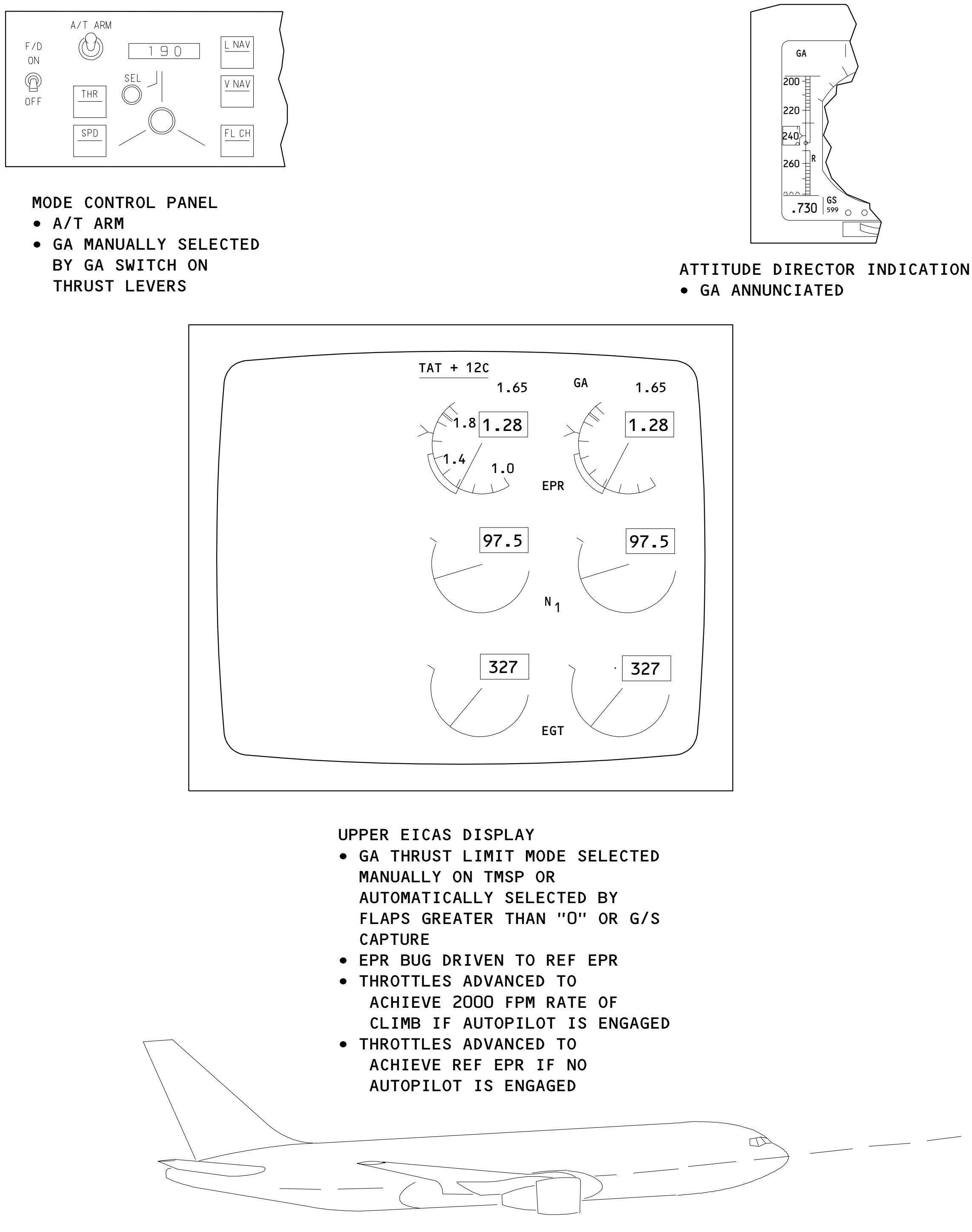
ATTITUDE DIRECTOR INDICATION
- FL CH MODE DISPLAYED



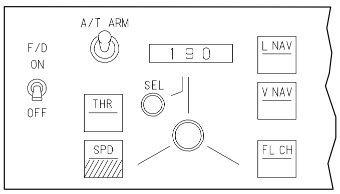
UPPER EICAS DISPLAY
- LIMIT EPR'S DISPLAYED
- REF BUG DRIVEN BY TMC



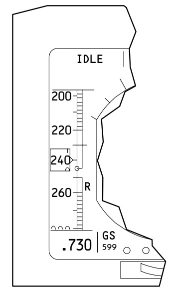
Autothrottle - FL CH Mode
Figure 15



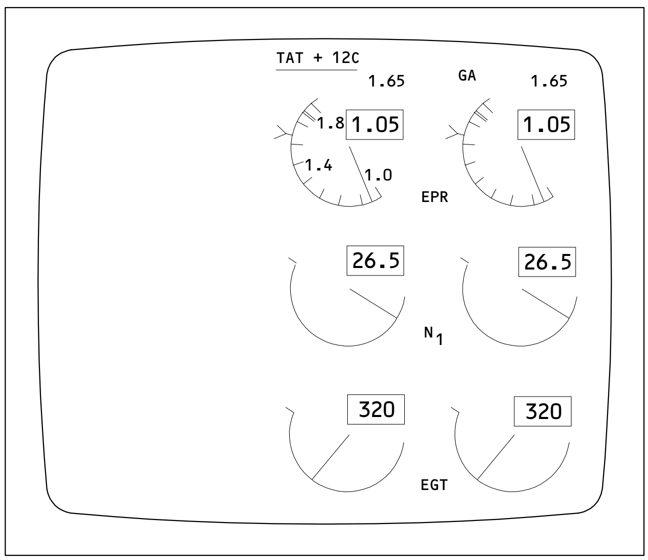
Autothrottle - Go-Around Mode
Figure 16



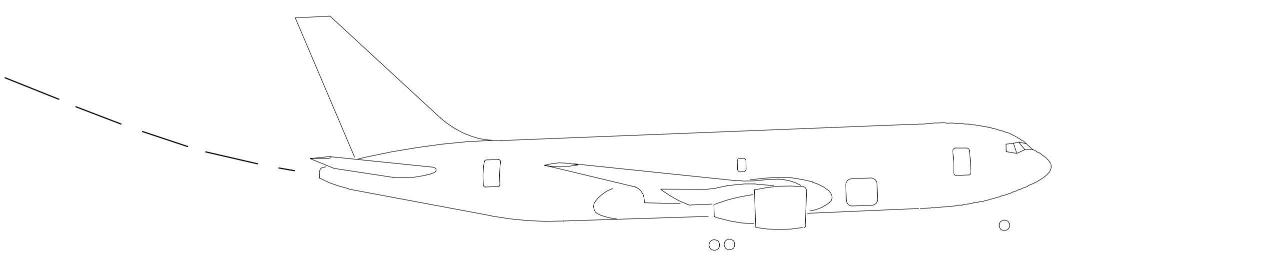
MODE CONTROL PANEL
- A/T ARM
- IN SPEED MODE, FLARE IS COMMANDED BY FCC AT 50 FEET RADIO ALTITUDE



ATTITUDE DIRECTOR INDICATION
- IDLE MODE SHOWN



UPPER EICAS DISPLAY
- GA MODE AUTOMATICALLY ARMED BY A FLAP POSITION GREATER THAN 0 OR G/S CAPTURE
- REF BUG DRIVEN BY TMC
- THROTTLES CONTROL TO AFT LIMIT AT FLARE



Autothrottle - Flare Retard
Figure 17

(a) Autothrottle flare retard occurs upon request from the FCCs when two or more A/P channels are engaged. The autothrottle must be engaged in the SPD mode.

(b) When flare retard is requested, the thrust levers retard to the flare idle limit at a controlled rate. For the autoland flare retard mode, the TLA is commanded to 2 degrees below the computed aft limit.

(19) Autothrottle - Idle Mode (Fig. 18)

(a) Autothrottle idle occurs when requested by the flight management computer. The autothrottle must be armed and engaged in the VNAV mode to accept this request.

(b) When the idle mode is engaged, the thrust levers retard to ground or flight idle limits at a controlled rate. While the thrust levers are moving to the idle position, IDLE is annunciated on the EADI. When the thrust levers reach the idle position, throttle hold is engaged. This removes autothrottle servomotor generator excitation and THR-HLD is annunciated on the EADI.

(20) Thrust Management System - EPR FLCH Mode Signals and Servo Drive

(a) Thrust lever position is determined in three steps:

1) The TMC generates a desired throttle rate command signal in order to reach the EPR target.

2) The TMC internal automatic limits override the rate command signal to prevent exceeding EPR limits.

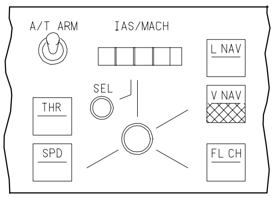
3) The rate command signal drives the A/T servomotor generator at a desired rate. The signal is limited by a scheduled rate limit which does not allow the command to exceed the performance of the engine.

(b) EPR Control in T/O or VNAV Mode

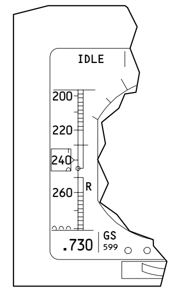
1) An ENABLE signal (SERVO CONTROL FROM THRUST) allows selection of EPR error when the T/O or VNAV mode is engaged. In the T/O mode, the difference between the engine actual EPR and the EPR reference is used for the EPR error. When the VNAV mode is engaged and a valid SELECT FMC EPR TARGET CMD signal is received, actual EPR and FMC target EPR determine EPR error.

(c) EPR Control when not in T/O or VNAV Mode

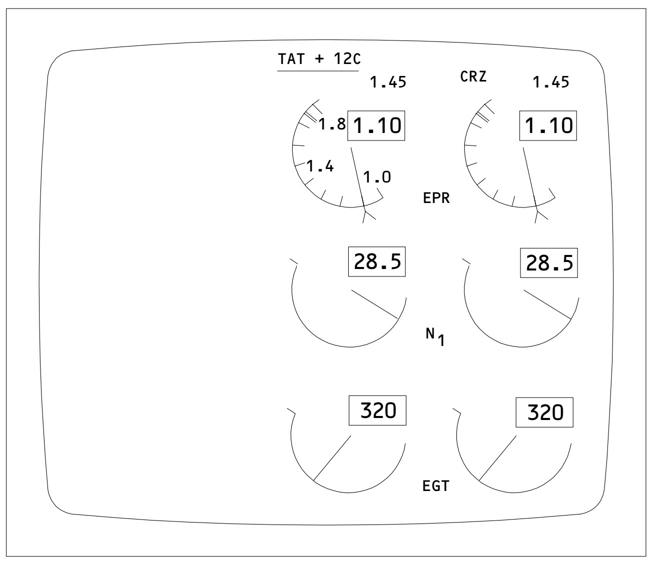
1) When in an EPR mode other than T/O or VNAV mode, the SPEED CONTROL THROTTLE RATE CMD signal provides initial rate. The rate is computed to increase airplane speed to the upper limit. With the flaps retracted, maximum operating airspeed (VMO) or maximum operating Mach (MMO) is taken as the limit. With the flaps extended, flap placard speed is taken as the limit. The airplane speed is at the upper speed limit unless actual EPR reaches the EPR REF limit. In this case, automatic override occurs and the airplane speed is limited to EPR REF.



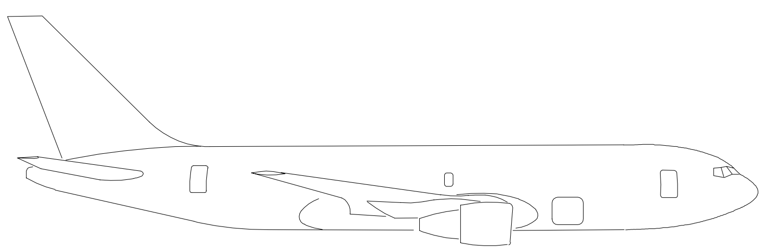
MODE CONTROL PANEL
- A/T ARM
- VNAV SELECTED



ATTITUDE DIRECTOR INDICATION
- IDLE AND THEN THR-HOLD ANNUNCIATED



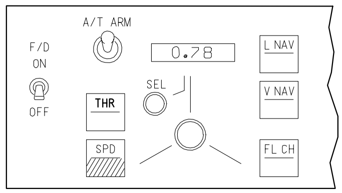
UPPER EICAS DISPLAY
- COULD BE ANY MODE
- FMC CONTROLS REF BUG
- IDLE EPR DISPLAYED



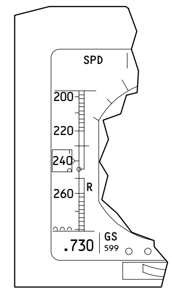
Autothrottle - Idle Mode
Figure 18

(d) MIN and MAX EPR Protection
   1) The maximum EPR protection logic limits the forward motion of the thrust lever to prevent EPR from exceeding the EPR REF value.
(e) Throttle Servo Rate Command Filter
   1) The throttle rate COMMAND is compared to the servo tachometer feedback and the difference is filtered to form the THROTTLE SERVO RATE COMMAND.
(f) Power Amplifier
   1) The THROTTLE SERVO RATE CMD is converted to an analog signal and amplified to drive the variable phase motor in the A/T servomotor generator. The rate command is compared to the tachometer feedback in order to reach the desired rate.
(g) Flight Level Change (FLCH) Control
   1) In the flight level change (FLCH) mode, the TMC controls the thrust to attain a vertical speed such that the transition to a new altitude takes about 2 minutes, subject to calculated minimum and maximum thrust settings. When the throttles are driven to the aft throttle stop, or are manually overridden by the crew, the Autothrottle mode transitions to 'THR HOLD'.
   2) The actual TLA position is compared to the command rate. If a difference is detected, pilot override is assumed to have occurred and FLCH hold is engaged. This removes the rate command and the excitation from the A/T servomotor generator.

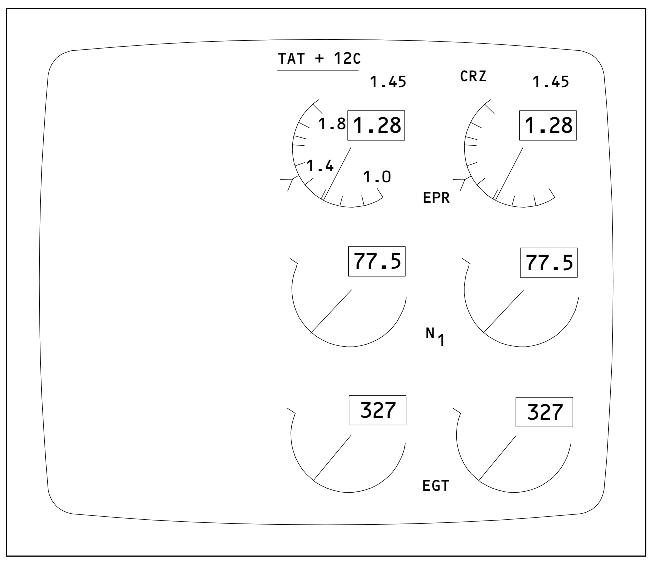
(21) Autothrottle Speed Mode (Fig. 19)
(a) The speed mode is engaged when the A/T arm switch is in the ARM position and the MCP SPD request is valid. Automatic speed mode engage occurs when the VNAV or FLCH mode is exited.
(b) Depressing the SPD pushbutton sends the mode change request to the TMS and AFDS. If the autothrottle is armed and there are no faults affecting operation in the speed mode, the autothrottle engages and controls airspeed (or Mach) to the reference value selected in the IAS/MACH display on the MCP.



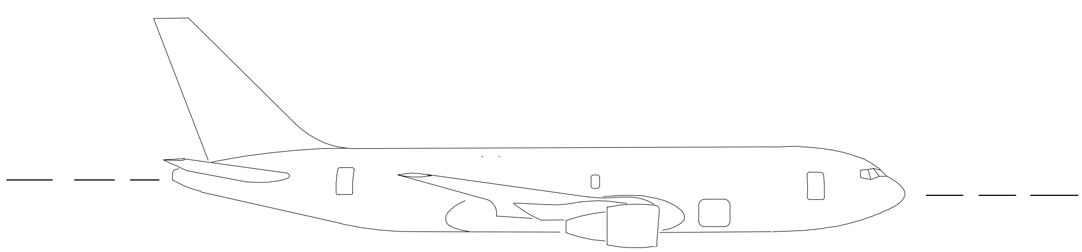
MODE CONTROL PANEL
- A/T ARM
- SPEED SELECTED
- 0.78 MACH SELECTED



ATTITUDE DIRECTOR INDICATION
- SPD MODE DISPLAYED



UPPER EICAS DISPLAY
- CRZ SELECTED
- REF EPR DISPLAYED
- MAX EPR DISPLAYED
- REF BUG DRIVEN TO
  REF EPR
- ACTUAL EPR DISPLAYED



Autothrottle Speed Mode
Figure 19

(c) The MCP IAS/MACH display is controlled by the SEL button and the unlabeled speed select knob located below the display window. The SEL button permits IAS or MACH reference speed to be displayed. Selection of the IAS or MACH reference speed is made by turning the unlabeled speed select knob. Pushing the center of the speed select knob transfers control of the speed reference from the MCP to the FMC. When a TMS control mode of SPD, FLCH or EPR (T/O or G/A) is engaged, pushing the speed select knob does not cause any speed reference change. When in VNAV mode, the IAS/MACH display will be blank when the FMC is in control of the reference and driving the airspeed bugs on the MACH/airspeed indicators. When the speed window is blank, pushing the speed select knob causes the current airspeed to be displayed in the window. If the airspeed reference is invalid, 200 knots is displayed.

(d) The ranges of the speed select knob are:

1) 0.4 to 0.95 for Mach

2) 100 to 399 knots for airspeed

(e) The SEL pushbutton allows selection of Mach or airspeed for display, provided it is within the allowable range.

(22) Thrust Management System - Speed Mode Signal

(a) Speed Mode Functions

1) The A/T COMMAND (SPEED CONTROL THR/RATE CMD) signal commands a throttle rate to capture and maintain the selected speed on the MCP. This command is automatically limited between minimum and maximum speed limits. When the EPR mode is engaged, the thrust levers advance toward the EPR REF but are limited to the flap limit speed or maximum speed (VMO). Normally the EPR REF limit overrides before the maximum speed is obtained.

(b) A/T COMMAND

1) The A/T COMMAND controls the thrust levers to maintain a target speed. The TMS uses True Airspeed (TAS) in controlling to the CAS reference target speed. TAS is computed from the CAS provided by the ADC to the TMC. The commanded true airspeed is limited by the maximum speed limits to prevent the speed from violating MMO, VMO, and flap placard limits.

(c) Minimum Speed
 1) The TMC causes an ALPHA speed message when the airplanes speed is at or below Vref. The TMC calculates Vref as a function of AOA and then compares the calculated Vref with actual speed. This eliminates nuisance ALPHA speed messages when the airplane AOA is high.

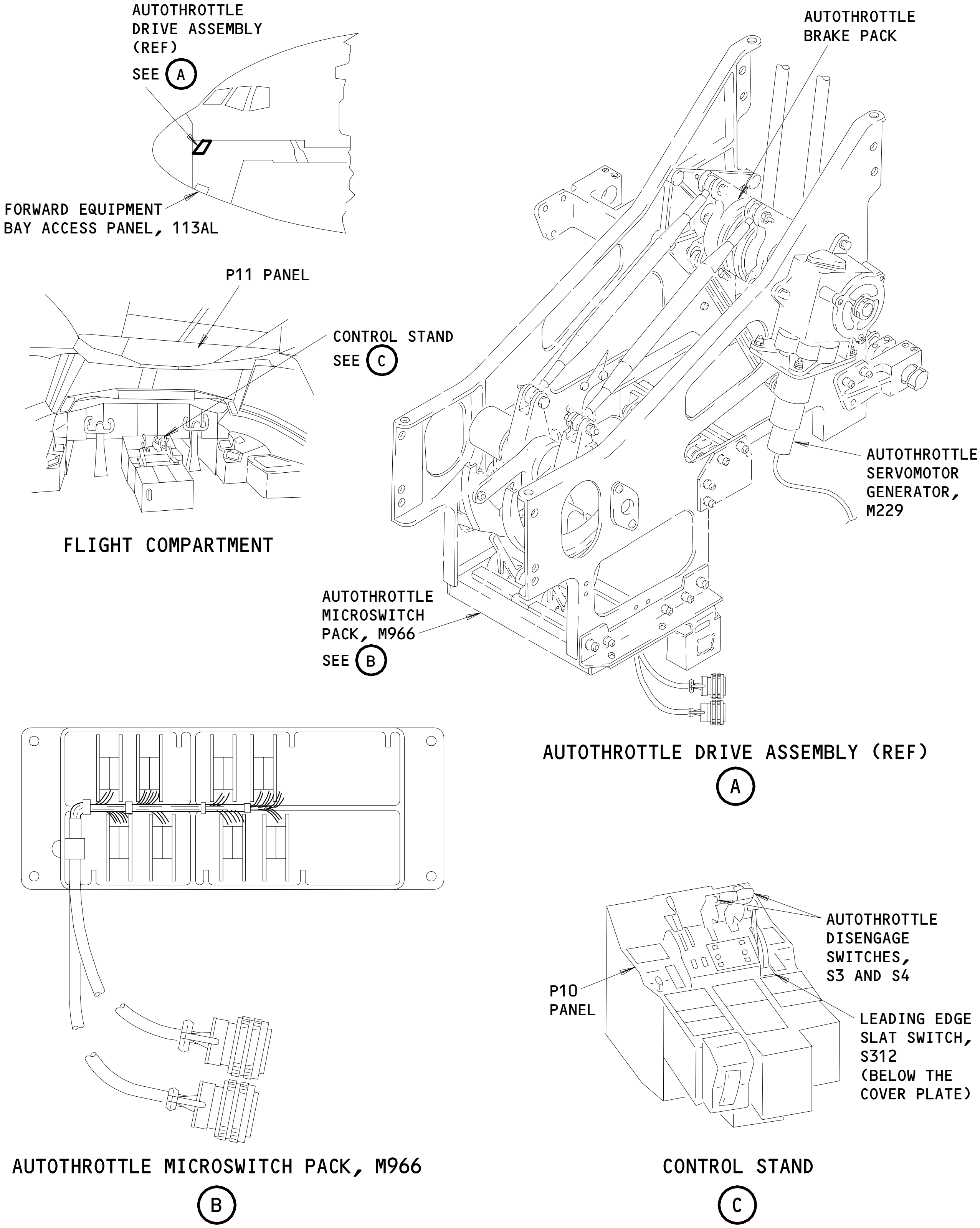
(d) Initial Target for EPR Mode
 1) When the A/T is in the EPR mode and T/O is disengaged, the thrust levers track the new reference EPR. Flap placard and maximum airspeed and Mach protection is provided.

THRUST MANAGEMENT SYSTEM

COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	REFERENCE
CARD - (REF 73-21-00, FIG. 101)				
L ENG EEC DISCRETES, M590				
R ENG EEC DISCRETES, M591				
CIRCUIT BREAKERS	--		FLT COMPT, P11	
AUTO FLIGHT WARN, C521		1	11A17	*
TMC AC, C501		1	11F14	*
TMC DC, C525		1	11F15	*
TMC SERVO, C512		1	11F16	*
COMPUTER - (REF 31-41-00, FIG. 101)				
EICAS L, M10181				
EICAS R, M10182				
COMPUTER - (REF 34-12-00, FIG. 101)				
AIR DATA L, M100				
AIR DATA R, M101				
COMPUTER - (REF 34-61-00, FIG. 101)				
FLIGHT MANAGEMENT L, M134				
FLIGHT MANAGEMENT R, M135				
CONTROL - (REF 73-21-00, FIG. 101)				
ELECTRONIC ENGINE, M541				
GENERATOR - AUTOTHROTTLE SERVOMOTOR, M229	--	1	113AL, FORWARD EQUIPMENT BAY	22-32-01
MODULE - (REF 27-58-00, FIG. 101)				
L FLAP/STAB POS, M838				
PACK - AUTOTHROTTLE BRAKE	--	1	113AL, FORWARD EQUIPMENT BAY	22-32-07
PACK - AUTOTHROTTLE MICROSWITCH, M966	--	1	113AL, FORWARD EQUIPMENT BAY	22-32-04
PANEL - (REF 22-11-00, FIG. 101)				
AFCS MODE CONTROL, M90				
PANEL - (REF 22-41-00, FIG. 101)				
MAINTENANCE CONTROL DISPLAY, M168				
RELAY - (REF 31-01-36, FIG. 101)				
SYSTEM 1 AIR GROUND, K140				
SWITCH - (REF 22-11-00, FIG. 101)				
L GO-AROUND, S1				
R GO-AROUND, S2				
SWITCH - LEADING EDGE SLAT, S312	--	1	FLT COMPT, P10	22-32-03
SWITCH - SYS L AUTOTHROTTLE DISENGAGE, S3	--	1	FLT COMPT, P10	22-32-02
SWITCH - SYS R AUTOTHROTTLE DISENGAGE, S4	--	1	FLT COMPT, P10	22-32-02
UNIT - (REF 34-21-00, FIG. 101)				
L INERTIAL REFERENCE, M159				
R INERTIAL REFERENCE, M161				

* SEE THE WDM EQUIPMENT LIST

Thrust Management System - Component Index
Figure 101



Thrust Management System - Component Location
Figure 102

THRUST MANAGEMENT SYSTEM - ADJUSTMENT/TEST

1. General
   A. This procedure has two tasks - an operational test and a system test for the Thrust Management System (TMS).
      (1) The operational test makes sure that the Thrust Management System operates correctly. The test does not use tools or test equipment.
      (2) The system test uses the MCDP to give a confidence test of the Thrust Management System and its interfaces.

TASK 22-32-00-715-001

2. TMS Operational Test
   A. General
      (1) The TMS operational test uses only Built-In-Test (BIT) and warning functions. The TMS is operational when these conditions are valid:
         (a) Power for normal system operation is supplied to the computer
         (b) No internal computer fault or system monitor invalid is detected
         (c) The input sensor interfaces and signals necessary for mode engage are valid
         (d) The Mode Control Panel autothrottle switch is in the ARM position.
   B. References
      (1) AMM 22-10-00/501, Autopilot (Flight Control)
      (2) AMM 24-22-00/201, Electrical Power - Control
      (3) AMM 31-41-00/501, EICAS
      (4) AMM 31-51-00/501, Warning System
      (5) AMM 34-12-00/501, Air Data Computing System
      (6) AMM 34-21-00/501, Inertial Reference System (IRS)
      (7) AMM 34-61-00/501, Flight Management Computer System
   C. Access
      (1) Location Zones
         119/120 Main Equipment Center
         211/212 Flight Compartment

      (2) Access Panel
         119AL Main Equipment Center

   D. Prepare to Test the Thrust Management System

      S 865-002
      (1) Supply electrical power (AMM 24-22-00/201).

      S 865-111
      (2) Make sure that these circuit breakers on the overhead circuit breaker panel, P11, are closed:
         (a) 11A17, AUTO FLIGHT WARN
         (b) 11F14, TMC AC
         (c) 11F15, TMC DC
         (d) 11F16, TMC SERVO

S 865-112
(3) Make sure that these systems operate:
  (a) Autopilot (Flight Control) (AMM 22-10-00/501)
  (b) Engine Indication and Crew Alerting System (AMM 31-41-00/501)
  (c) Warning System (AMM 31-51-00/501)
  (d) Air Data Computing System (AMM 34-12-00/501)
  (e) Inertial Reference System (AMM 34-21-00/501)
  (f) Flight Management Computer System (AMM 34-61-00/501).

E. Test the Thrust Management System Operation

S 865-113
(1) Set the A/T switch on the AFCS mode control panel (on the P55 panel) to ARM.

S 285-006
(2) Make sure that EICAS shows this information:
  (a) Total air temperature
  (b) The maximum EPR LIMIT and the EPR reference
  (c) TO mode
  (d) The temperature select display is blank.

S 285-007
(3) Make sure that the EPR indicator bug shows the same value as the EPR reference.

S 985-220
(4) Turn the TEMP SEL knob on the TMSP clockwise one detent position.

S 285-014
(5) Make sure that EICAS shows the flat-rated temperature.

S 985-015
(6) Continue to turn the TEMP SEL knob clockwise.

S 285-016
(7) Make sure that the temperature increases, the EPR reference decreases, and D is shown on the front of the mode display.

S 715-122
(8) Make sure that the A/T DISC light operates correctly:
  (a) Make sure that the A/T DISC light on the P1-3 panel is off.
  (b) Push the Master Dim and Test switch on the pilots' overhead panel, P5, and make sure that the A/T DISC light comes on.

F. Put the airplane back to its initial condition if the system test is not necessary.

S 865-123
(1) Set the A/T switch on the AFCS mode control panel to OFF.

S 865-027
(2) Remove electrical power if it is not necessary (AMM 24-22-00/201).

TASK 22-32-00-735-028
3. TMS System Test
A. General
(1) This test makes sure that the Thrust Management System (TMS) operates correctly. This test uses the Maintenance Control and Display Panel (MCDP) ground tests to examine the FMS interfaces.
(2) The MCDP is installed on the E1-2 shelf in the main equipment center. The MCDP control switches and display are on the front of the MCDP. If a remote MCDP control panel is not used, two technicians are necessary for this system test: one in the main equipment center and one in the flight compartment.
(3) Wiring from the MCDP to the flight compartment is installed for a hand-held remote MCDP control panel. The MCDP output is shown on the bottom EICAS display. Additional MCDP instructions are in the Autoflight BITE Maintenance Practices (AMM 22-00-02/201).
B. Equipment
(1) Remote MCDP Control Panel - A22001-22 (recommended); A22001-15 (optional)
C. References
(1) AMM 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
(2) AMM 22-00-02/201, Autoflight BITE
(3) AMM 22-10-00/501, Autopilot (Flight Control)
(4) AMM 22-41-00/501, Maintenance Monitor
(5) AMM 24-22-00/201, Electrical Power - Control
(6) AMM 31-41-00/501, EICAS
(7) AMM 34-12-00/501, Air Data Computing System
(8) AMM 34-21-00/501, Inertial Reference System (IRS)
(9) AMM 34-61-00/501, Flight Management Computer System
D. Access
(1) Location Zones

119/120	Main Equipment Center
211/212	Flight Compartment

(2) Access Panel
119AL Main Equipment Center

E. Prepare to Test the Thrust Management System

S 715-029
(1) Do the TMS Operational Test.

S 745-030
(2) Do the MCDP power-up and self-test (AMM 22-41-00/501).

S 865-124
(3) Make sure that the A/T switch on the AFCS mode control panel (on the P55 panel) is set to ARM.

S 865-125
(4) Make sure that these systems operate:
(a) Autopilot (Flight Control) (AMM 22-10-00/501)
(b) Engine Indication and Crew Alerting System (AMM 31-41-00/501)
(c) Air Data Computing System (AMM 34-12-00/501)
(d) Inertial Reference System (AMM 34-21-00/501)
(e) Flight Management Computer System (AMM 34-61-00/501)

S 865-126
(5) Make sure that these circuit breakers on the overhead circuit breaker panel, P11, are closed:
(a) 11A17, AUTO FLIGHT WARN
(b) AIRPLANES WITH THE "LANDING GEAR POSITION AIR/GND SYSTEM 2 ALTN" CIRCUIT BREAKER INSTALLED AT PANEL GRID LOCATION 11C29; 11C29, LANDING GEAR POSITION AIR/GND SYSTEM 2 ALTN
(c) 11C30, LANDING GEAR POSITION AIR/GND SYS 1
(d) 11F14, TMC AC
(e) 11F15, TMC DC
(f) 11F16, TMC SERVO
(g) 11J17, FLAP STAB POS SENSING L
(h) 11U9, MAINT CONT DSPL

(i) SAS 050-149;
11U24, LANDING GEAR POSITION AIR/GND SYS 2

F. Do a Test of the Thrust Management System -- TMC

NOTE: This test makes sure that the TMC and the TMS input interfaces operate correctly.

S 845-034
(1) Initialize the Inertial Reference System in the NAV mode (AMM 34-21-00/501).

S 745-041
(2) Do the MCDP Ground Test 02-TMC (AMM 22-00-02/201).

S 285-042
(3) Make sure that no fault messages are shown after the test starts.

NOTE: Any system that is not powered before you enter the MCDP Ground Test Mode will cause an INFC fault messages to be shown for that system.

G. Do a Test of the Thrust Management System -- TMSP

NOTE: This test makes sure that the TMSP mode switches and the TMSP-TMC interfaces operate correctly.

S 285-047
(1) Make sure that these circuit breakers on the P11 panel are closed:
(a) 11F14, TMC AC
(b) 11F15, TMC DC
(c) 11F16, TMC SERVO

S 745-048
(2) Do MCDP Ground Test 05-TMSP (AMM 22-00-02/201).

S 285-049
(3) Make sure that no fault messages are shown after the test starts.

S 985-050
(4) Push the YES/ADV switch after the MCDP message 05 MAN TEST is shown.

S 985-094
(5) The message 05 PUSH OFF SW PER MODE DSPLY is shown on the MCDP.
(a) Momentarily push the TMSP mode switch for the mode shown on the top EICAS display.

(b) Do this for each mode shown on the EICAS display.

S 285-051

(6) After all of the mode switches have been pushed, the MCDP message 05 TURN TEMP SEL CW-DSPLY INC ? will be shown.

S 985-052

(7) Turn the TEMP SEL knob on the TMSP clockwise and make sure that the temperature select display on top EICAS display increases.

S 985-053

(8) Push the YES/ADV switch.

S 285-054

(9) The MCDP message 05 TURN TEMP SEL CCW-DSPLY DEC? is shown.

S 985-055

(10) Turn the TEMP SEL knob on the TMSP counterclockwise and make sure that the temperature display decreases.

S 985-056

(11) Push the YES/ADV switch and the TEST COMPLETE message will be shown.

H. Do the MCDP Ground Test 52 TMC RLY/SW

S 735-168

(1) Do the MCDP Ground Test 52 TMC RLY/SW task.

I. Put the Airplane back to its initial condition.

S 985-141

(1) Stop the MCDP.

S 085-142

(2) Disconnect the remote MCDP control panel from connector D1447 at the main distribution panel, P6, if it is connected.

S 865-143

(3) Remove electrical power if it is not necessary (AMM 24-22-00/201).

TASK 22-32-00-725-144

4. <u>MCDP Ground Test 52 TMC RLY/SW</u>

A. General

(1) The procedure that follows uses the control switches on the overhead panel, P5, to apply the high and low inputs to the ECS cards. This procedure makes sure that the Thrust Management Computer/ECS card interfaces operate correctly. The procedure uses pneumatic power.

B. Equipment
   (1) Remote MCDP Control Panel - A22001-22 (recommended); A22001-15 (optional)
C. References
   (1) AMM 22-41-00/501, Maintenance Monitor
   (2) AMM 24-22-00/201, Electrical Power - Control
   (3) AMM 29-11-00/201, Main (Left, Right and Center) Hydraulic Systems
   (4) AMM 31-41-00/501, EICAS
   (5) AMM 36-00-00/201, Pneumatic - General
D. Access
   (1) Location Zones

119/120	Main Equipment Center
211/212	Flight Compartment

   (2) Access Panel

119AL	Main Equipment Center

E. Prepare to Test the Thrust Management System

S 865-254
   (1) Make sure the flap control lever is in the zero (FLAPS UP) detent.

S 745-148
   (2) Do the MCDP power-up and self-test (AMM 22-41-00/501).

S 865-149
   (3) Make sure that these systems operate:
      (a) Engine Indication and Crew Alerting System (AMM 31-41-00/501)
      (b) Pneumatic Power (AMM 36-00-00/201)
F. Do the MCDP Ground Test 52 TMC RLY/SW

S 865-064
   (1) Open this circuit breaker on the P11 panel:
      (a) 11G11, AUTO SPEED BRAKE

S 285-066
   (2) Make sure that these circuit breakers on the P11 panel are closed:
      (a) 11A13, PACK FLOW CONT L
      (b) 11A14, AIR SUPPLY ISOL VLV CONT L
      (c) 11A16, ANTI-ICE ENG L
      (d) 11A26, PACK FLOW CONT R
      (e) 11A31, ANTI-ICE WING
      (f) 11C14, FLAP/STAB POS SENSING CTR

(g) AIRPLANES WITH THE "LANDING GEAR POSITION AIR/GND SYSTEM 2 ALTN" CIRCUIT BREAKER INSTALLED AT PANEL GRID LOCATION 11C29;
11C29, LANDING GEAR POSITION AIR/GND SYSTEM 2 ALTN
(h) 11C30, LANDING GEAR POSITION AIR/GND SYS 1
(i) 11D17, LEFT ENGINE EEC DISCRETES
(j) 11D31, HYDRAULIC AIR PUMP
(k) 11F14, TMC AC
(l) 11F15, TMC DC
(m) 11F16, TMC SERVO
(n) 11J17, FLAP/STAB POS SENSING L
(o) 11J26, FLAP/STAB POS SENSING R
(p) 11M32, RIGHT ENGINE EEC DISCRETES
(q) 11S10, LEFT ENG BLEED IND
(r) 11S11, LEFT ENG BLEED CONT
(s) 11S12, ISOL VALVE PWR L
(t) 11S14, ISOL VALVE PWR C
(u) 11S15, ISOL VALVE CONT C
(v) 11S19, RIGHT ENG BLEED IND
(w) 11S20, RIGHT ENG BLEED CONT
(x) 11S21, RIGHT ISOL VALVE PWR
(y) 11S22, RIGHT ISOL VALVE CONT
(z) 11T19, ANTI-ICE ENG R
(aa) 11U9, MAINT CONT DSPL
(ab) 11U15, AIR/GND SYS 1
(ac) SAS 050-149;
11U24, LANDING GEAR POSITION AIR/GND SYS 2
(ad) ALL MTH AIRPLANES;
SAS 150-999;
11U23, LANDING GEAR POSITION AIR/GND SYS 2

S 985-067

(3) Put the switches given in TABLE 1 in the positions shown.

TABLE 1		
SWITCH NAME	SWITCH POSTION	OVERHEAD P5 PANEL LOCATION
L AND R PACK	OFF	TEMPERATURE CONTROL MODULE, M14
WING ANTI-ICE, L AND R ENGINE ANTI-ICE	OFF	WING AND ENGINE ANTI-ICE CONTROL MODULE, M10397
C HYDRAULIC AIR DEMAND PUMP	OFF	HYDRAULIC SYSTEM CONTROL MODULE, M10
L, C, AND R ISLN VALVE	OPEN	BLEED AIR SUPPLY PANEL, M15

S 865-068
(4) Supply Pneumatic Power (AMM 36-00-00/201).

S 865-262
(5) Make sure that the thrust reverser is in the stowed position.

S 745-069
(6) Do MCDP TEST 52 - TMC RLY/SW (AMM 22-00-02/201):

NOTE: In the steps that follow, the operator must do the steps in the right column of the table for each MCDP message shown. If the MCDP does not go on to the subsequent step, it means that there is a fault.

TEST 52 - TMC RLY/SW		
TEST STEP	MCDP MESSAGE	OPERATOR STEP
1	52 TMC RLY/SW TEST?	PRESS THE MCDP YES/ADV SWITCH.

TEST 52 - TMC RLY/SW		
TEST STEP	MCDP MESSAGE	OPERATOR STEP
2	52 IN PROGRESS	The MCDP will show TMC FAIL if the TMC fails the self-test when the GND TEST MODE starts. If the TMC FAIL message is not shown, the MCDP will show any TMC interface faults found when GND TEST MODE starts. Correct all faults before you continue this test.
3	52 VFY PNEU PRESS ON-ADV	PRESS THE YES/ADV SWITCH TO SHOW THAT PNUEMATIC PRESSURE IS SUPPLIED TO THE ECS SYSTEM. MAKE SURE THAT THE LEFT, RIGHT, AND CENTER ISOLATION VALVES ARE OPEN. NOTE: IF POWER IS NOT APPLIED TO THE PNUEMATIC SYSTEM, THE SUBSEQUENT STEPS WILL FAIL.
4	52 SET L PACK SW ON	PUT THE L PACK SWITCH ON THE TEMPERATURE CONTROL PANEL M14 IN THE AUTO POSITION. NOTE: THE TEST ADVANCES TO THE NEXT STEP AUTOMATICALLY IF THE L PACK SWITCH IS ALREADY IN THE AUTO POSITION.
5	52 SET L PACK SW OFF	PUT THE L PACK SWITCH ON THE TEMPERATURE CONTROL PANEL M14 IN THE OFF POSITION.
6	52 SET R PACK SW ON	PUT THE R PACK SWITCH ON THE TEMPERATURE CONTROL PANEL M14 IN THE AUTO POSITION. NOTE: THE TEST ADVANCES TO THE NEXT STEP AUTOMATICALLY IF THE R PACK SWITCH IS ALREADY IN THE AUTO POSITION.
7	52 SET R PACK SW OFF	PUT THE R PACK SWITCH ON THE TEMPERATURE CONTROL PANEL M14 IN THE OFF POSITION.

TEST 52 - TMC RLY/SW		
TEST STEP	MCDP MESSAGE	OPERATOR STEP
8	52 ECS PACK L/R LO LO	WITH THE L AND R PACK SWITCHES IN THE OFF POSITION, THE MCDP BOTTOM LINE MESSAGE SHOWS LO LO. PUT THE L PACK SWITCH IN THE AUTO POSITION.
9	52 ECS PACK L/R HI LO	PUT THE R PACK SWITCH IN THE AUTO POSITION.
10	52 ECS PACK L/R HI HI	PUT THE L AND R PACK SWITCHES IN THE OFF POSITION.
11	52 ECS PACK L/R LO LO	PUSH THE MCDP YES/ADV SWITCH.
12	52 PUSH WG AI ON	PUSH AND HOLD THE WING ANTI ICE TEST SWITCH ON THE MISC TEST PANEL, M10398 (P61).
13	52 PUSH ALL AI SW OFF	SET THE WING ANTI ICE SW TO OFF. MAKE SURE THAT THE LEFT AND THE RIGHT ENGINE # ANTI ICE SWITCHES ARE OFF.
14	52 COWL AI L/R OFF OFF	SET THE L ENGINE ANTI-ICE SWITCH TO ON.
15	52 COWL AI L/R ON OFF	SET THE R ENGINE ANTI-ICE SWITCH TO ON.
16	52 COWL AI L/R ON ON	SET THE L ENGINE ANTI-ICE SWITCH TO OFF.
17	52 COWL AI L/R OFF ON	SET THE R ENGINE ANTI-ICE SWITCH TO OFF.

TEST 52 - TMC RLY/SW		
TEST STEP	MCDP MESSAGE	OPERATOR STEP
18	52 COWL AI L/R OFF OFF	PUSH THE MCDP YES/ADV SWITCH.
19	52 SET ADP ON	PUT THE C HYDRAULIC AIR DEMAND PUMP SWITCH. IN THE ON POSITION (AMM 29-00-00).
20	52 SET ADP OFF	PUT THE C HYDRAULIC AIR DRIVEN PUMP SWTICH IN THE OFF POSITION.
21	52 PUSH ISOL VLV SW L CLOSE	PUT THE L ISLN VALVE IN THE CLOSED POSITION (OPEN BAR NOT SHOWN).
22	52 PUSH ISOL VLV SW L OPEN	PUT THE L ISLN VALVE SWITCH IN THE OPEN POSITION (OPEN BAR SHOWN).
23	52 PUSH ISOL VLV SW R CLOSE	PUT THE R ISLN VALVE SWITCH IN THE CLOSED POSITION (OPEN BAR NOT SHOWN).
24	52 PUSH ISOL VLV SW R OPEN	PUT THE R ISLN VALVE SWITCH IN THE OPEN POSITION (OPEN BAR SHOWN).
25	52 VFY THRST LVR SYST CLR-ADV	PUSH THE MCDP YES/ADV SWITCH UNTIL THE TEST COMPLETE MESSAGE IS SHOWN. NOTE: IGNORE INTERMEDIATE TEST STEPS.
26	52 TEST COMPLETE	PUSH THE MCDP YES/ADV SWITCH OR THE NO/SKIP SWITCH TO EXIT THE TEST.

G. Put the Airplane back to its initial condition.

S 865-070

(1) Remove pneumatic power if it is not necessary.

S 985-071
(2) Stop the MCDP.

S 085-072
(3) Disconnect the remote MCDP control panel from connector D1447 at the main distribution panel, P6, if it is connected.

S 865-073
(4) Remove electrical power if it is not necessary (AMM 24-22-00/201).

AUTOTHROTTLE SERVOMOTOR GENERATOR AND GEARBOX - REMOVAL/INSTALLATION

1. General
   A. The autothrottle servomotor generator is installed on the gearbox of the autothrottle drive assembly. The servo is held in position by three bolts. The unit has one multi-pin electrical connector. Access to the servomotor generator/gearbox assembly is through the forward equipment bay access panel 113AL. The servomotor generator and gearbox can be removed and installed as an assembly, to make other maintenance procedures easier.

TASK 22-32-01-004-001

2. Remove the Servomotor Generator and/or the Gearbox (Fig. 401)
   A. Reference
      (1) AMM 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
   B. Access
      (1) Location Zones
          113/114 Area Forward of NLG Wheel Well
          211/212 Flight Compartment

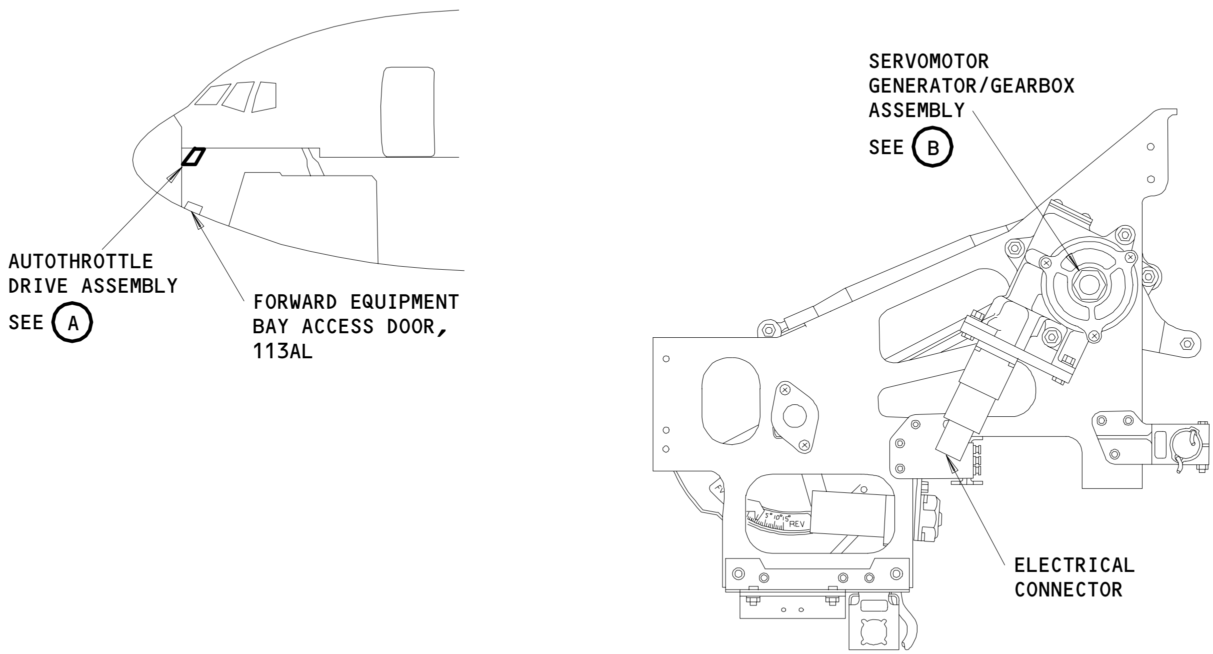
      (2) Access Panel
          113AL Forward Equipment Bay

   C. Prepare for Removal

      S 864-002
      (1) Open this circuit breaker on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tag:
          (a) 11F14, TMC AC

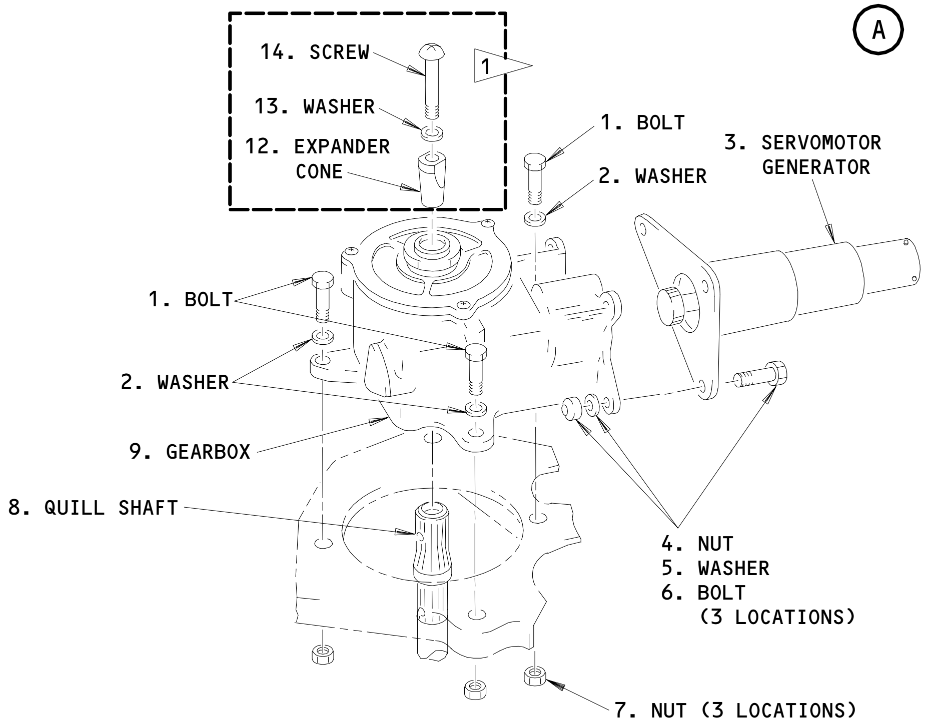
      S 864-003
      (2) Attach DO-NOT-OPERATE tags on the left and right throttle levers.

      S 014-004
      (3) Open the access panel, 113AL, for the autothrottle servomotor generator and gearbox (AMM 06-41-00/201).



AUTOTHROTTLE DRIVE ASSEMBLY

A



SERVOMOTOR GENERATOR/GEARBOX ASSEMBLY

B

1 SAS 150-154 WITH SB 22-81, AND
SAS 050,051,162-166;
MTH 275,276

Servomotor Generator/Gearbox - Installation
Figure 401

D. Remove the Servomotor Generator and/or the Gearbox

S 034-005
(1) Disconnect the electrical connector (A).

S 024-006
(2) Remove the servomotor generator/gearbox assembly (B) as follows:
(a) SAS 150-154 POST-SB 22-81;
SAS 050-051, 155-157, 162-166;
MTH 275-276;
Remove the screw (14), washer (13), and expander cone (12) from the end of the quill shaft (8) (adjacent to the servomotor generator/gearbox assembly).
(b) SAS 050-059, 150-274;
MTH 275-299;
Hold the servomotor generator/gearbox assembly and remove bolts (1), washers (2) and nuts (7) that attach the gearbox (9) to the autothrottle drive assembly.

S 024-007
(3) Remove the servomotor generator (3) as follows:
(a) Hold the servomotor generator (3) and remove the bolts (6), washers (5), and nuts (4) that attach the generator (3) to the gearbox (9).

TASK 22-32-01-404-008
3. Install the Servomotor Generator (Fig. 401)
A. Consumable Material
(1) D00633 Grease - BMS3-33 (Preferred)
(2) D00013 Grease - MIL-PRF-23827 (Supersedes MIL-G-23827) (Alternate)
B. Parts
(1) SAS 050-059, 150-274;
MTH 275-299;
Use the table that follows:

AMM			AIPC		
FIG	ITEM	NOMENCLATURE	SUBJECT	FIG	ITEM
401	1	Bolt	22-32-01	02	335
	1	Bolt			336
	2	Washer			340
		Washer	22-32-01	02A	385
	3	Servomotor Generator	22-32-01	02	373
	4	Nut			355
	5	Washer			350
	6	Bolt			345
	7	Nut			342
	8	Quill Shaft			375
	9	Gearbox			360

(2) SAS 150-154 POST-SB 22-81;
SAS 050-051, 155-157, 162-166;
MTH 275-276;
Use the table that follows:

AMM		NOMENCLATURE	AIPC		
FIG	ITEM		SUBJECT	FIG	ITEM
	12	Expander Cone	22-32-01	02	315
	13	Washer			325
	14	Screw			320

C. Access
(1) Location Zones
113/114 Area Forward of NLG Wheel Well

(2) Access Panel
113AL Forward Equipment Bay

D. Install the Servomotor Generator

S 644-009
(1) Apply a layer of grease on all the mating surfaces.

S 424-010

CAUTION: YOU MUST BE CAREFUL WHEN YOU INSTALL THE SERVOMOTOR GENERATOR AS THE SPLINE ON THE GEARBOX CAN EASILY BE DAMAGED.

(2) Hold the servomotor generator (3) on the gearbox (9) and install the bolts (6), washers (5), and nuts (4).

S 424-076
(3) Connect the electrical connector.

S 714-012
(4) Do a test of the servomotor generator/gearbox assembly.

TASK 22-32-01-404-019
4. Install the Servomotor Generator/Gearbox Assembly (Fig. 401)
A. Access
(1) Location Zones
113/114 Area Forward of NLG Wheel Well

(2) Access Panel
113AL Forward Equipment Bay

B. Install the Servomotor Generator/Gearbox Assembly

S 424-020

(1) Hold the generator/gearbox assembly and connect the gearbox assembly spline with the quill shaft (8).

(a) SAS 050-059, 150-274;
MTH 275-299;
Install the bolts (1), washers (2) and nuts (7) that attach the gearbox (9) to the autothrottle drive assembly.

S 824-000

(2) SAS 150-154 POST SB 22-81;
SAS 050-051, 155-157, 162-166;
MTH 275-276;
Install the Expander Cone (12).

(a) Measure the expander cone attach-screw (14) run-on torque:

1) Apply a layer of grease on the screw (14).
2) Install the screw (14) in the quill shaft (8) without the expander cone (12).
3) Tighten the screw (14) by hand until the screw starts to engage the self-locking insert in the end of the quill shaft (8).
4) Tighten the screw (14) six turns more.
5) Use a dial torque wrench to measure the torque necessary to turn the screw (14) (after the screw was turned six times).
6) The torque value is the run-on torque for the screw (14).

NOTE: The run-on torque of the screw will increase as the screw is installed in the quill shaft.

7) Remove the screw (14).

(b) Apply a layer of grease on the screw (14), expander cone (12), and quill shaft (8).

(c) Install the expander cone (12) in the quill shaft (8) with the screw (14) and washer (13).

CAUTION: YOU MUST NOT TORQUE THE EXPANDER CONE ATTACH-SCREW MORE THAN NECESSARY AS THE GEARBOX CAN BE DAMAGED.

(d) Tighten the screw (14) to 5-8 in-lbs more than run-on torque.

S 434-033

(3) Connect the electrical connector.

S 714-034
(4) Do a test of the servomotor generator/gearbox assembly.

TASK 22-32-01-714-035
5. Test the Servomotor Generator/Gearbox Assembly
A. References
(1) AMM 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
(2) AMM 22-00-02/201, Autoflight BITE
(3) AMM 24-22-00/201, Electrical Power - Control
(4) AMM 27-61-00/201, Spoiler/Speedbrake Control System
B. Access
(1) Location Zones

113/114	Area Forward of NLG Wheel Well
211/212	Flight Compartment

(2) Access Panel

113AL	Forward Equipment Bay

C. Prepare for Test

S 864-036
(1) Supply electrical power (AMM 24-22-00/201).

S 864-037
(2) Remove the DO-NOT-CLOSE tag and close this circuit breaker on the P11 panel:
(a) 11F14, TMC AC

S 864-038
(3) Remove the DO-NOT-OPERATE tags on the left and right throttle levers.

S 864-039
(4) Set the A/T switch on the AFCS mode control panel (on the P55 panel) to ARM.

S 864-040

WARNING: THIS TEST USES MOVEMENT OF THRUST LEVERS AND CAN CAUSE AIRPLANE MOVEMENT IF ENGINES ARE ON OR SPOILER/SPEEDBRAKE MOVEMENT IF PRESSURE TO THE HYDRAULIC SYSTEMS IS ON. REFER TO THE SPOILER/SPEEDBRAKE DEACTIVATION PROCEDURE (AMM 27-61-00/201). ACCIDENTAL ENGINE OPERATION OR SPOILER/SPEEDBRAKE MOVEMENT CAN CAUSE BAD INJURY TO PERSONS.

(5) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoilers.

S 864-041

(6) Make sure the engines are not on.

D. Procedure

S 714-042

(1) Do the MCDP Ground Test 10 SERVO A/T (AMM 22-00-02/201).

E. Put the Airplane Back to Its Initial Condition

S 864-043

(1) Set the A/T switch on the AFCS mode control panel to OFF.

S 864-044

(2) Do the activation procedure for the spoilers if you did the deactivation procedure (AMM 27-61-00/201).

S 414-045

(3) Close the access panel, 113AL (AMM 06-41-00/201).

S 864-046

(4) Remove electrical power if it is not necessary (AMM 24-22-00/201).

DISENGAGE SWITCHES - REMOVAL/INSTALLATION

1. General
   A. This procedure has two tasks:
      (1) Removal of the autothrottle disconnect switch.
      (2) Installation of the autothrottle disconnect switch.

   TASK 22-32-02-004-002
2. Remove the Disengage Switch (Fig. 401)
   A. General
      (1) The autothrottle disconnect switches are installed in the handles of the forward thrust levers.
   B. Reference
      (1) AMM 27-61-00/201, Spoiler and Speedbrake Control System
   C. Access
      (1) Location Zone
          211/212 Flight Compartment

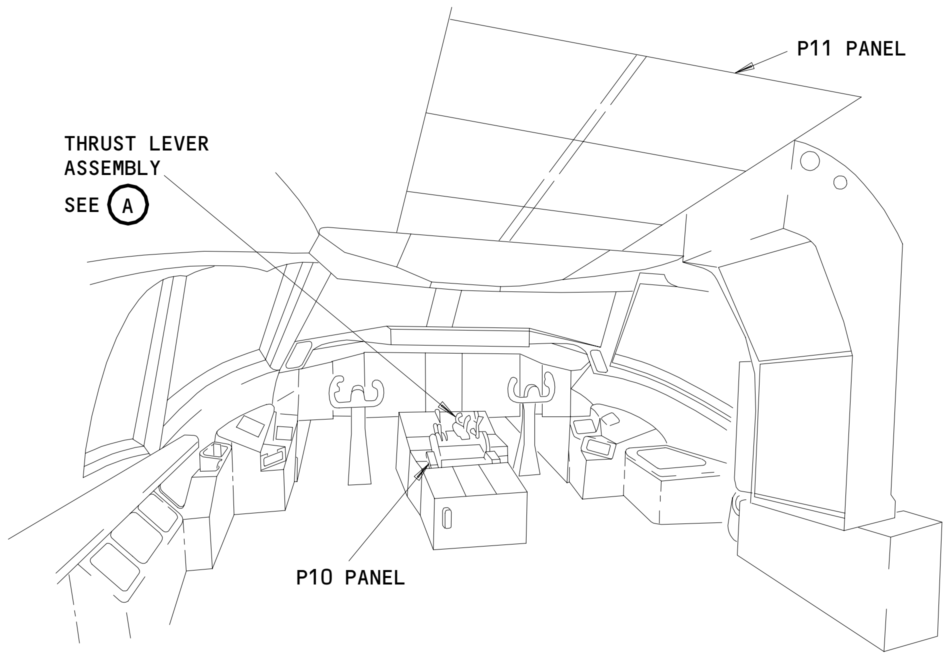
   D. Prepare to Remove the Disengage Switch

      S 044-021

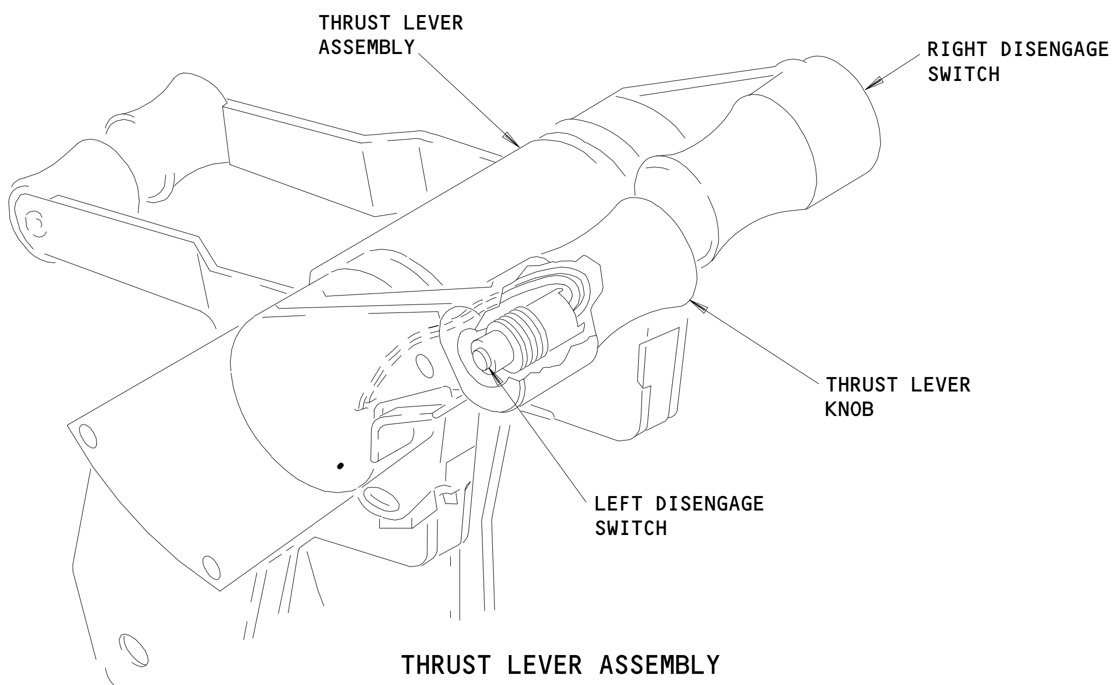
      WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

      (1) Do the deactivation procedure for the spoilers or move all persons and equipment away from the spoilers (AMM 27-61-00/201).

      S 864-001
      (2) Open this circuit breaker on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tag:
          (a) 11F15, TMC DC



FLT COMPT



THRUST LEVER ASSEMBLY

A

Disengage Switches - Installation
Figure 401

E. A/T Disengage Switch Removal

S 014-003
(1) Remove the thrust lever knob.

S 034-004
(2) Remove the heat-shrink tubing from the wire terminals.

S 034-019
(3) Remove the screw-attached wires from the disengage switch.

S 024-005
(4) Remove the disengage switch.

TASK 22-32-02-404-006
3. Install the Disengage Switch (Fig. 401)
A. References
(1) FIM 22-00-03/101, MCDP Ground Test Mode
(2) AMM 24-22-00/201, Electrical Power - Control
(3) AMM 27-61-00/201, Spoiler and Speedbrake Control System
B. Access
(1) Location Zones
211/212 Flight Compartment

C. Consumable Materials
(1) A00950 Locktite 222
(2) G02104 Heat Shrink Tubing - RT876, Color Yellow, Raychem Corp., Menlo Park, CA.
D. Install the Disengage Switch

S 164-007
(1) Remove all of the remaining locktite from the thrust lever threads.

S 394-008
(2) Apply new locktite to the disengage switch threads.

S 424-017
(3) Install the disengage switch.

S 434-009
(4) Apply the heat-shrink tubing to the wire terminals (heat shrink after the wires are installed).

S 434-010
(5) Use the screws supplied with the switch to install the wires on the switch.

S 394-018
(6) Heat shrink the heat-shrink tubing.

S 414-011
(7) Install the thrust lever knob on the thrust lever.

E. Autothrottle Disengage Switch Test

S 864-012
(1) Supply electrical power (AMM 24-22-00/201).

S 864-013
(2) Remove the DO-NOT-CLOSE tag and close this circuit breaker on the P11 panel:
(a) 11F15, TMC DC

S 864-014
(3) Make sure these circuit breakers on the P11 panel are closed:
(a) 11F14, TMC AC
(b) 11F16, TMC SERVO

S 714-015
(4) Do MCDP Ground Test 12 - SW A/T DISC (FIM 22-00-03/101).

F. Put the Airplane Back to Its Usual Condition

S 864-016
(1) Remove electrical power if it is not necessary (AMM 24-22-00/201).

S 864-032
(2) If the Spoiler Deactivation was accomplished, then do this task: Spoiler Activation (AMM 27-61-00/201).

LEADING EDGE SLAT SWITCH - REMOVAL/INSTALLATION

1. General
    A. This procedure has two tasks.
        (1) Removal of the leading edge slat switches.
        (2) Installation of the leading edge slat switches.
    B. The leading edge slat switch assembly is installed below the coverplate aft of the flap lever. The switch assembly has four microswitches and a spacer. A placard on the bottom side of the coverplate shows the switch locations. Electrical connections to the switches are soldered.

TASK 22-32-03-004-003

2. Remove the Leading Edge Slat Switch (Fig. 401)
    A. Access
        (1) Location Zone
            211/212 Flight Compartment

    B. Equipment
        (1) Soldering Iron
    C. Prepare For Removal

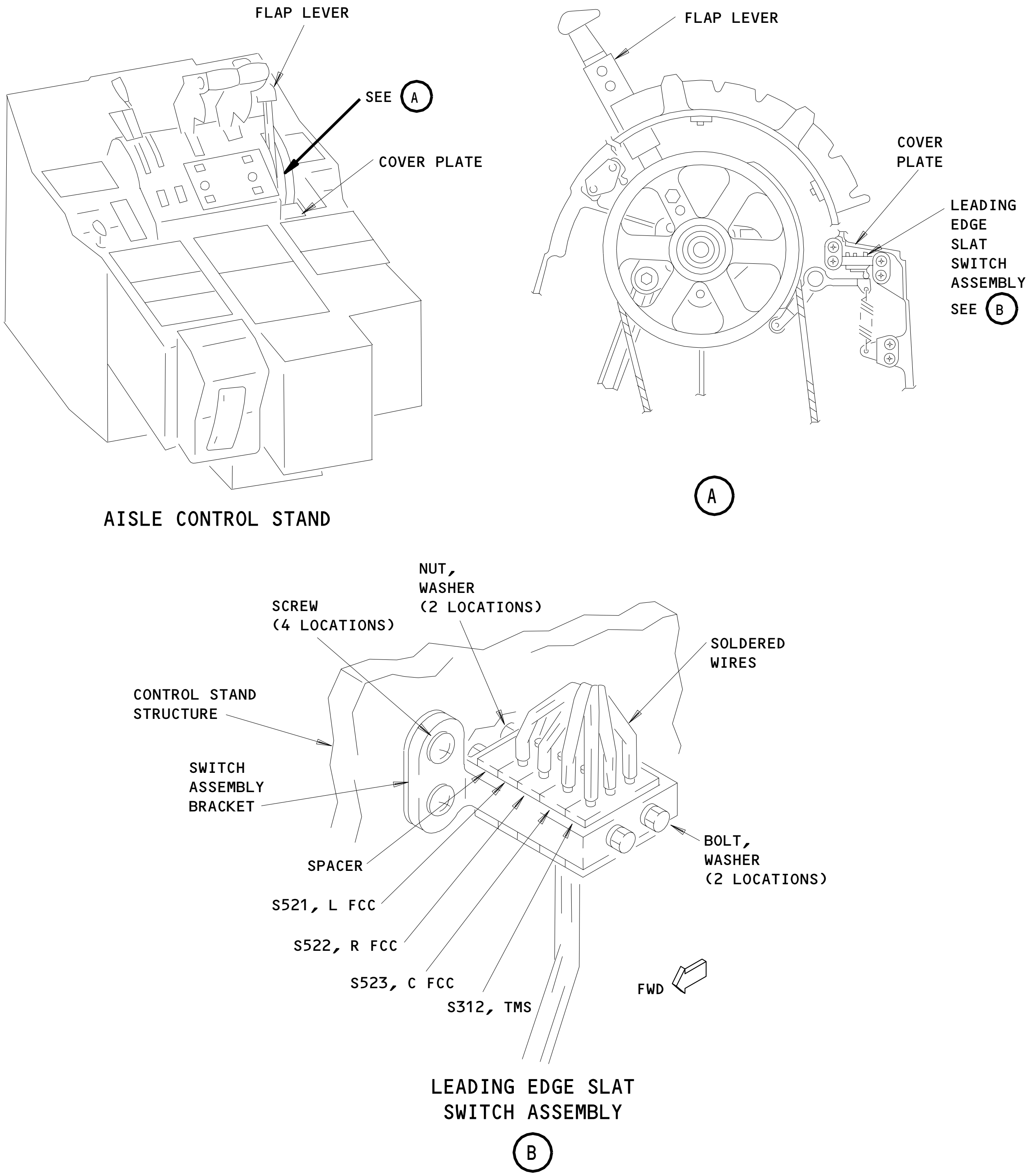
        S 864-001
        (1) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
            (a) 11E17, FLT CONT CMPTR PWR L
            (b) 11E20, FLT CONT CMPTR PWR C or FLT CONT CMPTR PWR
            (c) 11E35, FLT CONT CMPTR PWR R or RIGHT FLT CONT CMPTR PWR
            (d) 11F15, TMC DC
    D. Leading Edge Slat Switch Removal

        S 014-002
        (1) Remove the two coverplate screws and remove the coverplate.

        S 034-004
        (2) Remove the four screws that attach the switch assembly bracket to the control stand structure.

        S 034-005
        (3) Remove the bolts, washers, and nuts that hold the switches in the switch assembly bracket.

        NOTE: Remove and keep shims as installed.



Leading Edge Slat Switch - Installation
Figure 401

S 024-020
(4) Use the soldering iron to remove wing from the switch (AMM 20-10-26/201).

S 024-007
(5) Remove the switch.

TASK 22-32-03-404-008
3. Install the Leading Edge Slat Switch (Fig. 401)
A. References
(1) AMM 20-10-26/201, Heat Guns, Soldering Irons, and Soldering Guns
(2) AMM 24-22-00/201, Electrical Power - Control
(3) FIM 22-00-03/101, Autoflight BITE MCDP Ground Tests
B. Access
(1) Location Zone
211/212 Flight Compartment

C. Equipment
(1) Soldering Iron
D. Leading Edge Slat Switch Installation

S 314-009
(1) Solder the wires to the switch (AMM 20-10-26/201).

S 424-010
(2) Install the switch in the switch assembly bracket.

S 434-011
(3) Install the bolts, washers, and nuts that hold the switches in the switch assembly bracket.

NOTE: Add laminated shims (or washers, P/N NAS1149DN316J) if the switch gap is greater than 0.003 inch.

S 434-012
(4) Install the screws that attach the switch assembly bracket to the control stand structure.
E. Do the Leading Edge Slat Switch Test

S 864-013
(1) Supply electrical power (AMM 24-22-00/201).

S 864-014
(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11E17, FLT CONT CMPTR PWR L

(b) 11E20, FLT CONT CMPTR PWR C or FLT CONT CMPTR PWR
(c) 11E35, FLT CONT CMPTR PWR R or RIGHT FLT CONT CMPTR PWR
(d) 11F15, TMC DC

S 714-015
(3) Do the MCDP Ground Test 64 SPD BK/FLAP XDCR (FIM 22-00-03/101).

F. Put the Airplane Back to Its Usual Condition

S 414-026
(1) Install the coverplate with the screws.

S 864-017
(2) Remove electrical power if it is not necessary (AMM 24-22-00/201).

MICROSWITCH PACK - MAINTENANCE PRACTICES

1. General
   A. The microswitch pack assembly is attached to the bottom of the autothrottle clutch pack and cable quadrant frame. You can get access to the microswitch assembly through the forward equipment bay access hatch, 113AL (AMM 06-41-00/201). The microswitches are installed on movable mounting arms that can be adjusted as necessary to position the switches for correct operation. The switches are activated by one of four arm assemblies controlled by the rotation of the autothrottle cams.
   B. A check of all the microswitches must be done after any microswitch pack assembly removal and installation. To adjust the switches, you push down on the locking channel, which releases the adjusting bolt, and turn the adjusting bolt as required to correctly position the switch. The adjusting bolt is then turned the minimum amount necessary to allow the locking channel to rise and lock the bolt in postion.

TASK 22-32-04-002-067

2. Remove the Microswitch Pack (Fig. 201)
   A. Equipment
      (1) DVM, John Fluke, Model 8020B
      (2) Rig Pin P7 - P/N A20004-23, part of set A20004-XX (AMM 20-10-24/201)
   B. References
      (1) AMM 06-41-00/201, Fuselage (Major zones 100 and 200) Access Doors and Panels
      (2) AMM 24-22-00/201, Electrical Power - Control
      (3) AMM 27-61-00/201, Spoiler/Speedbrake Control System
      (4) AMM 78-31-00/201, Thrust Reverser System (Deactivation/Activation)
   C. Access
      (1) Location Zone
          113/114 Area Forward of NLG Wheel Well

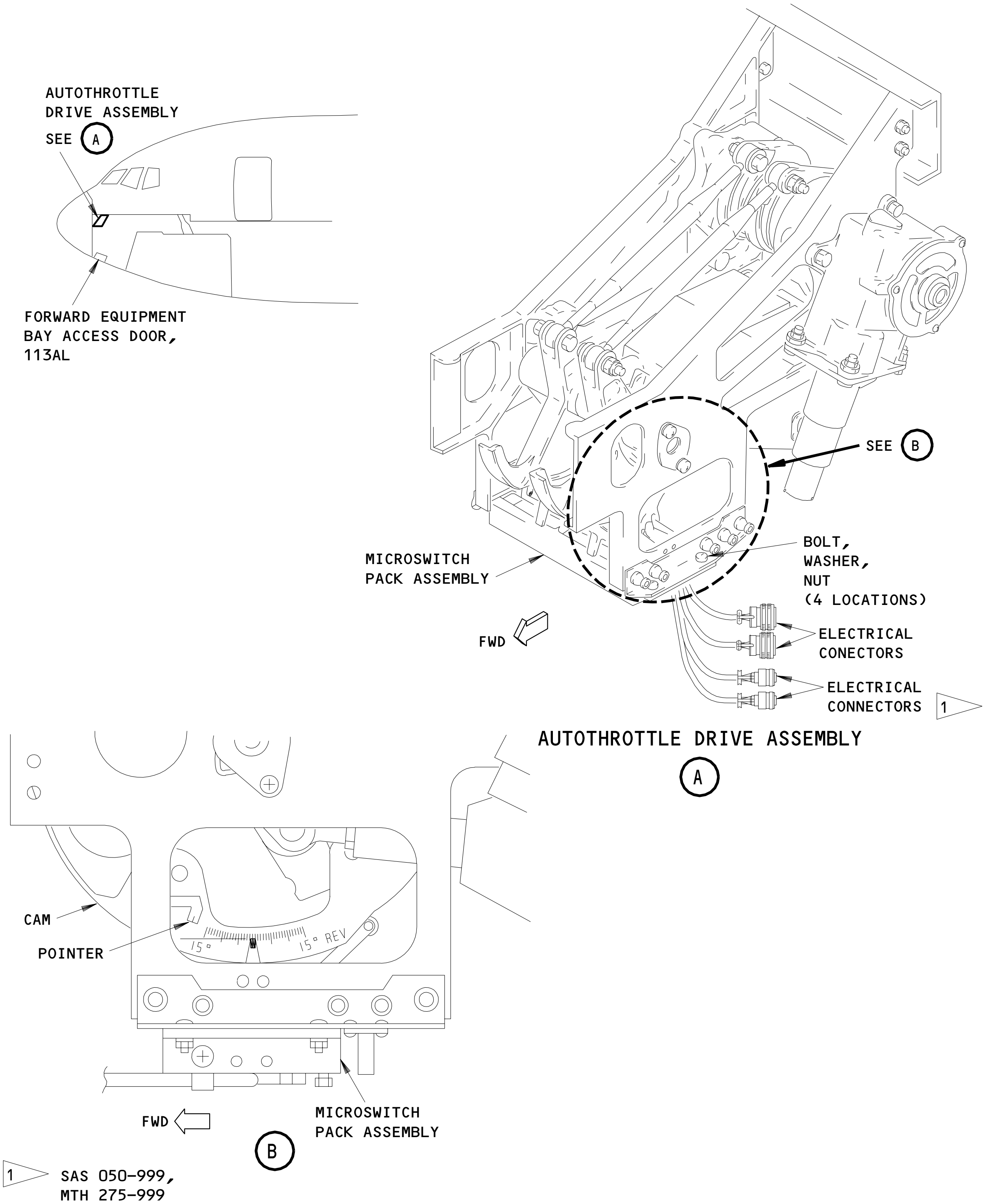
      (2) Access Panel
          113AL Forward Equipment Center

   D. Prepare For Removal

      S 042-002

      WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILER PANELS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURY TO PERSONS OR DAMAGE TO EQUIPMENT.

      (1) Do the deactivation procedure for the spoilers (AMM 27-61-00/201) or move all persons and equipment away from the spoiler panels.



Microswitch Pack - Installation
Figure 201

S 842-068

WARNING: FAILURE TO DEACTIVATE THE THRUST REVERSER ISOLATION VALVE FOR GROUND MAINTENANCE COULD RESULT IN ACCIDENTAL THRUST REVERSER OPERATION WITH POSSIBLE INJURY TO PERSONS AND/OR DAMAGE TO EQUIPMENT.

(2) Deactivate the THRUST REVERSER ISOLATION VALVE for ground maintenance (AMM 78-31-00/201).

S 862-002

(3) Open these circuit breakers on the overhead circuit breaker panel, P11, and attach DO-NOT-CLOSE tags:
  (a) 11F15, TMC DC
  (b) 11G11, AUTO SPEEDBRAKE
  (c) 11M5, LEFT ENGINE EEC DISCRETES
  (d) 11M32, RIGHT ENGINE EEC DISCRETES
  (e) 11U12, AUTOBRKS ANTISKID TEST/IND 1
  (f) 11U21, AUTOBRKS ANTISKID TEST/IND 2

E. Remove the Microswitch Pack Assembly (Fig. 201)

S 032-003

(1) Disconnect all electrical connectors from the microswitch assembly.

S 032-004

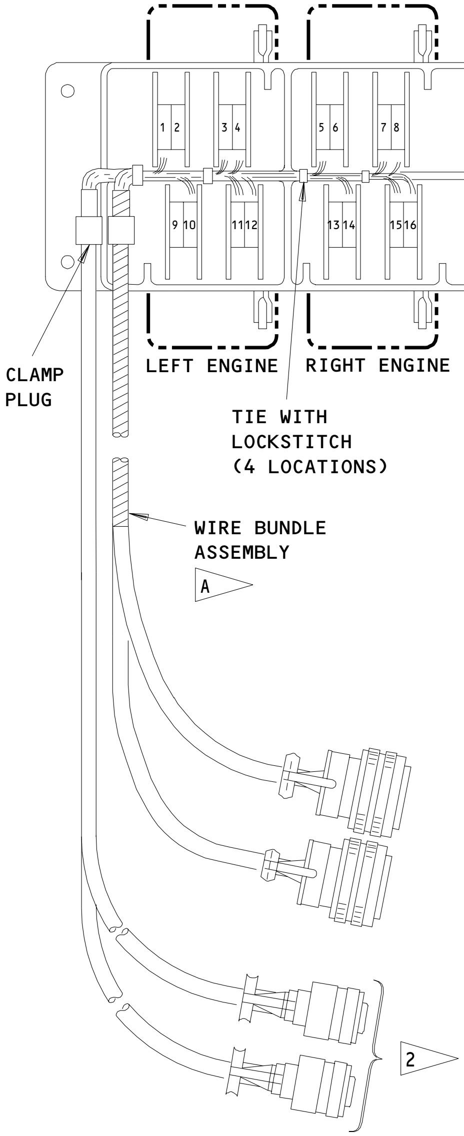
(2) Remove the four bolts, washers, and nuts that hold the microswitch assembly to the autothrottle frame assembly.

S 022-005

(3) Remove the microswitch pack assembly.

TASK 22-32-04-402-069

3. Install the Microswitch Pack (Fig. 201)
  A. Equipment
    (1) DVM, John Fluke, Model 8020B



MICROSWITCH PACK SWITCHES

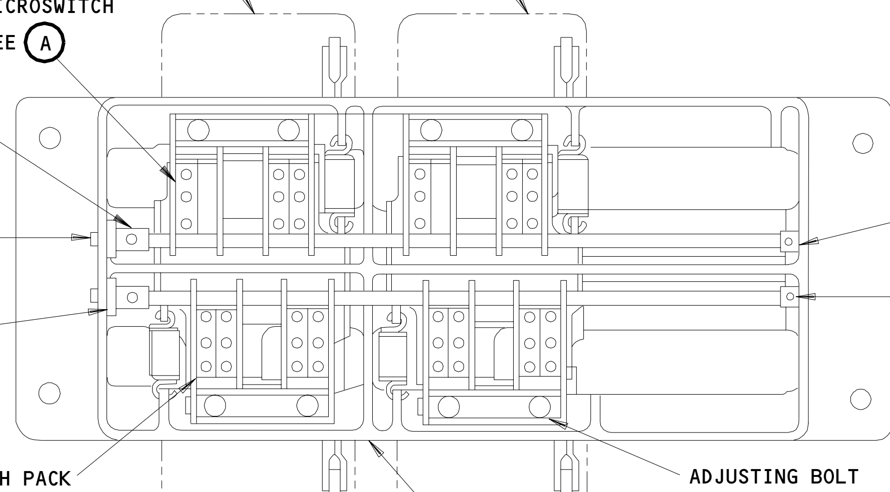
	SWITCH POSITION	SWITCH OR SPACER	FUNCTION
LEFT ENGINE	1	S3	LEFT AUTOBRAKE/AUTOBRAKE RTO
	2	SPACER	NONE
	3	S1	LANDING WARNING LEFT
	4	S2	LEFT AUTOBRAKE/AUTOBRAKE RTO
	9	S11	REVERSE THRUST DIRECTIONAL VALVE LEFT
	10	S21 [1]	LEFT THRUST REVERSE LOCK
	11	S17	LOAD SHED/PRESSURE CONTROL LEFT
	12	S10	SPEED BRAKE RETRACT LEFT
RIGHT ENGINE	5	S7	RIGHT AUTOBRAKE/AUTOBRAKE RTO
	6	SPACER	NONE
	7	S6	RIGHT AUTOBRAKE/AUTOBRAKE RTO
	8	S5	LANDING WARNING RIGHT
	13	S8	REVERSE THRUST DIRECTIONAL VALVE RIGHT
	14	S22 [1]	RIGHT THRUST REVERSE LOCK
	15	S18	LOAD SHED/PRESSURE CONTROL RIGHT
	16	S14	SPEED BRAKE RETRACT RIGHT

TMS: THRUST MANAGEMENT SYSTEM
RTO: REJECT TAKE-OFF

[A] BACP20BA FILLER PLUG CAN BE USED TO PREVENT CABLE MOVEMENT IN CLAMP.

[1] SAS 051-999;
MTH 275-999

Microswitch Pack Assembly Switches - Installation
Figure 202 (Sheet 1)



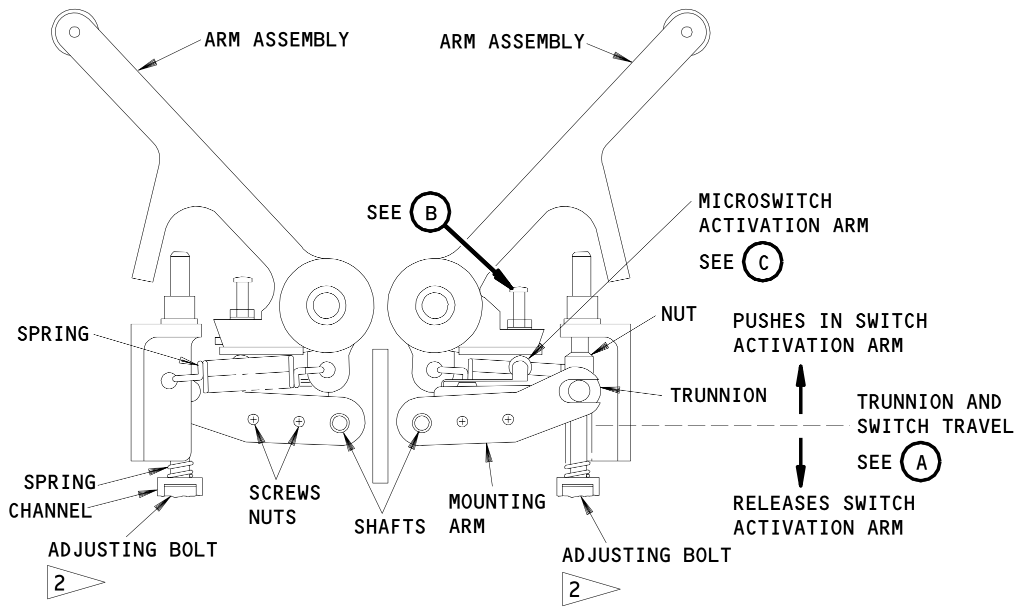
MICROSWITCH PACK ASSEMBLY
(BOTTOM VIEW)

SWITCH LOCATIONS
1 THRU 8

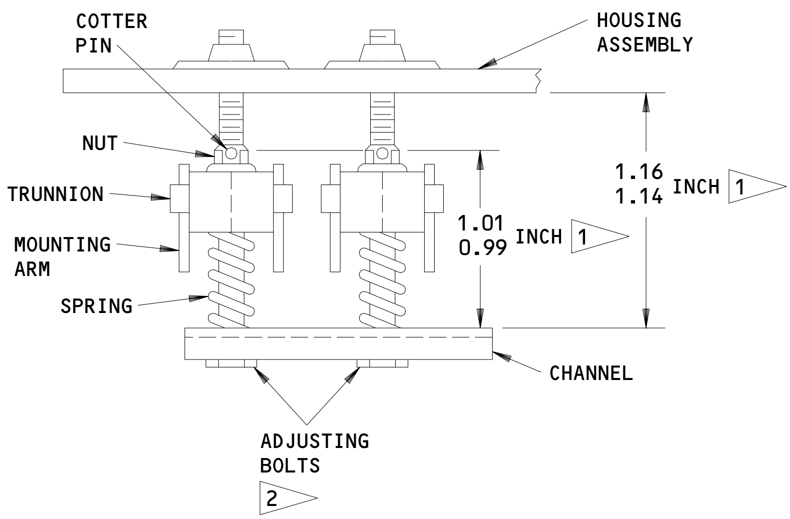
SWITCH LOCATIONS
9 THRU 16

MICROSWITCH PIN ARRANGEMENT
(EXAMPLE)
A

Microswitch Pack Assembly Switches - Installation
Figure 202 (Sheet 2)



MICROSWITCH PACK - ARM ASSEMBLY AND MICROSWITCH OPERATION
(EXAMPLE)



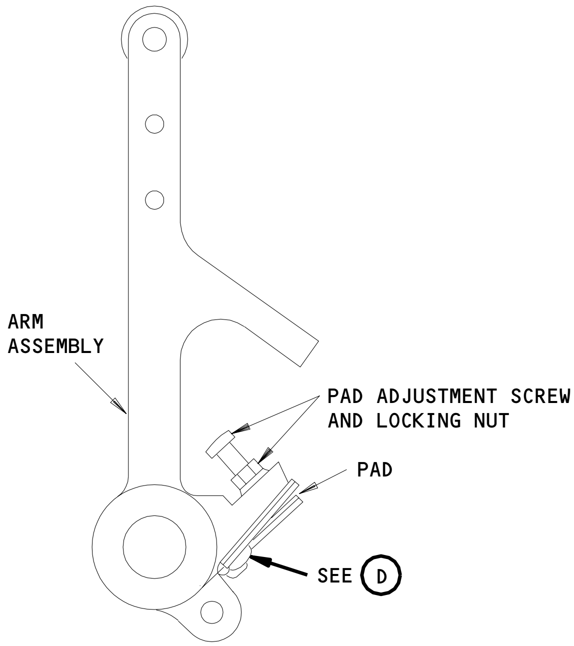
MICROSWITCH PACK - MICROSWITCH MOUNTING ARM/TRUNION DETAIL

A

1 INITIAL ADJUSTMENT RANGE

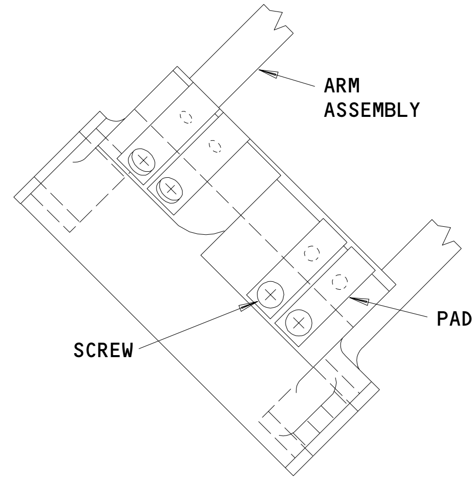
2 ADJUSTING BOLT ADJUSTMENT DIRECTIONS:
- CW DIRECTION - PUSHES IN THE MICROSWITCH ACTIVATION ARM.
- CCW DIRECTION - RELEASES THE MICROSWITCH ACTIVATION ARM.

Microswitch Pack Assembly - Switch Operation and Adjustment
Figure 203 (Sheet 1)



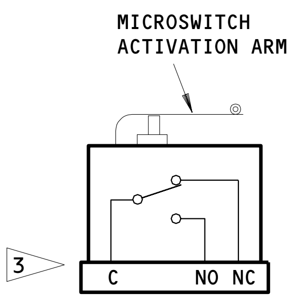
ARM ASSEMBLY
(EXAMPLE

B

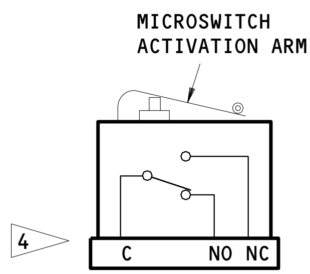


PAD ADJUSTMENT

D



RELEASED STATE
(NC - NORMALLY CLOSED)



PUSHED IN (ACTIVATED) STATE
(NO - NORMALLY OPEN)

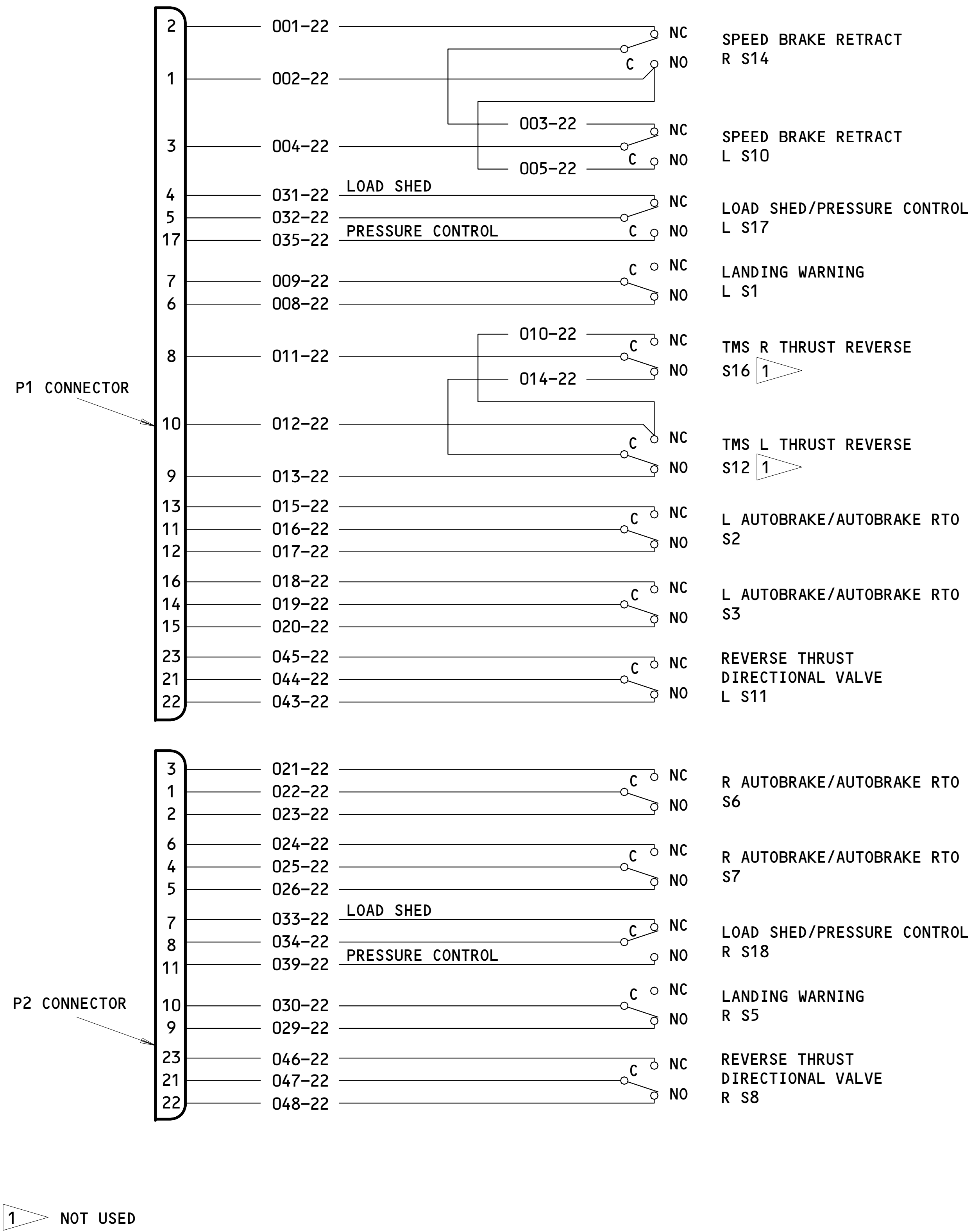
MICROSWITCH PACK - SWITCH OPERATION MODES
(EXAMPLE)

C

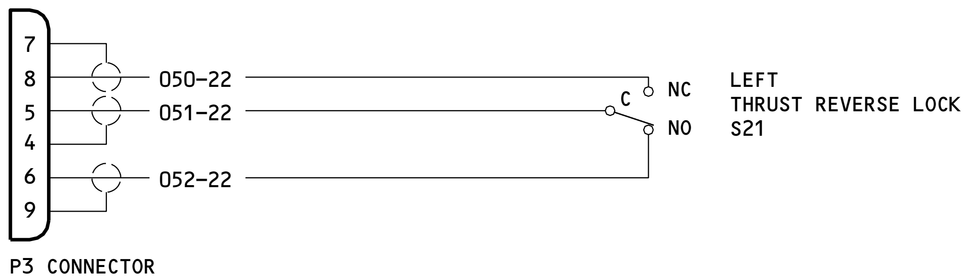
3 POSITION OF MICROSWITCH WHEN THE ACTIVATION ARM IS IN THE RELEASED STATE.

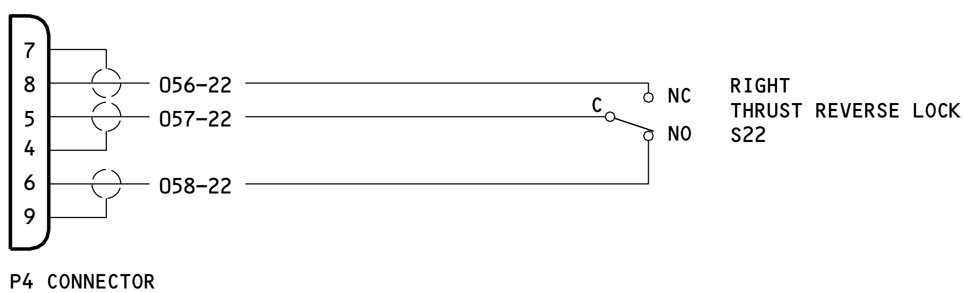
4 POSITION OF MICROSWITCH WHEN THE ACTIVATION ARM IS IN THE PUSHED IN (ACTIVATED) STATE.

Microswitch Pack Assembly - Switch Operation and Adjustment
Figure 203 (Sheet 2)



Microswitch Pack - Wire Schematic
Figure 204 (Sheet 1)





NOTE: SWITCH CONDITIONS SHOWN WITH THROTTLE AT IDLE POSITION (A/T CAM AT 0°).

Microswitch Pack - Wire Schematic
Figure 204 (Sheet 2)

(2) Rig Pin P7 - P/N A20004-23, part of set A20004-XX (AMM 20-10-24/201)

B. References
(1) AMM 06-41-00/201, Fuselage (Major zones 100 and 200) Access Doors and Panels
(2) AMM 24-22-00/201, Electrical Power - Control
(3) AMM 27-61-00/201, Spoiler/Speedbrake Control System
(4) AMM 78-31-00/201, Thrust Reverser System (Deactivation/Activation)

C. Access
(1) Location Zone

113/114 Area Forward of NLG Wheel Well

(2) Access Panel

113AL Forward Equipment Center

D. Install the Microswitch Pack Assembly (Fig. 201)

S 422-016

(1) Install the microswitch pack and attach the four bolts, washers, and nuts.

E. Do this microswitch adjustment procedure.

S 832-042

(1) With the throttles in the idle position, insert rig pin through the autothrottle side frames and brakes. The rig pin must fit freely to make sure that the autothrottle clutch link is centered.

S 282-043

(2) Make sure that the "0" degree mark on the side of each cam aligns with the pointer on the side frames. Bend the pointer as necessary to align it with the "0" degree mark (Fig. 201).

S 832-044

(3) Remove the rig pin.

S 822-092

(4) Do these steps to adjust switches S1, S2, S3, S5, S6, and S7 (Fig. 205).
(a) Step 1:
1) Align the left and right autothrottle CAM's to the side frame pointers as directed in step 1, fig 205, for switches S1, S2, S3, S5, S6 and S7. (Align the quadrant decals with the pointers on the side frame.)

SWITCH	STEP 1	AUTOTHROTTLE CAM POSITION 2		CONNECTOR 3	PIN 3	PIN CONTINUITY CONDITION 4	SWITCH ACTIVATION ARM CONDITION 5
		LEFT	RIGHT				
S1	1	0°	-	P1	6,7	CLOSED	PUSHED IN
S2	1	0°	-	P1	11,12	CLOSED	PUSHED IN
S3	1	0°	-	P1	14,15	CLOSED	PUSHED IN
S5	1	-	0°	P2	9,10	CLOSED	PUSHED IN
S6	1	-	0°	P2	1,2	CLOSED	PUSHED IN
S7	1	-	0°	P2	4,5	CLOSED	PUSHED IN
S1	2	3.5 ±0.5° (FWD)	-	P1	6,7	OPEN	RELEASED
S2	2	3.5 ±0.5° (FWD)	-	P1	11,12	OPEN	RELEASED
S3	2	3.5 ±0.5° (FWD)	-	P1	14,15	OPEN	RELEASED
S5	2	-	3.5 ±0.5° (FWD)	P2	9,10	OPEN	RELEASED
S6	2	-	3.5 ±0.5° (FWD)	P2	1,2	OPEN	RELEASED
S7	2	-	3.5 ±0.5° (FWD)	P2	4,5	OPEN	RELEASED

1 SWITCHES MUST BE ADJUSTED IN THE SEQUENCE SHOWN.

2 SEE FIG. 201 TO SET AUTOTHROTTLE QUADRANT:
- (FWD) FORWARD THRUST DIRECTION FROM 0° REFERENCE
- (REV) REVERSE THRUST DIRECTION FROM 0° REFERENCE

3 SEE FIG. 204 FOR WIRING SCHEMATIC.

4 ELECTRICAL CONDITION OF SWITCH CIRCUIT MEASURED BETWEEN INDICATED PINS.

5 MECHANICAL CONDITION OF MICROSWITCH ASSEMBLY ARM

Microswitch - Adjustment Settings and Conditions
Figure 205

SWITCH 1	STEP	AUTOTHROTTLE QUADRANT POSITION ±0.5°		MICROSWITCH PACK 2	
		LEFT	RIGHT	CONNECTOR	PIN
S10 & S14 4	1	0°	0°	P1	2 & 3
S10	2	8.5° FWD	0°	P1	1 & 3
S14	3	0°	8.5° FWD	P1	1 & 3
S21 & S22 3	1	0°	0°	P3 & P4	5 & 6
S11	1	0°	0°	P1	21 & 22
S8	1	0°	0°	P2	21 & 22
S21 3	2	3.5° REV	0°	P3	5 & 8
S11	2	3.5° REV	0°	P1	21 & 23
S22 3	3	0°	3.5° REV	P4	5 & 8
S8	3	0°	3.5° REV	P2	21 & 23

1 SWITCHES MUST BE RIGGED IN THE ADJUSTMENT SEQUENCE SHOWN.

2 SEE FIG. 204 FOR WIRING SCHEMATIC

3 SAS 050-999; MTH 275-999

4 SWITCHES S17 AND S18 ARE ASLO ADJUSTED WHEN S10 AND S14 ARE ADJUSTED

Microswitch - Adjustment
Figure 206

2) Measure the continuity between the pins shown for each switch and do these adjustments as required.
   a) If an open circuit exists:
      Push the locking channel on the microswitch pack and turn the adjustment bolt clockwise just until a closed circuit exists. Continue to turn the adjustment bolt clockwise the minimum amount necessary to lock the bolt in the channel.
   b) If a closed circuit exists, go on to step 2.

(b) Step 2:
1) Align the left and right autothrottle CAM's to the side frame pointers as directed in step 2, Fig. 205, for switches S1, S2, S3, S5, S6 and S7. (Align the quadrant decals with the pointers on the side frame.)
2) Measure the continuity between the pins shown for each switch and do these adjustments as required.
   a) If a closed circuit exists:
      Push the locking channel on the microswitch pack and turn the adjusting bolt counterclockwise just until an open circuit exists. Continue to turn the adjustment bolt counterclockwise the minimum amount necessary to lock the bolt in the channel.

      NOTE: The final adjustment must be made by turning the adjustment bolt counterclockwise.

   b) If an open circuit exists:
      Push the locking channel on the microswitch pack and turn the adjustment screw clockwise until a closed circuit exists. Then turn the adjusting bolt counterclockwise just until an open circuit exists. Continue to turn the adjustment bolt counterclockwise the minimum amount necessary to lock the bolt in the channel.

      NOTE: The final adjustment must be made by turning the adjustment bolt counterclockwise.

S 822-098

(5) Do these steps to adjust switches S10, S14, S17, and S18 (Fig. 206).

NOTE: Switches S10, S14, S17, and S18 must be adjusted in the sequence shown.

NOTE: Switches S10/S17 and S14/S18 are mechanically paired together, use a common adjusting bolt (Fig. 202).

NOTE: Switches S10/S14 are electrically paired together and must be adjusted in the sequence shown (Fig. 204).

(a) Step 1:
1) Align the left and right A/T CAM's to the side frame pointers as directed in step 1, Fig. 206, for switches S10, S14, S17 and S18.
2) Measure the continuity between the pins shown for switches S17 and S18 and do these adjustments as required.
a) If an open circuit exists:
Push the locking channel on the microswitch pack and turn the adjustment bolt counterclockwise just until a closed circuit exists. Continue to turn the adjustment bolt counterclockwise the minimum amount necessary to lock the bolt in the channel.
b) If a closed circuit exists, go on to the next instruction.
3) Measure the continuity between the pins shown for switches S10 and S14 and do these adjustments as required.

NOTE: Switches S10 and S14 are adjusted simultaneously.

a) If an open circuit exists:
Push the locking channel on the microswitch pack and turn the S10 adjustment bolt one turn then turn the S14 adjustment bolt one turn in the counterclockwise direction. Continue this sequence just until a closed circuit exists for both switches. Continue to turn the adjustment bolts counterclockwise the minimum amount necessary to lock the bolt in the channel. Go on to step 2.
b) If a closed circuit exists, go on to step 2.

(b) Step 2:
1) Align the left and right A/T CAM's to the pointer on the side frame as directed in step 2, Fig. 206, for switch S10.

2) Measure the continuity between the pins shown for switch S10 and do these adjustments as necessary.

NOTE: Switch S17 is adjusted when switch S10 is adjusted.

a) If a closed circuit exists:
Push the locking channel on the microswitch pack and turn the adjustment bolt counterclockwise until an open circuit exists. Then turn the adjustment bolt clockwise just until a closed circuit exists. Continue to turn the adjustment bolt clockwise the minimum amount necessary to lock the bolt in the channel. Go on to step 3.

NOTE: Final adjustment must be made from the clockwise direction.

b) If an open circuit exists:
Push the locking channel on the microswitch pack and turn the adjustment bolt clockwise just until a closed circuit exists. Continue to turn the adjustment bolt clockwise the minimum amount necessary to lock the bolt in the channel. Go on to step 3.

NOTE: Final adjustment must be made from the clockwise direction.

(c) Step 3:
1) Align the left and right A/T CAM's to the side frame pointer as directed in step 3, Fig. 206, for switch S14.
2) Measure the continuity between the pins shown for switch S14 and do these adjustments as necessary.

NOTE: Switch S18 is adjusted when switch S14 is adjusted.

a) If a closed circuit exists:
Push the locking channel on the microswitch pack and turn the adjustment bolt counterclockwise until an open circuit exists. Then turn the adjustment bolt clockwise just until a closed circuit exists. Continue to turn the adjustment bolt clockwise the minimum amount necessary to lock the bolt in the channel.

NOTE: Final adjustment must be made from the clockwise direction.

b) If an open circuit exists:
Push the locking channel on the microswitch pack and turn the adjustment bolt clockwise just until a closed circuit exists. Continue to turn the adjustment bolt clockwise the minimum amount necessary to lock the bolt in the channel.

NOTE: Final adjustment must be made from the clockwise direction.

S 822-004

(6) Do these steps to adjust switches S8, S11, S12, S16, S21, and S22 (Fig. 206).

NOTE: Switches S08, S11 and switches S12, S16 (or S21/S22 as installed) must be adjusted in the sequence shown.

NOTE: Switch combinations S08/S16 and S11/S12 (or S08/S22 and S11/S21 as installed) are mechanically paired together and use a common adjusting bolt, see Fig. 202.

NOTE: Switches S12 and S16 are electrically paired together and must be adjusted in the sequence shown (Fig. 204).

(a) Step 1:
1) Align the left and right A/T CAM's to the side frame pointers as directed in step 1, Fig. 206, for switches S08, S11, and S12, S16, (or S21, S22 as installed).
2) Measure the continuity between the pins shown for switches S12, S16, as installed, and do these adjustments as required.

NOTE: Switches S12 and S16 are adjusted at the same time.

a) If an open circuit exists:
Push the locking channel on the microswitch pack and turn both the S12 and S16 adjusting bolts one turn each in the clockwise direction. Then check the pin continuity for a closed circuit. Continue this sequence just until a closed circuit exists for both switches. Continue to turn the adjustment bolts clockwise the minimum amount necessary to lock the bolt in the channel.

b) If a closed circuit exists, go on to the next instruction and do the continuity check for switches S08 and S11.

3) Measure the continuity between the pins shown for switches S08, S11 and do these adjustments as necessary.
   a) If an open circuit exists:
      Push the locking channel on the microswitch pack and turn the adjusting bolt in the clockwise direction until a closed circuit exists.

   b) If a closed circuit exists, go on to step 2.

(b) Step 2:
   1) Align the left and right A/T CAM's to the pointer on the side frame as directed in step 2, Fig. 206, for switches S11, and S12 (or S21 as installed).
   2) Measure the continuity between the pins shown for each switch and do these adjustments as required.
      a) If a closed circuit exists:
         Push the locking channel on the microswitch pack and turn the adjustment bolt clockwise until an open circuit exists. Then turn the adjustment bolt counterclockwise just until a closed circuit exists. Continue to turn the adjustment bolt counterclockwise the minimum amount necessary to lock the bolt in the channel. Go on to step 3.

         NOTE: Final adjustment must be made from the counterclockwise direction.

      b) If an open circuit exists:
         Push the locking channel on the microswitch pack and turn the adjustment bolt counterclockwise just until a closed circuit exists. Continue to turn the adjustment bolt counterclockwise the minimum amount necessary to lock the bolt in the channel. Go on to step 3.

         NOTE: Final adjustment must be made from the counterclockwise direction.

(c) Step 3:
   1) Align the left and right A/T CAM's to the side frame pointer as directed in step 3, Fig. 206, for switches S08 and S16 (or S22 as installed).

2) Measure the continuity between the pins shown for each switch and do these adjustments as required.
   a) If a closed circuit exists:
      Push the locking channel on the microswitch pack and turn the adjustment bolt clockwise until an open circuit exists. Then turn the adjustment bolt counterclockwise just until a closed circuit exists. Continue to turn the adjustment bolt counterclockwise the minimum amount necessary to lock the bolt in the channel.

      NOTE: Final adjustment must be made from the counterclockwise direction.

   b) If an open circuit exists:
      Push the locking channel on the microswitch pack and turn the adjustment bolt counterclockwise just until a closed circuit exists. Continue to turn the adjustment bolt counterclockwise the minimum amount necessary to lock the bolt in the channel.

      NOTE: Final adjustment must be made from the counterclockwise direction.

F. Autothrottle Microswitch Pack - Post Installation Test

NOTE: This procedure has two tasks:
Microswitch Pack - Microswitch Adjustment Check
Microswitch Pack - Load Shed Function Check

S 762-049
(1) Do this task: Micoswitch Adjustment Check (Fig. 207)

NOTE: This procedure makes sure that the microswitches are adjusted properly.

NOTE: You must do this test to check for correct microswitch operation.

(a) Set the left and right Autothrottle Cam's to the quardrant position for the first grouping of microswitches shown in Fig. 207.

(b) For each switch in the grouping measure the pin continuity for both the OPEN and CLOSED conditions at the connector and pin locations shown (Fig. 207).

NOTE: NOT ALL SWITCHES REQUIRE BOTH AN OPEN AND CLOSED CONTINUITY CHECK.

(c) Set the left and right Autothrottle Cam's to the quadrant positions shown for the next grouping of microswitches. Measure the pin continuity shown for each switch.

(d) Repeat this test for each grouping of microswitches shown in Fig. 207

NOTE: If the microswitches fail this test. Remove the microswitch pack and check the microswitches initial adjustment settings (Refer to the vendor recommendation).

S 722-075

(2) Do this task: Microswitch Pack - Load Shed Function Check

NOTE: This test uses either the APU or the engine generators for power. Make sure that you take proper precautions for running engines.

S 422-006

(3) Connect all microswitch assembly electrical connectors.

S 862-050

(4) Make sure that these circuit breakers on the P11 panel are closed:
(a) 11T4, UTIL BUS L
(b) 11T31, UTIL BUS R

S 862-051

(5) Make sure that these circuit breakers on the main power distribution panel, P6, are closed:
(a) 6B1, GEN CONT UNIT L

SWITCHES TO EXAMINE	AUTOTHROTTLE CAM POSITION ±0.5°		PIN CONTINUITY (CONNECTOR, PINS) [2]	
	LEFT	RIGHT	OPEN	CLOSED
S1 S2 S3 S5 S6 S7	4.5° FWD	4.5° FWD	P1 6,7 P1 11,12 P1 14,15 P2 9,10 P2 1,2 P2 4,5	----------------- P1 11,13 P1 14,16 ----------------- P2 1,3 P2 4,6
S1 S2 S3 S5 S6 S7	1.0° FWD	1.0° FWD	----------------- P1 11,13 P1 14,16 ----------------- P2 1,3 P2 4,6	P1 6,7 P1 11,12 P1 14,15 P2 9,10 P2 1,2 P2 4,5
S10 & S14 S17 S18	9.5° FWD	9.5° FWD	P1 2,3 P1 4,5 P2 7,8	P1 1,3 P1 5,17 P2 8,11
S10 & S14 S17	5.5° FWD	9.5° FWD	P1 2,3 P1 5,17	P1 1,3 P1 4,5
S10 & S14 S18	9.5° FWD	5.5° FWD	P1 2,3 P2 8,11	P1 1,3 P2 7,8
S10 & S14	5.5° FWD	5.5° FWD	P1 1,3	P1 2,3
S8 S11 S21 } [1] S22	4.5° REV	4.5° REV	P2 21,22 P1 21,22 P3 5,6 P4 5,6	P2 21,23 P1 21,23 P3 5,8 P4 5,8
S8 S11 S21 } [1] S22	4.5° REV	1.5° REV	P2 21,23 P1 21,22 P3 5,6 P4 5,8	P2 21,22 P1 21,23 P3 5,8 P4 5,6
S8 S11 S21 } [1] S22	1.5° REV	4.5° REV	P2 21,22 P1 21,23 P3 5,8 P4 5,6	P2 21,23 P1 21,22 P3 5,6 P4 5,8
S8 S11 S21 } [1] S22	1.5° REV	1.5° REV	P2 21,23 P1 21,23 P3 5,8 P4 5,8	P2 21,22 P1 21,22 P3 5,6 P4 5,6

[1] SAS 050-999; MTH 275-999

[2] SEE FIG. 204 FOR WIRING SCHEMATIC

Microswitch Adjustment - Check
Figure 207

(b) 6B2, GEN CONT UNIT R
(c) 6B3, GEN CONT UNIT APU
(d) 6B4, BUS PWR CONT UNIT

S 282-052
(6) Make sure that only a single electrical source (APU generator, Left IDG or Right IDG) is used to supply power to the airplane buses.

S 862-114
(7) Push the L and R UTILITY BUS switches on the electrical system control panel (on the P5 panel) to ON (latched-in position).

S 282-054
(8) Make sure that the OFF lights in the switches do not come on.

S 862-109

WARNING: THE THROTTLES WILL MOVE IN THE STEPS THAT FOLLOW. TAKE CORRECT PRECAUTIONS WHEN THE ENGINES ARE RUNNING OR INJURY TO PERSONS AND/OR DAMAGE TO EQUIPMENT COULD OCCUR.

(9) Move the left thrust lever to approximate 1/4 throttle (or equivalent) and the right thrust lever to idle position.

S 282-058
(10) Make sure that the OFF lights in the L and R UTILITY BUS switches do not come on.

S 862-110
(11) Move the left thrust lever back to the idle position and move the right thrust lever to 1/4 throttle (or equivalent).

S 282-060
(12) Make sure that the OFF lights in the L and R UTILITY BUS switches do not come on.

S 862-111
(13) Move the left thrust lever to 1/4 throttle (or equivalent) and keep the right thrust lever at 1/4 throttle (or equivalent).

S 282-107

(14) Make sure that the OFF lights in the L and R UTILITY BUS switches come on.

S 862-112

(15) Move the L and R thrust levers to the aft stops.

G. Put the Airplane Back to Its Initial Condition

S 862-113

(1) Return the two thrust levers to the idle position.

S 862-077

(2) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) 11F15, TMC DC
(b) 11G11, AUTO SPEEDBRAKE
(c) 11M5, LEFT ENGINE EEC DISCRETES
(d) 11M32, RIGHT ENGINE EEC DISCRETES
(e) 11U12, AUTOBRKS ANTISKID TEST/IND 1
(f) 11U21, AUTOBRKS ANTISKID TEST/IND 2

S 842-064

(3) Do the activation procedure for the Spoiler/Speedbrake System if you did the deactivation procedure (AMM 27-61-00/201).

S 842-065

(4) Restore Thrust Reverser System to normal (AMM 78-31-00/201).

S 862-066

(5) Remove electrical power if it is not necessary (AMM 24-22-00/201).

AUTOTHROTTLE BRAKE PACK ASSEMBLY - REMOVAL/INSTALLATION

1. General
   A. SAS 050-051, 150-156, 162-166;
      MTH 275-276;
      The autothrottle brake pack assembly is below the flight compartment floor under the aisle control stand. The brake pack assembly has two brake assemblies on the same shaft. The shaft connects with a gearbox and a servomotor generator through the quill shaft. The access door, 113AL, for the forward equipment compartment gives access to the brake pack assembly.
   B. SAS 052-149, 157-161, 167-999;
      MTH 277-999;
      The autothrottle brake pack assembly is below the flight compartment floor under the aisle control stand. The brake pack assembly has three brake assemblies on a splined shaft. The shaft connects with a gearbox and a servomotor generator. The access door, 113AL, for the forward equipment compartment gives access to the brake pack assembly.

TASK 22-32-07-004-001

2. Autothrottle Brake Pack Assembly Removal Procedure (Fig. 401)
   A. Consumable Material
      (1) D00633 Grease - BMS 3-33 (Preferred)
      (2) D00013 Grease - MIL-PRF-23827 (Supersedes MIL-G-23827) (Alternate)
   B. Parts
      (1) SAS 050-051, 150-156, 162-166;
          MTH 275-276;
          Refer to the table that follows.

AMM		NOMENCLATURE	AIPC		
FIG	ITEM		SUBJECT	FIG	ITEM
401A	15	Brake Pack Assembly	22-32-01	02	420
	24	Shaft			590
	25	Locknut, Bearing			425
	26	Support Bracket Assembly			435
	27	Brake Assembly			480
	28	Brake Assembly			475

	29	Support Bracket Assembly			440
	30	Spacer			465
	31	Spacer			585

C. References
(1) AMM 06-41-00/201, Section 41 Access Doors and Panels
(2) AMM 22-32-01/401, Autothrottle Servomotor Generator and Gearbox

D. Access
(1) Location Zones
121/122 Forward Cargo Compartment
211/212 Flight Compartment

E. Prepare for Removal

S 864-002
(1) Open this circuit breaker on the overhead circuit breaker panel, P11, and attach a DO-NOT-CLOSE tag:
(a) 11F14, TMC AC

S 864-003
(2) Attach DO-NOT-OPERATE tags to the thrust levers.

S 014-004
(3) Open the 113AL access panel (AMM 06-41-00/201).

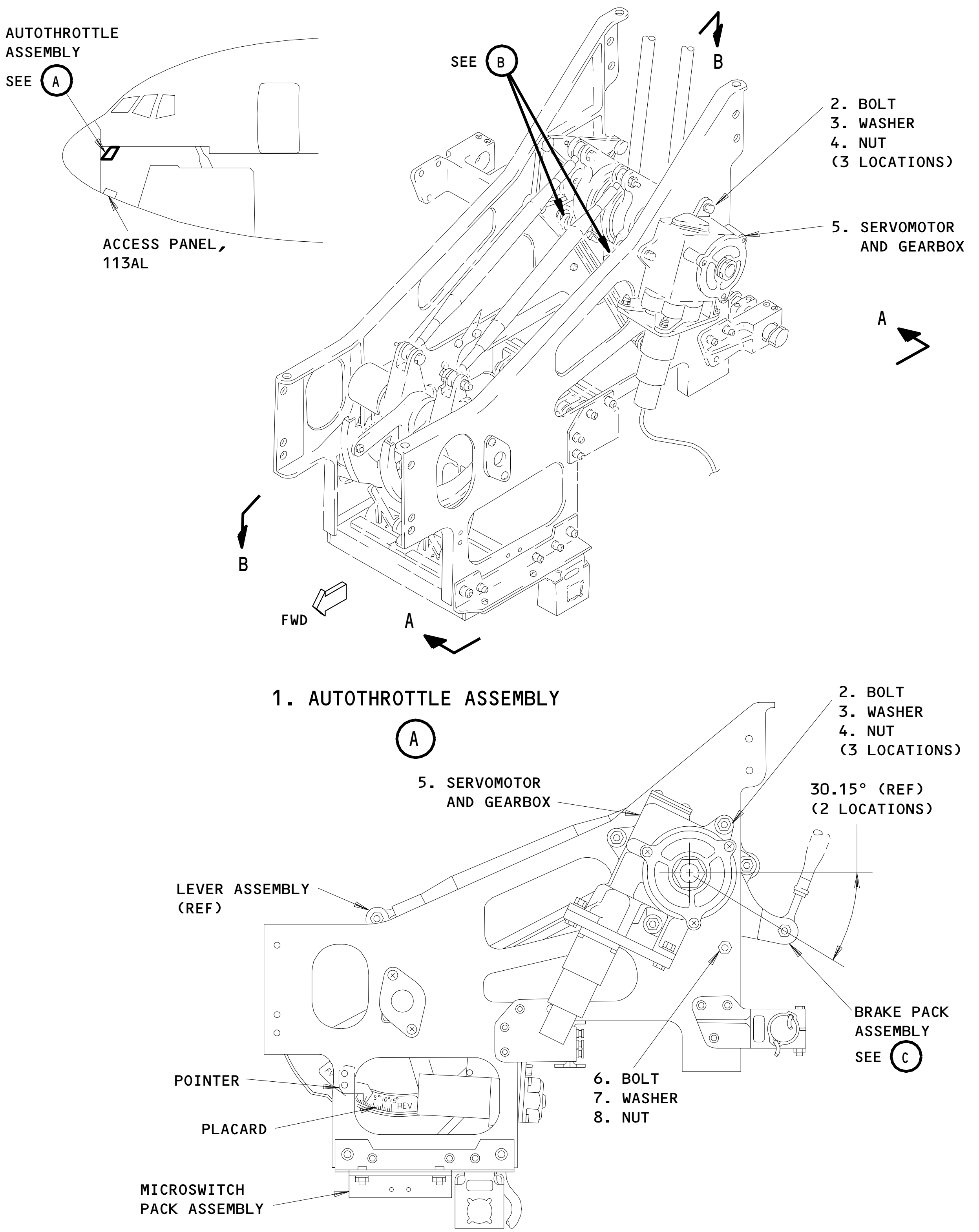
F. SAS 050-051, 150-156, 162-166;
MTH 275-276;
Remove Brake Pack Assembly (15) (Fig. 401A).

NOTE: This removal procedure is applicable to A/T Installations with two Brake Assemblies installed.

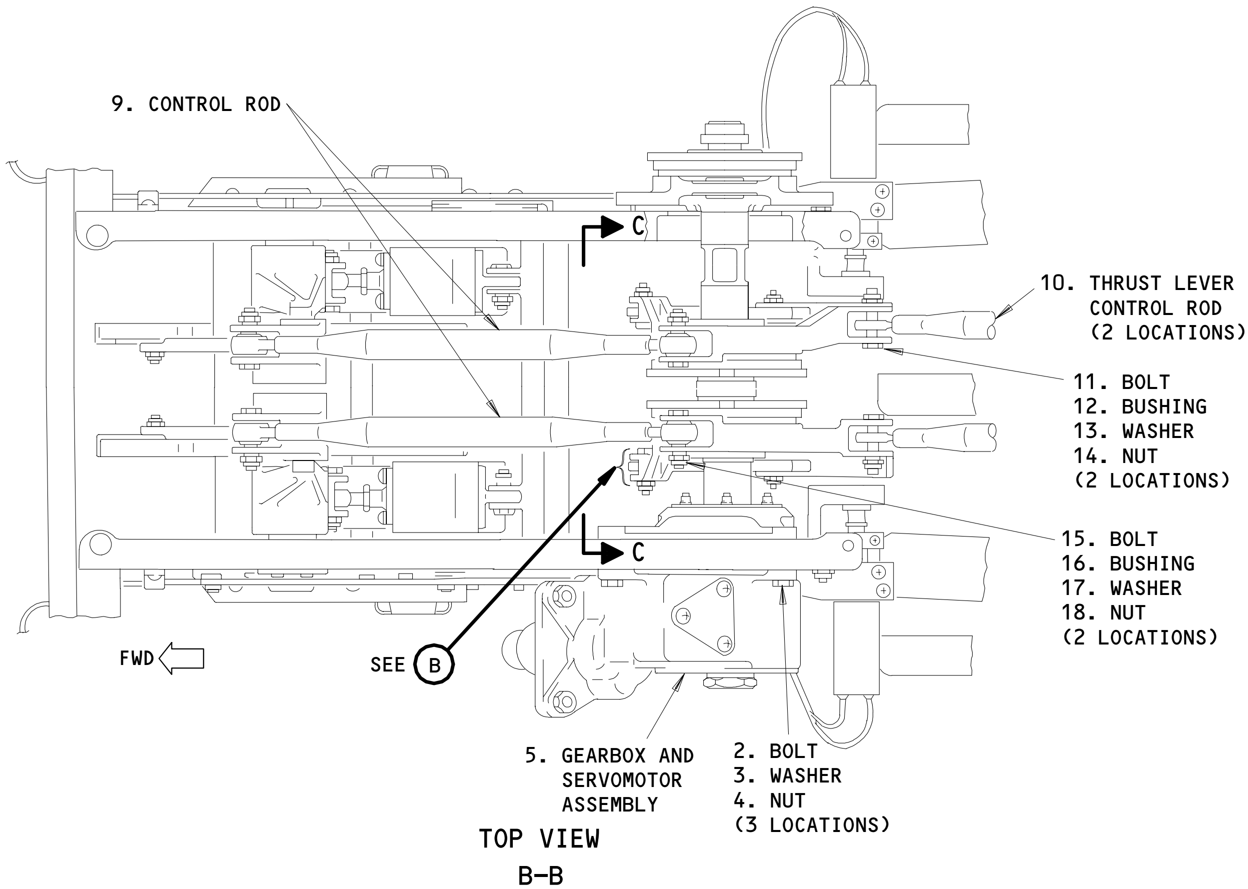
NOTE: Brake Assemblies (27, 28) are components of the Brake Pack Assembly (15).

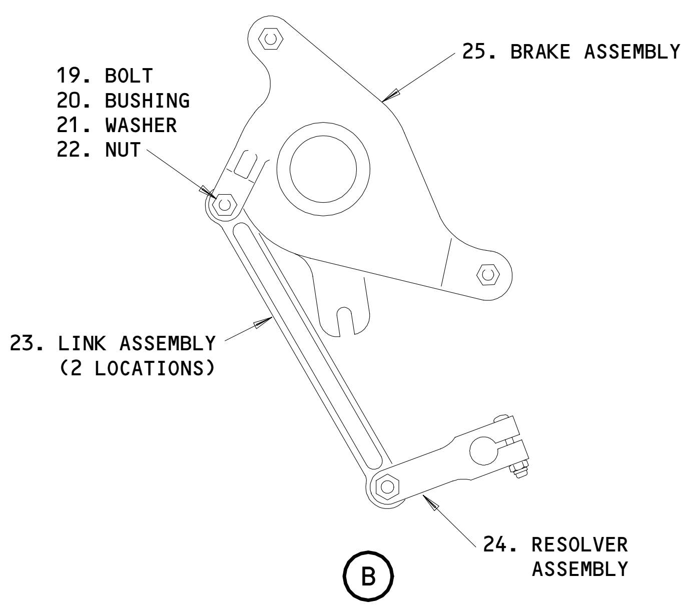
S 034-010
(1) SAS 150-154 POST-SB 22-81;
SAS 050-051, 162-166;
MTH 275-276;
remove the screws (33) and expander cones (32) from the ends of the quill shaft (4).

S 034-011
(2) Remove the bolts (2) and washers (3) from the gearbox and servo assembly (5).

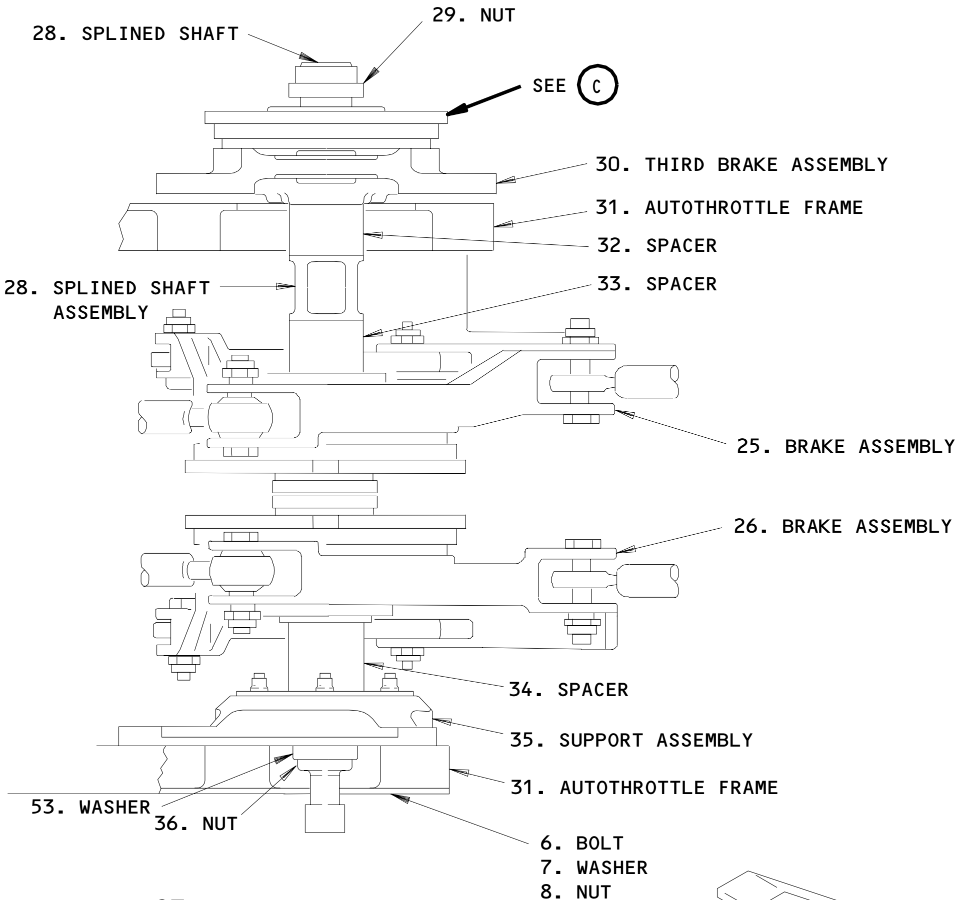


Autothrottle Brake Pack Assembly - Installation
Figure 401 (Sheet 1)

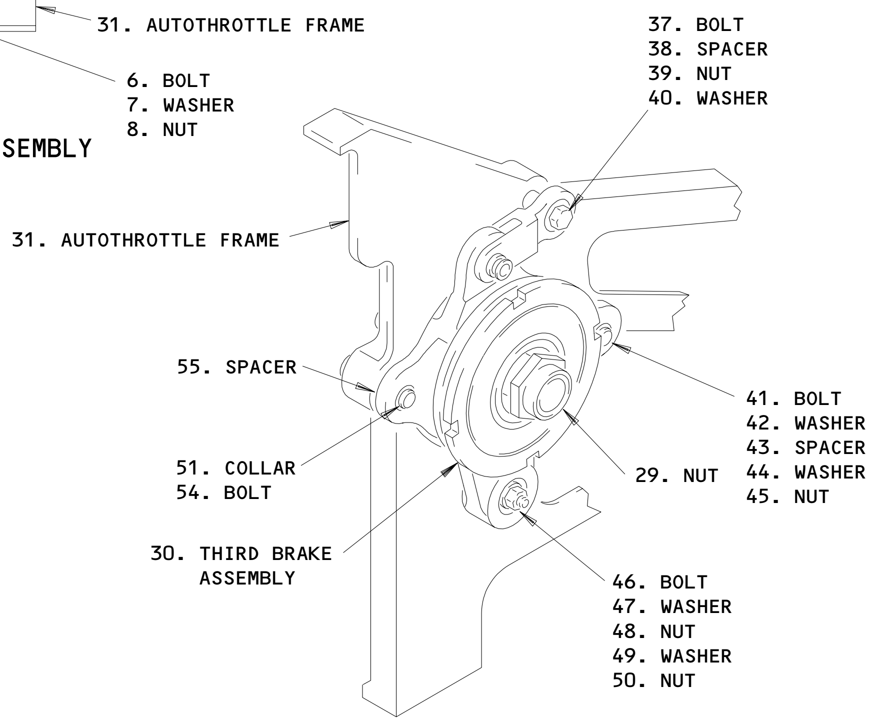




Autothrottle Brake Pack Assembly - Installation
Figure 401 (Sheet 2)

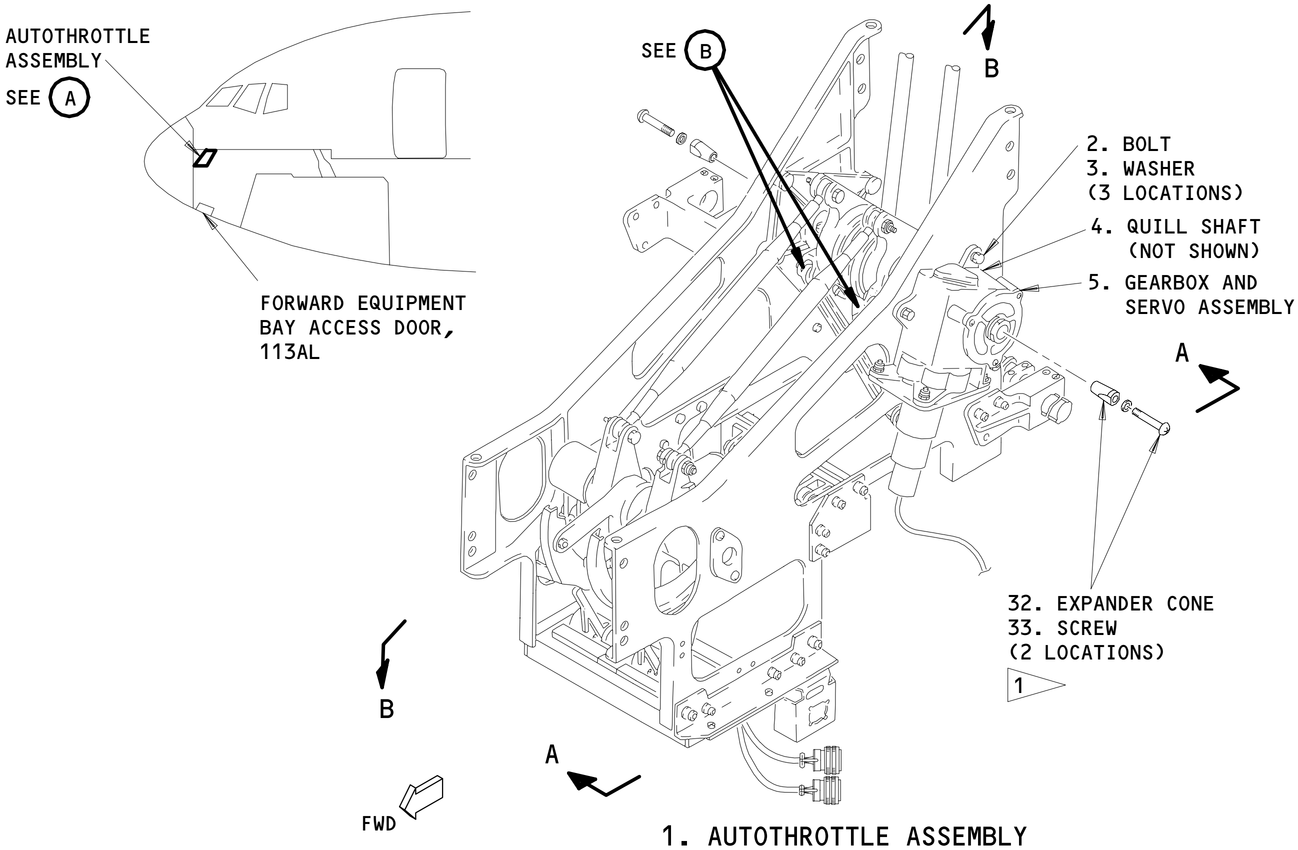


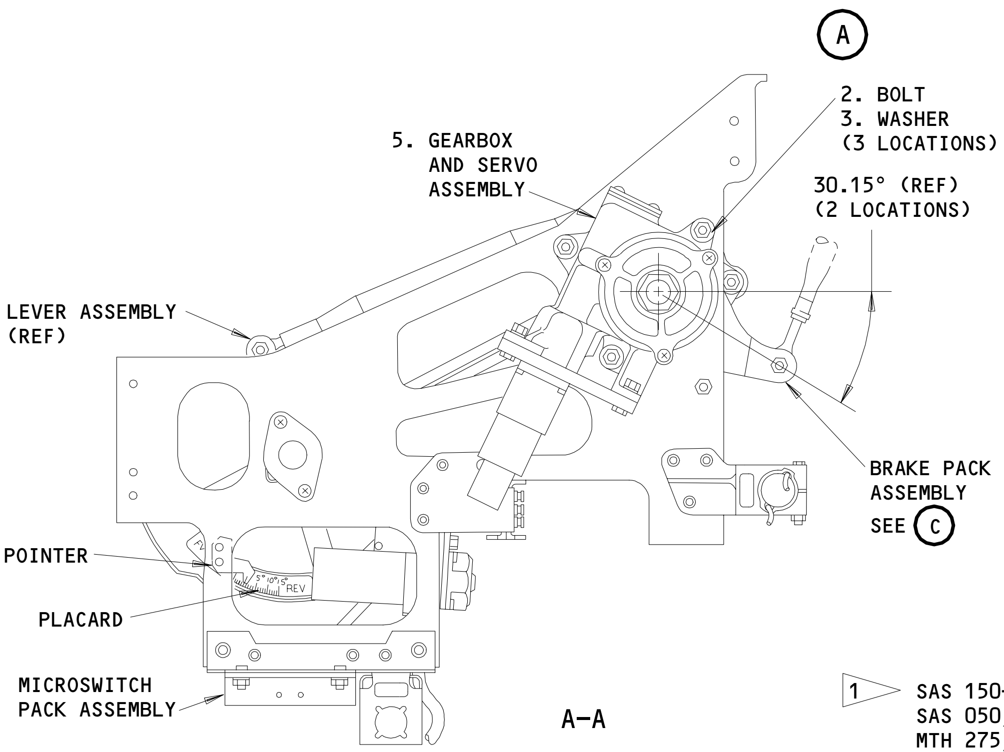
27. BRAKE PACK ASSEMBLY
C-C



30. THIRD BRAKE ASSEMBLY DETAIL
C

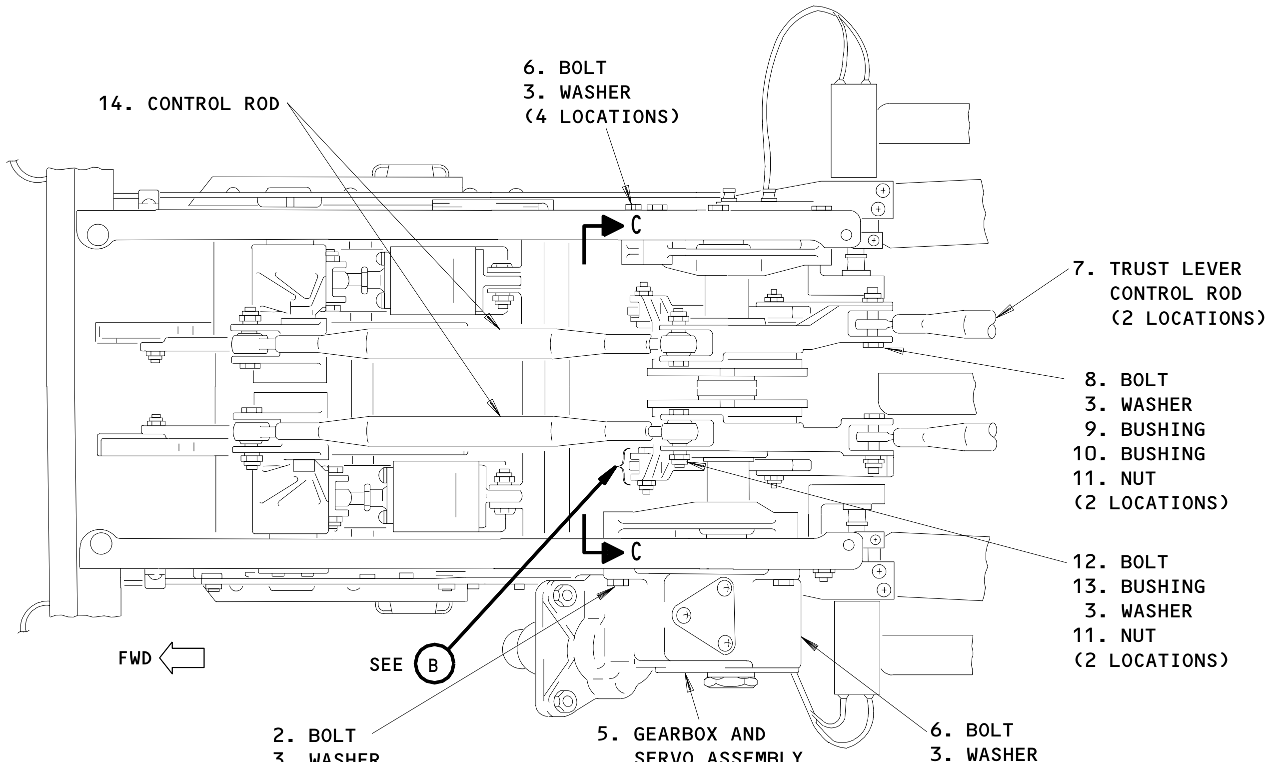
Autothrottle Brake Assembly - Installation
Figure 401 (Sheet 3)



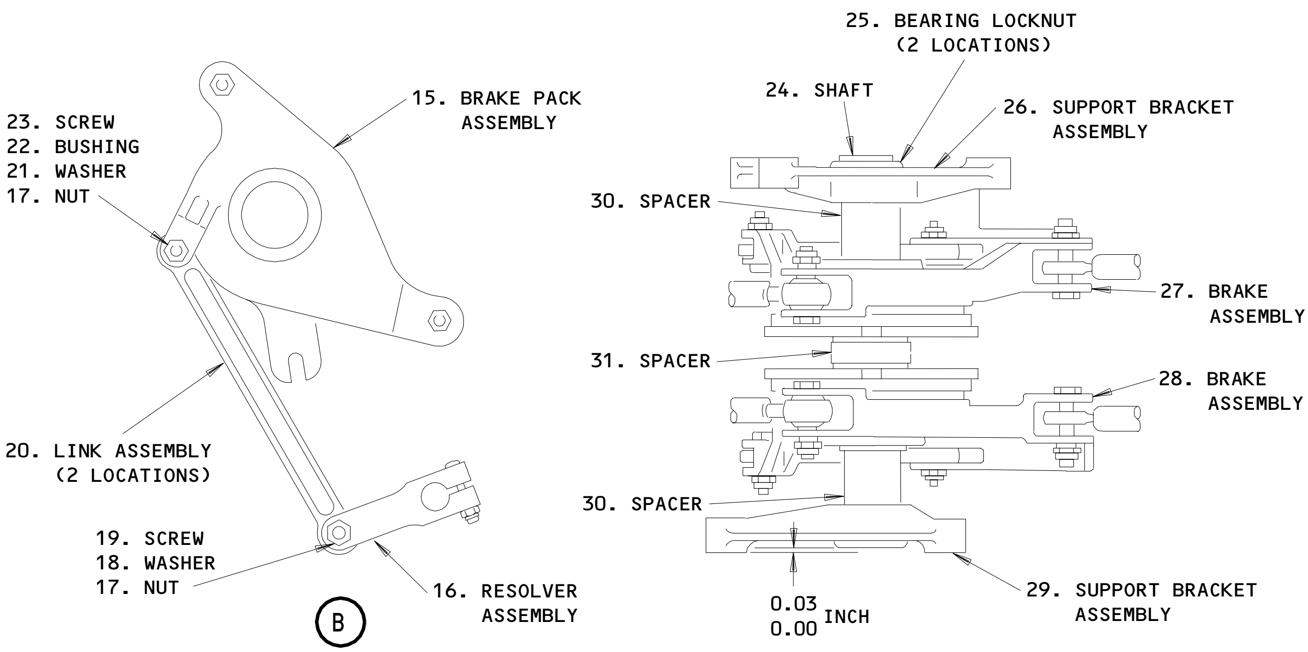


1 SAS 150-154 WITH SB 22-81, AND
SAS 050,051,162-166
MTH 275,276

Autothrottle Brake Pack Assembly - Installation
Figure 401A (Sheet 1)



TOP VIEW
B-B



15. BRAKE PACK ASSEMBLY
C-C

Autothrottle Brake Pack Assembly - Installation
Figure 401A (Sheet 2)

S 864-012
(3) Remove the autothrottle servomotor generator and gearbox assembly (5) (AMM 22-32-01/401).

S 034-013
(4) Pull the quill shaft assembly (4) out of the brake pack assembly (15).

S 034-014
(5) Turn one lever assembly, rod assembly (14), and brake pack assembly (15) until the rod end and the bolts are not aligned.

S 034-015
(6) Remove the nuts (11), washers (3), bushings (13), bolts (12) and rod assemblies (14) from the lever assemblies and the brake pack assembly (15).

S 034-016
(7) Remove the nuts (17), washers (21), bushing (22), screws (23) and link assemblies (20).

NOTE: Do not remove the bearings from the link assemblies (20) unless repair or replacement is necessary.

S 034-017
(8) Remove the nuts (11), washers (3), bushings (9, 10), bolts (8) and thrust lever rod assemblies (7) from the brake pack assembly (15).

S 034-018
(9) Remove the bolts (6) and washers (3) from the support brackets (26, 29).

S 024-019
(10) Remove the brake pack assembly (15) from between the frames.

G. SAS 052-149, 157-161, 167-999;
MTH 277-999;
Remove the Brake Assemblies (25, 26, 30) (Fig. 401).

NOTE: This removal procedure is applicable to A/T Installations with three Brake Assemblies installed.

NOTE: Brake Assemblies (25, 26) are components of the Brake Pack Assembly (27).

S 034-053
(1) Move one thrust lever so the bolts (11) that hold the thrust lever control rods (10) to the brake assemblies (25, 26) are not aligned.

S 034-054
(2) Remove the bolts (11), bushings (12), washers (13), and nuts (14) that hold the thrust lever control rods (10) to the brake assemblies (25, 26).
(a) Move the control rods away from the brake assemblies.

S 034-055
(3) Remove the bolts (15), bushings (16), washers (17), and nuts (18) that hold the control rods (9) to the brake assemblies (25, 26).
(a) Move the control rods away from the brake assemblies.

S 034-056
(4) Remove the screws (19), bushings (20), washers (21), and nuts (22) that hold the link assembly (23) to the brake assemblies (25, 26).
(a) Move the resolver control rods away from the brake assemblies.

NOTE: Do not remove the bearings from the control rods (9, 10, 23) unless repair or replacement is necessary.

S 034-057
(5) Remove the nut (29) from the end of the splined shaft (28).

S 034-058
(6) Remove these fasteners to remove the third brake assembly (30) from the autothrottle frame (31).
(a) Remove the bolt (37), spacer (38), washer (40), and nut (39).
(b) Remove collar (51), bolt (54) and spacer (55).

(c) Remove the bolt (41), washers (42, 44), spacer (43), and nut (45).
(d) Remove the bolt (46), washers (47, 49) and nuts (48, 50).

S 034-061
(7) Remove the third brake assembly (30) and the spacer (32) from the autothrottle frame (31).

S 034-062
(8) Remove the three bolts (2), washers (3), and nuts (4) that hold the servomotor and gearbox (5) and the support assembly (35) to the autothrottle frame (31).

S 034-063
(9) Remove the servomotor and gearbox (5) from the autothrottle frame (31).

S 034-064
(10) Remove the nut (36) from the end of the splined shaft (28).

S 034-065
(11) Remove the splined shaft (28) as follows:
(a) To remove the shaft, move the shaft through the end of the autothrottle frame (31).

CAUTION: HOLD THE SPACERS AND BRAKE ASSEMBLIES AS YOU REMOVE THE SHAFT. THEY WILL FALL AS THEY MOVE OFF THE SHAFT. DAMAGE TO EQUIPMENT CAN OCCUR IF THEY FALL.

(b) Hold the spacers (33, 34) and brake assemblies (25, 26) as they move off the shaft.

S 034-067
(12) Remove the brake pack assembly (27) - support bracket (35) from the autothrottle frame (31) as follows:
(a) Remove the bolt (6), washer (7), and nut (8) that holds the support bracket (35) to the autothrottle frame (31).

(b) Remove the support bracket (35).

H. SAS 050-051, 150-156, 162-166;
MTH 275-276;
Brake Assembly replacement procedure (Fig. 401A).

NOTE: This procedure is applicable to A/T Installations with two Brake Assemblies installed.

NOTE: This procedure gives disassemble and assemble instructions for the Brake Pack Assembly (15).

S 034-020

(1) Do these steps to remove brake assemblies (27, 28) from the brake pack assembly (15) .
   (a) Remove the nut (25) from either end of the brake pack assembly (15).
   (b) Remove the spacers (30, 31) and brake assemblies (27, 28).

S 434-021

(2) Do these steps to install the brake assemblies (27, 28) in the brake pack assembly (15).
   (a) Before assembly, cover all the mating surfaces with grease.
   (b) Move the spacer (31) onto the shaft (24), then the brake assembly (28) on one side and the brake assembly (27) on the other side (see fig. 401A).
   (c) Move one spacer (30) onto each side of the shaft against the brake assemblies.
   (d) On the side with the brake assembly (28), attach a support bracket assembly (29) and nut (25).
   (e) On the side with the brake assembly (27), attach a support bracket assembly (26) and nut (25).
   (f) Tighten the nuts (25) to 200-220 pound-inches (22.6-24.9 Newton-meters) above the run-on torque.
   (g) Make sure the end gap between the shaft (24) and the support bracket assemblies (26, 29) is less than .03 inches (0.762 millimeters) .

TASK 22-32-07-404-022

3. Install the Autothrottle Brake Pack Assembly (Fig. 401)
   A. Consumable Material
      (1) D00633 Grease - BMS 3-33 (Preferred)

(2) D00013 Grease - MIL-PRF-23827 (Supersedes MIL-G-23827) (Alternate)

B. Parts

(1) SAS 050-051, 150-156, 162-166;
MTH 275-276;
Refer to the table that follows.

AMM			AIPC		
FIG	ITEM	NOMENCLATURE	SUBJECT	FIG	ITEM
401A	1	Autothrottle Assembly	22-32-01	02	75
	2	Bolt			335
	3	Washer			15, 120 340, 415
	4	Quill Shaft			375
	5	Gearbox and Servo Assembly			330
	6	Bolt	22-32-01	02	410
	7	Control Rod, Thrust Lever	76-11-00	01	55
	8	Bolt	22-32-01	02	10
	9	Bushing			26
	10	Bushing			27
	11	Nut			20, 125
	12	Bolt			110
	13	Bushing			135
	14	Control Rod			145
	15	Brake Pack Assembly			420
	16	Resolver Assembly			600, 605
	17	Nut			395
	18	Washer			15
	19	Screw			380
	20	Link Assembly			405
	21	Washer			390
	22	Bushing			400
	23	Screw			385
	24	Shaft			590
	25	Locknut, Bearing			425
	26	Support Bracket Assembly			435
	27	Brake Assembly			480
	28	Brake Assembly			475
	29	Support Bracket Assembly			440
	30	Spacer			465
	31	Spacer			585
	32	Expander Cones			315
	33	Screw			320
	34	Washer			325

(2) SAS 052-149, 157-161, 167-999;
MTH 277-999;
Refer to the table that follows.

AMM		NOMENCLATURE	AIPC		
FIG	ITEM		SUBJECT	FIG	ITEM
401	1	Autothrottle Assembly	22-32-01	02A	95
	2	Bolt			375,380
	3	Washer			385
	4	Nut			390
	5	Gearbox and Servomotor Assembly			370
	6	Bolt			455
	7	Washer			460
	8	Nut			465
	9	Control Rod			140
	10	Control Rod, Thrust Lever	76-11-00	01	55
	11	Bolt	22-32-01	02A	5
	12	Bushing			20,25
	13	Washer			10
	14	Nut			15
	15	Bolt			120
	16	Bushing			135
	17	Washer			125
	18	Nut			130
	19	Bolt			385,425
	20	Bushing			400,445
	21	Washer			390,430
	22	Nut			395,440
	23	Link Assembly			405,450
	24	Resolver Assembly			635,640
	25	Brake Assembly			530
	26	Brake Assembly			525
	27	Brake Pack Assembly			470
	28	Splined Shaft Assembly			615
	29	Nut			480
	30	Third Brake Assembly			245

	32	Spacer			365
	33	Spacer			515
	34	Spacer			520
	35	Support Assembly			485
	36	Nut			480
	37	Bolt			175
	38	Spacer			210
	39	Nut			230
	40	Washer			225
	41	Bolt			185
	42	Washer			205
	43	Spacer			215
	44	Washer			225
	45	Nut			195
	46	Bolt			180
	47	Washer			200
	48	Nut			195
	49	Washer			225
	50	Nut			230
	51	Collar			235
	53	Washer			475
	54	Bolt			195
	55	Spacer			215

C. References
  (1) AMM 06-41-00/201, Section 41 Access Doors and Panels
  (2) AMM 22-00-02/201, Autoflight BITE
  (3) AMM 22-32-01/401, Autothrottle Servomotor Generator and Gearbox
  (4) AMM 24-22-00/201, Electrical Power - Control
  (5) AMM 27-61-00/201, Spoiler/Speedbrake Control System
  (6) AMM 76-11-00/501, Engine Control System

D. Access
  (1) Location Zones
      121/122   Forward Cargo Compartment
      211/212   Flight Compartment

E. SAS 050-051, 150-156, 162-166;
   MTH 275-276;

Install Brake Pack Assembly (15) (Fig. 401A).

NOTE: This procedure is applicable to A/T Installations with two Brake Assemblies installed.

S 424-023

(1) Put the brake pack assembly (15) into position between the frames, and install support brackets (26, 29) with the bolts (6) and washers (3).

S 434-024

(2) Install the link assemblies (20) between the brake pack assembly (15), screw (23), bushing (22), washer (21), and nut (17).

S 434-025

(3) Turn the lever assembly to offset the lever assemblies lugs.

S 434-026

(4) Install the rod assemblies (14) between the lever assemblies and the brake pack assembly (15) with the bolts (12), bushings (13), washers (3), and nuts (11).

S 284-096

(5) Do the travel check as follows:

(a) Disconnect the interlock actuators from the stop crank assemblies.

NOTE: Make sure the actuators do not fall onto the microswitch pack assembly during the travel check.

(b) Make sure the autothrottle turns freely through these minimum angles:

NOTE: Measure at the brake pack assembly (15) from the reference 30.15 degree position (Fig. 401A). Measure clockwise and counterclockwise from the left side of the autothrottle assembly (1).

1) Forward thrust: 53.6 degrees counterclockwise

2) Reverse thrust: 31.4 degrees clockwise

(c) Reconnect the actuators to the stop crank assemblies with the fasteners.

S 974-099

(6) SAS 150-154 POST-SB 22-81;
SAS 050-051, 162-166;
MTH 275-276;
do these steps as follows to find the run-on torque for the quill shaft (4):

(a) Lubricate the screw (33) with grease.

(b) Turn the screw (33) into the gearbox end of the quill shaft (4) without the expander cone (32).
(c) Tighten the screw (33) until the self-locking insert in the end of the quill shaft (4) starts to engage.
(d) With a dial torque wrench, measure the amount of torque necessary to turn the screw (33) after it has engaged the insert.
(e) This value is the run-on torque. Make sure to write it down.

NOTE: The run-on torque of the screws will increase as the screw is turned into the quill shaft.

(f) Remove the screw (33).

S 434-119
(7) Do these steps to install the quill shaft assembly (4).
(a) Cover the mating surfaces of the quill shaft assembly (4) with grease, before assembly.
(b) Insert the quill shaft assembly (4) into the brake pack assembly shaft (24) and push fully against the brake assembly (15).

S 434-125
(8) Do these steps to install the Gearbox/Servomotor assembly (5).
(a) Engage the splines of the quill shaft assembly (4) with the splines of the gearbox and servo assembly (5).
(b) Attach the gearbox and servo assembly (5) to the frame with the bolts (2) and washers (3) (Ref 22-32-01).

S 434-113
(9) SAS 150-154 POST-SB 22-81;
SAS 050-051, 162-166;
MTH 275-276;
do these steps as follows to install the expander cones (32) and screws (33) in the ends of the quill shaft (4):
(a) Lubricate the screw (33), expander cone (32), and the gear box end of the quill shaft (4) with grease.
(b) Install the expander cone (32) in the Gearbox/Servomotor end of the quill shaft (4) with the screw (33) and washer (34).
(c) Tighten the screw (33) to 5-8 pound-inches (0.56-0.90 newton-meters) more than the run-on torque.
(d) Lubricate the screw (33), expander cone (32), and the opposite end of the quill shaft (4).
(e) Install the expander cone (32) in the quill shaft (4) with the screw (33) and washer (34).
(f) Tighten the screw (33) to 12-15 pound-inches (1.4-1.7 newton-meters) torque.

S 434-126

(10) Install the thrust lever control rods (7) between the brake pack assembly (15) and attach with bolts (8), washers (3), bushings (9, 10) and nut (11).

F. SAS 052-149, 157-161 167-999;
MTH 277-999;
Install the Brake Assemblies (25, 26, 30) (Fig. 401).

NOTE: This procedure is applicable to A/T Installations with three Brake Assemblies installed.

NOTE: Brakes Assemblies (25, 26) are components of the Brake Pack Assembly (27).

S 644-068

(1) Before you assemble the autothrottle brake assembly, cover all the mating surfaces with grease.

S 434-069

(2) Install the support bracket (35) on the autothrottle frame (31) with the bolt (6), washer (7), and nut (8).

S 434-070

(3) Install the splined shaft (28), spacers (33, 34) and brake assemblies (25, 26) as follows:
   (a) Move the splined shaft (28) through the end of the autothrottle frame (31).
   (b) Install the spacer (34), brake assemblies (26, 25) and spacer (33) on the splined shaft (28) as you move the shaft through the end of the autothrottle frame (31).

S 434-071

(4) Install the washer (53) and nut (36) on the Gearbox/Servomotor end of the splined shaft (28).
   (a) Tighten the nut to 200 to 220 pound-inches (22.6-24.8 newton-meters) more than the run-on torque.

S 434-072

(5) Install the servomotor and gearbox (5) on the autothrottle frame (31) and support assembly (35) with three bolts (2), washers (3), and nuts (4).

S 434-073

(6) Install the third brake (30) and spacer (32) on the splined shaft (28) and attach with these fasteners:
   (a) Install the bolt (46), washer (47), and nut (48) on the brake side, and washer (49) and nut (50) on the autothrottle frame side.
   (b) Install the bolt (41), washer (42), spacer (43) and nut (45).

(c) Install the spacer (55), bolt (54) and collar (51).
(d) Install the bolt (37), spacer (38), washer (40) and nut (39).

S 434-076
(7) Install the nut (29) on the end of the splined shaft (28).
(a) Tighten the nut to 100-150 inch-pounds (11.3-16.9 newton-meters) more than the run on torque.

S 434-077
(8) Attach the link assembly (23) to the brake assemblies (25, 26) with bolts (19), bushings (20), washers (21) and nuts (22).

S 434-078
(9) Attach the thrust lever control rods (10) to the brake assemblies (25, 26) with bolts (11), bushings (12), washers (13) and nuts (14).

S 434-079
(10) Attach the control rods (9) to the brake assemblies (25, 26) with bolts (15), bushings (16), washers (17) and nuts (18).

S 714-027
(11) Do the travel check as follows:
(a) Disconnect the interlock actuators from the stop crank assemblies.

NOTE: Make sure the actuators do not fall onto the microswitch pack assembly during the travel check.

(b) Make sure the autothrottle turns freely through these minimum angles:

NOTE: Measure at the brake pack assembly (15) from the reference 30.15 degree position (Fig. 401). Measure clockwise and counterclockwise from the left side of the autothrottle assembly (1).

1) Forward thrust: 53.6 degrees counterclockwise
2) Reverse thrust: 31.4 degrees clockwise

(c) Reconnect the actuators to the stop crank assemblies with the fasteners.

S 434-127
(12) Install the thrust lever control rods (10) between the brake pack assembly (27) and attach with bolts (11), washers (13), bushings (12) and nut (14).

G. Autothrottle Brake Pack Assembly Test

S 864-039
(1) Supply electrical power (AMM 24-22-00/201).

S 864-040
(2) Remove the DO-NOT-CLOSE tag and close this circuit breaker on the P11 panel:
(a) 11F14, TMC AC

S 864-041
(3) Remove the DO-NOT-OPERATE tags from the thrust levers.

S 864-042
(4) Set the A/T switch on the AFCS mode control panel to the A/T ARM position.

S 714-095

WARNING: THIS TEST INCLUDES AUTOMATIC MOVEMENT OF THE THRUST LEVERS AND THEREFORE MAY NOT BE OPERATED WITH ANY ENGINE OPERATING.

WARNING: REFER TO AMM 27-61-00/201 FOR APPROPRIATE SPOILER/SPEEDBRAKE DEACTIVATION PROCEDURE. INADVERTENT SPOILER MOVEMENT RESULTING FROM THIS TEST COULD CAUSE SERIOUS INJURY TO PERSONNEL.

(5) Perform MCDP Ground Test 10-SERVO A/T (AMM 22-00-02/201).

S 864-046
(6) Set the A/T switch on the AFCS mode control panel to the OFF position.

S 864-043
(7) Do the thrust lever load test (AMM 76-11-00/501).

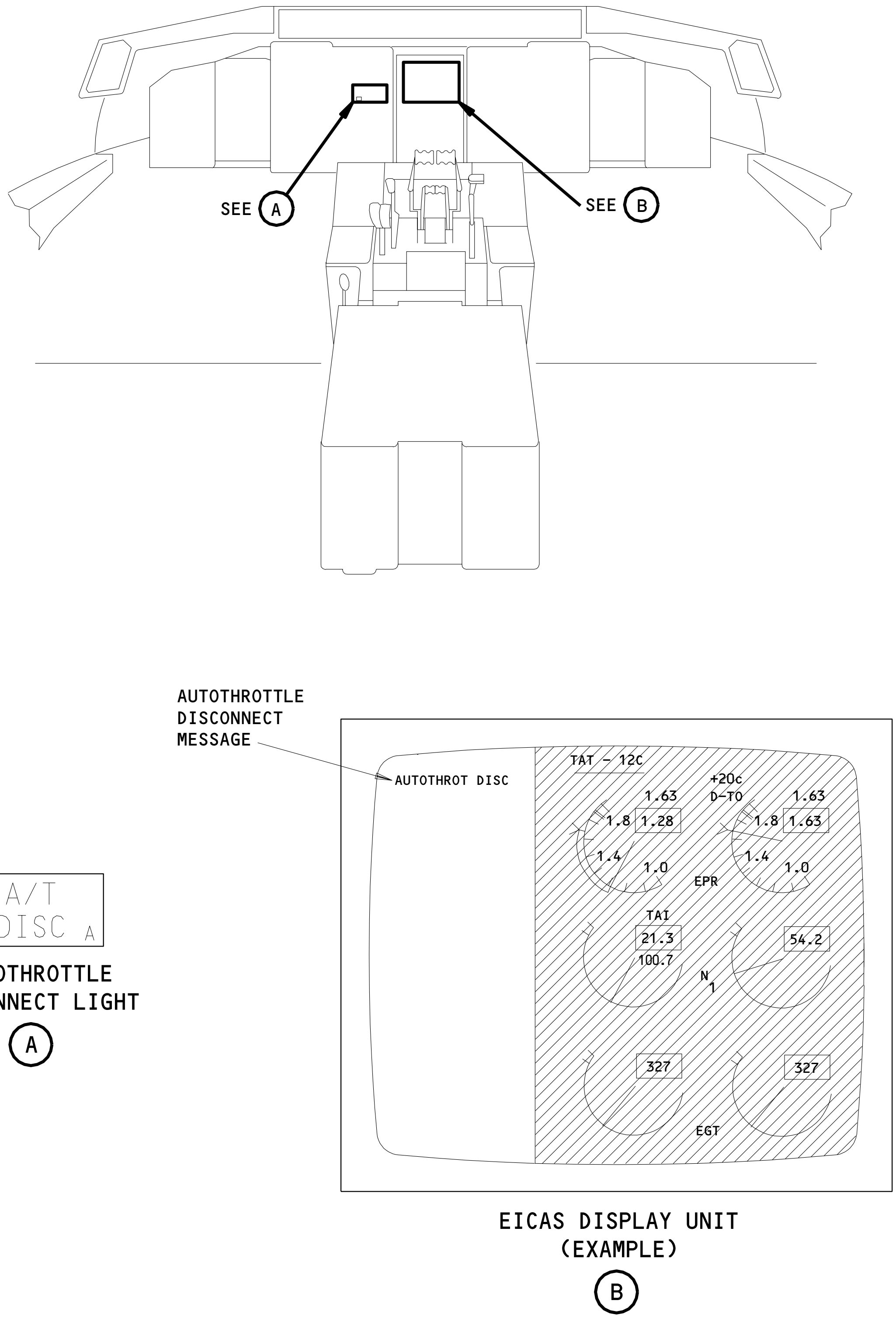
H. Put the Airplane Back to Its Usual Condition

S 414-044
(1) Close the 113AL access panel (AMM 06-41-00/201).

S 864-045
(2) Remove electrical power if it is not necessary (AMM 24-22-00/201).

THRUST MANAGEMENT WARNING AND ANNUNCIATION - DESCRIPTION AND OPERATION

1. General (Fig. 1)
   A. The Thrust Management System (TMS) is monitored for autothrottle disengagement. Disengagement results in caution annunciations that include an amber A/T DISC light, an AUTOTHROT DISC message on the Engine Indication and Crew Alerting System (EICAS) upper display unit (Ref 31-41-00), an aural warning, and flashing master CAUTION lights.
2. Component Details
   A. Autothrottle Disconnect Light
      (1) The A/T DISC light is on the P2 panel. It is an amber, dual-lamp indicator. The lamps use 28 vdc from the master dim and test system. They are turned on by application of a ground from the TMC or the master dim and test system (Ref AMM 33-16-00/001, Master Dim and Test). The A/T DISC light will also be illuminated whenever the TMC is in ground test mode.
3. Operation
   A. Functional Description
      (1) Thrust Management System - Warning and Annunciation
         (a) When the autothrottle disengages, the A/T outputs a pair of grounded discretes signals. One pair goes to the A/T DISENGAGE light and the other pair goes to left and right EICAS computers. These grounded discretes are controlled by both hardware and software flip-flop outputs. Each discrete pair has one discrete powered from the left 28V DC bus and the other from the 28V DC standby bus.
         (b) The hardware flip-flop is set by one of the following conditions: when commanded by software, the watchdog monitor senses a fault, a power failure occurs, or an A/T disengage switch is pressed.
             The software flip-flop is set by either software sensed failure (no A/T ENABLE signal from TMS engage-disengage logic) or when the A/T disengage switch is pressed.
             The flip-flops are reset by one of the following actions: a second press of a disengage switch, the A/T ARM switch positioned to OFF, or re-engagement of the autothrottle.
         (c) The following flight deck effects will occur after the A/T disconnect discretes are grounded:
             The A/T DISCONNECT light will turn on and then 0.5 seconds after the EICAS computer detects a discrete ground an AUTOTHROT DISC message will show on the EICAS display unit, the master CAUTION lights turn on, and an aural warning sounds.
             Pressing a master CAUTION light resets both master CAUTION lights and the aural warning. However, the amber A/T DISC light and EICAS message remain until both flip-flops are reset.



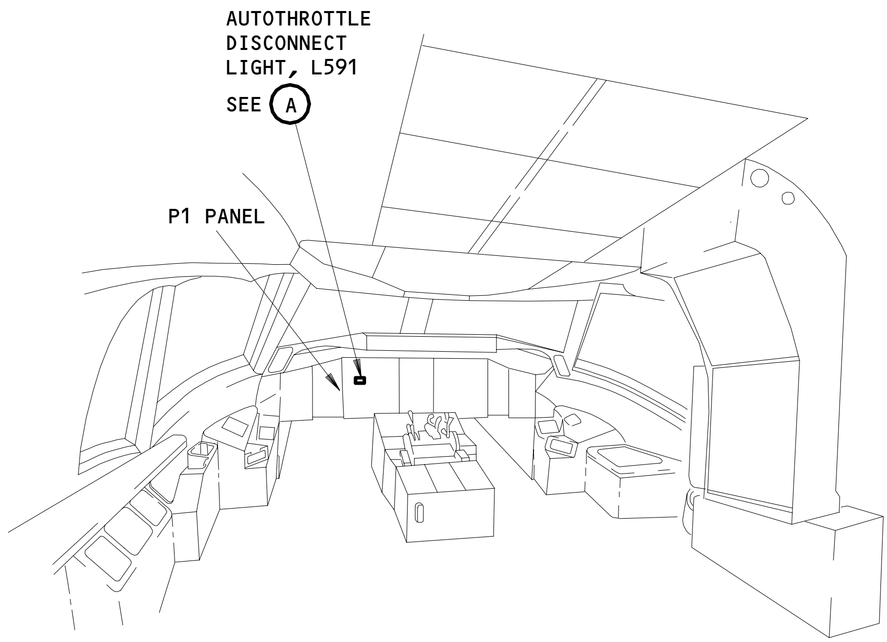
Flight Deck Indications
Figure 1

THRUST MANAGEMENT WARNING AND ANNUNCIATION

COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
LIGHT - AUTOTHROTTLE DISCONNECT, L591	--	1	FLT COMPT, P1	*

* SEE THE WDM EQUIPMENT LIST

Thrust Management Warning and Annunciation - Component Index
Figure 101



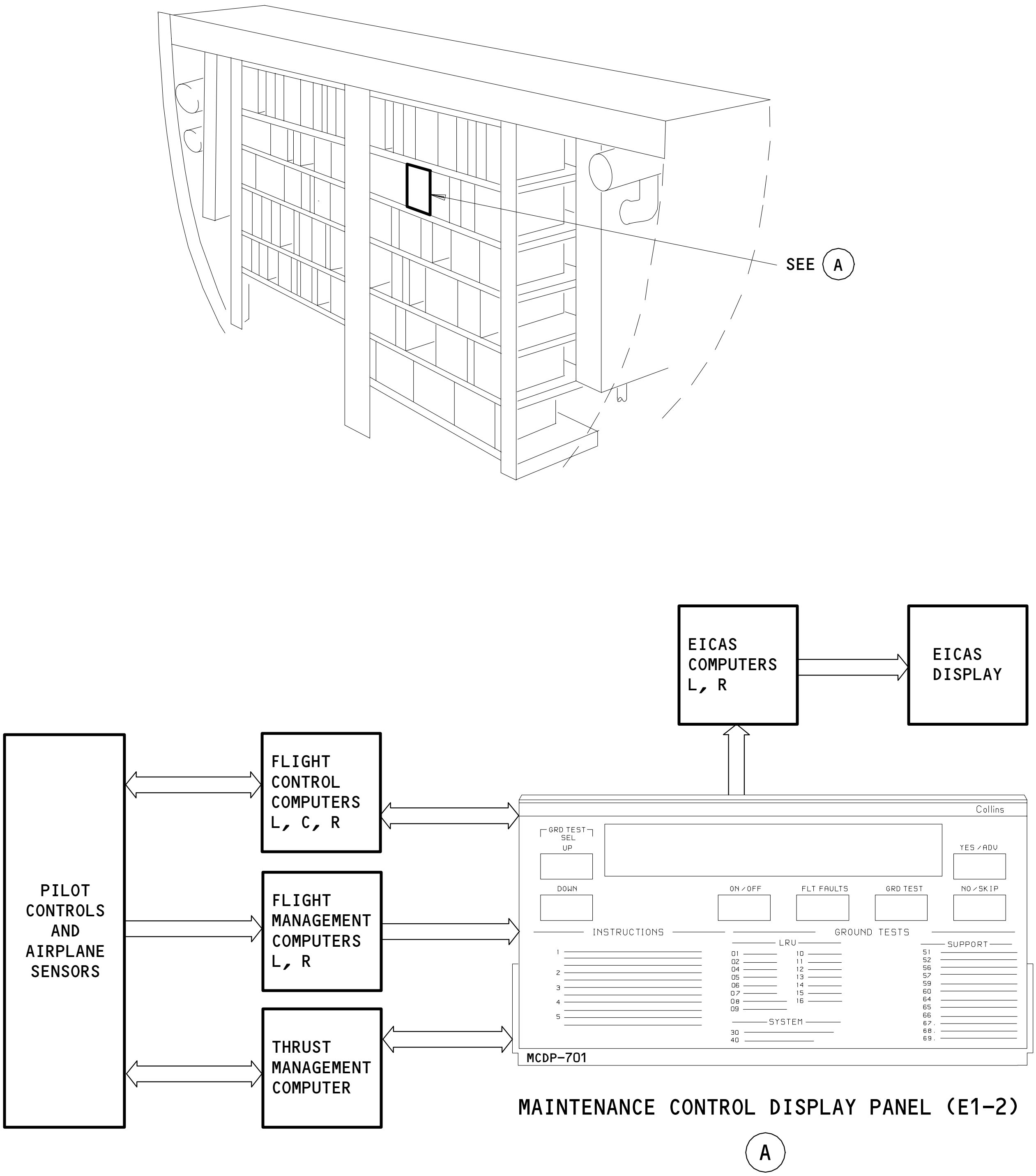
FLIGHT COMPARTMENT

AUTOTHROTTLE DISCONNECT LIGHT, L591
A

Thrust Management Warning and Annunciation - Component Location
Figure 102

MAINTENANCE MONITOR - DESCRIPTION AND OPERATION

1. General (Fig. 1)
    A. The maintenance monitor system consists of a Maintenance Control Display Panel (MCDP) that monitors the status of the Flight Control Computers (FCC), Flight Management Computers (FMC), Thrust Management Computer (TMC), and their related sensors. The MCDP interrogates the computers after each landing, and stores up to five flight faults per computer in non-volatile memory for later interrogation by maintenance personnel.
    B. A central processing unit in the MCDP controls and processes ground test commands, ground test functions, and on-ground and in-flight faults. The MCDP displays flight and ground faults, test number, name or status, and operator instructions. The MCDP controls and display panel are located on the front of the unit. The controls enable maintenance personnel to display faults and ground tests that include operator instructions. The MCDP is located in the main Electrical/Electronic (E/E) equipment center.
    C. There is wiring and connector provisions for a remote control panel. The remote control panel is plugged into the connector on the P6 panel and is used for displaying flight faults and for running ground tests from the flight deck. The MCDP remote display is through the EICAS system maintenance panel in the P61 panel. The MCDP display can be displayed on the EICAS with or without the hand-held carry-on remote control panel by pressing the CONF/MCDP switch on the EICAS maintenance panel.



Maintenance Monitor System
Figure 1

MCDP ABBREVIATIONS AND ACRONYMS	
MCDP ABBREVIATION	NOMENCLATURE
A-ATCK	ANGLE OF ATTACK
A/C	AIRCRAFT
A/G	AIR/GROUND
A/L	AUTOLAND
A/P	AUTO/PILOT
A/T	AUTO/THROTTLE
ACC	ACCELERATION
ACT	ACTIVITY
ADC	AIR DATA COMPUTER
ADP	AIR DRIVEN PUMP
AI, A-ICE	ANTI-ICE
AIL	AILERON
ALT	ALTITUDE
ALTNT	ALTERNATE
ANG	ANGLE
AOA	ANGLE OF ATTACK
ARINC IO	ARINC I/O ERROR
ARNC STP	ARINC I/O UART DATA STRIP ERROR
ASA	AUTOLAND STATUS ANNUNCIATOR
AUG	AUGMENT
AUTO	AUTOMATIC

MCDP ABBREVIATIONS AND ACRONYMS	
MCDP ABBREVIATION	NOMENCLATURE
BALTI	INERTIAL BAROMETRIC ALTITUDE
BARO	BAROMETRIC
BAT	BATTERY
BK	BRAKE
BUS	ELECTRICAL POWER/429 DIGITAL DATA
BUTTON	BUTTON/SWITCH
C	CENTER
CAS	COMPUTED AIRSPEED
C/B	CIRCUIT BREAKER
CCW	COUNTERCLOCKWISE
CDU	CONTROL DISPLAY UNIT FOR FMC
CH	CHANNEL
CLR	CLEAR
CMD	COMMAND
CMM	COMMON MODE MONITOR
COL	COLUMN
COMB	COMBINED

MCDP ABBREVIATIONS AND ACRONYMS	
MCDP ABBREVIATION	NOMENCLATURE
COMP	COMPARE
CONFG	CONFIGURATION
CP	CONTROL PANEL
CTN	CAUTION
CUST	CUSTOMER
CW	CLOCKWISE
DEC	DECREASE/DECREMENT
DEG	DEGREE(S)
DET	DETECTOR
DETNT	DETENT
DISC	DISCONNECT
DISENG	DISENGAGE
DME	DISTANCE MEASURING EQUIPMENT
DN	DOWN
DSCRT IO	DISCRETE I/O ERROR
DSPLY	DISPLAY
ECS	ENVIRONMENTAL CONTROL SYSTEM
EEC	ELECTRONIC ENGINE CONTROL
EFIS	ELECTRONIC FLIGHT INSTRUMENT SYSTEM
EICAS	ENGINE INDICATION AND CREW ALERTING SYSTEM

MCDP ABBREVIATIONS AND ACRONYMS	
MCDP ABBREVIATION	NOMENCLATURE
ELEV, ELV	ELEVATOR
ENG	ENGAGE/ENGINE
EPR	ENGINE PRESSURE RATIO
ERR	ERROR
F/D	FLIGHT DIRECTOR
F/O	FIRST OFFICER
F/PLN	FLIGHT PLAN
FCC	FLIGHT CONTROL COMPUTER
FL	FLIGHT LEVEL
FLT	FLIGHT
FLTP	FLIGHT PATH
FMC	FLIGHT MANAGEMENT COMPUTER
FWD	FORWARD
G/S	GLIDESLOPE
GA	GO-AROUND
GRD	GROUND
GRD SPD	GROUND SPEED

MCDP ABBREVIATIONS AND ACRONYMS	
MCDP ABBREVIATION	NOMENCLATURE
HDOTI	INERTIAL ALTITUDE RATE
H/L	HIGH/LOW
HDG	HEADING
HYD	HYDRAULIC
IAS	INDICATED AIR SPEED
IDENT	IDENTIFICATION
ILS	INSTRUMENT LANDING SYSTEM
IN	IN/INPUT
INC	INCREASE/INCREMENT
INFC	INTERFACE
INHIB, INH	INHIBIT
INIT	INITIATE
INST	INSTRUMENT
INTER SYST	INTER-SYSTEM
INTLK	INTERLOCK
IRU	INERTIAL REFERENCE UNIT
ISLN, ISOL	ISOLATION
L	LEFT
L/S	LOW SPEED
LCR	LEFT-CENTER-RIGHT
LDR	LOADER
LIM	LIMIT

MCDP ABBREVIATIONS AND ACRONYMS	
MCDP ABBREVIATION	NOMENCLATURE
LNAV	LATERAL NAVIGATION
LOC	LOCALIZER
LT	LIGHT
LVDT	LINEAR VARIABLE DIFFERENTIAL TRANSDUCER
LVR	LEVER
MAG	MAGNETIC
MAG HDG	MAGNETIC HEADING
MAN	MANUAL
MCDP	MAINTENANCE CONTROL DISPLAY PANEL
MCP	MODE CONTROL PANEL
MON	MONITOR
MOT	MOTION
MSTR	MASTER
MY	DETECTED BY DIAGNOSTIC IDENTIFIED SOURCE COMPUTER, NOT RECEIVED ON CROSS-CHANNEL DATA BUS
NAV	NAVIGATION
NCD	NO COMPUTED DATA (429 SSM)
NEG	NEGATIVE
NO	NUMBER/NO
NRM	NORMAL
NVMEM RD	NON-VOLATILE MEMORY READ ERROR

MCDP ABBREVIATIONS AND ACRONYMS	
MCDP ABBREVIATION	NOMENCLATURE
NVMEM WR	NON-VOLATILE MEMORY WRITE ERROR
OFF-SIDE CDU, OFSID	LT CDU AND RT FMC OR RT CDU AND LT FMC
ON-SIDE CDU, ONSID	LT CDU AND LT FMC OR RT CDU AND RT FMC
OP	OPERATION
OPT	OPTION
OUT	OUTPUT
P/N	PART NUMBER
P/S	PITOT STATIC
PACK	PACK (AIR CONDITIONING)
PCT	PERCENTAGE
PLA	POWER LEVER ANGLE
PNL	PANEL
POS	POSITION
PRE-ENG	PRE-ENGAGE
PREV	PREVIOUS
PRFM	PERFORMANCE
PROBE	PITOT-PROBE
PROG FLOW	PROGRAM FLOW ERROR
PROG MEM	ROM MEMORY ERROR

MCDP ABBREVIATIONS AND ACRONYMS	
MCDP ABBREVIATION	NOMENCLATURE
PS	PITOT STATIC
PWR	POWER
QC	QUALITY CHECK
QTY, QUAN	QUANTITY
R	RIGHT
RA	RADIO ALTIMETER
RCVR	RECEIVER
PS	PITOT STATIC
PVD	PARA-VISUAL DISPLAY
PVDC	PARA-VISUAL DISPLAY COMPUTER
PWR	POWER
QC	QUALITY CHECK
QTY, QUAN	QUANTITY
R	RIGHT
RA	RADIO ALTIMETER
RCVR	RECEIVER
REF	REFERENCE
RESP	RESPONSE
RESTART	POWER INTERRUPT RESTART ERROR
REV	REVERSE
RLSE	RELEASE
RLY	RELAY

MCDP ABBREVIATIONS AND ACRONYMS	
MCDP ABBREVIATION	NOMENCLATURE
RLY/SW	RELAY/SWITCH
RMT	REMOTE
RMT OUT	HIGH SPEED ARINC OUTPUT ERROR
RST	RESET
RSVD	RESERVED
RT	RECEIVER/TRANSMITTER
RUD	RUDDER
R/W MEM	RAM MEMORY ERROR
RWY	RUNWAY
SAM	STABILIZER/AILERON LOCKOUT MODULE
SEL	SELECT
S/N	SERIAL NUMBER
SOV	SHUTOFF VALVE
SPD BK	SPEED BRAKE

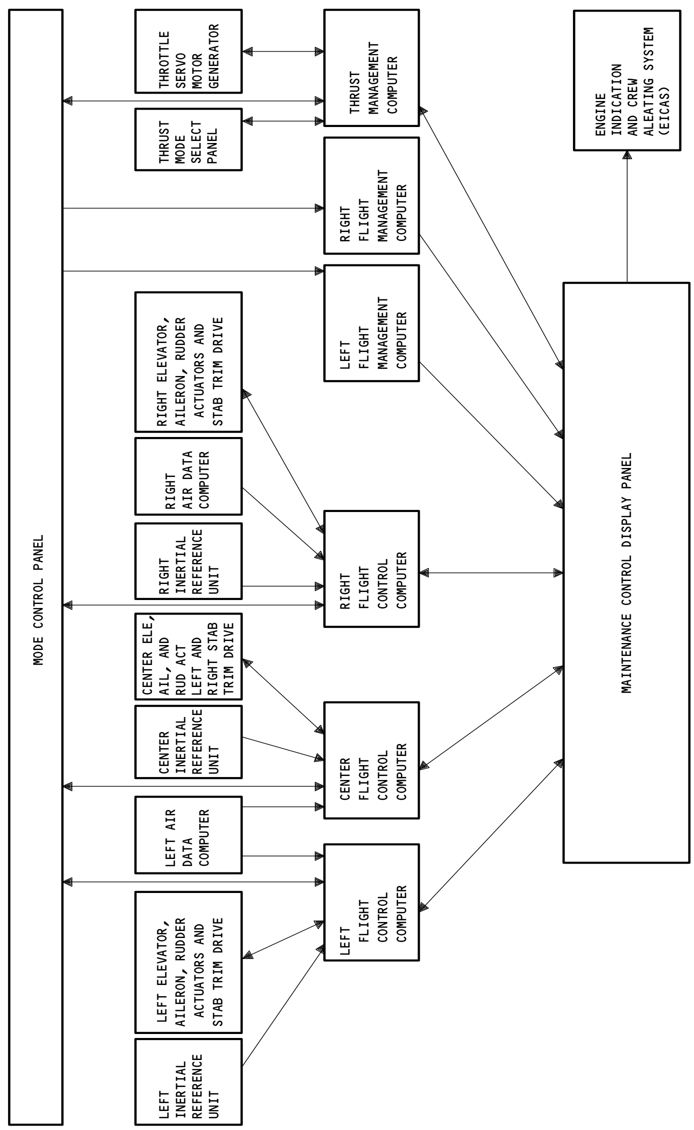
MCDP ABBREVIATIONS AND ACRONYMS	
MCDP ABBREVIATION	NOMENCLATURE
SRVO	SERVO
SSFD	SIGNAL SELECTION FAULT DETECTION
SSM	SIGN STATUS MATRIX
STAB	STABILIZER
STAT	STATIC
SURF	SURFACE
SW	SWITCH
S/W	SOFTWARE
SW-HW	SOFTWARE-HARDWARE
SYST	SYSTEM
TACH	TACHOMETER
TAS	TRUE AIRSPEED
TAT	TOTAL AIR TEMPERATURE
TEMP	TEMPERATURE
TERMINAT	TERMINATE
THROT	THROTTLE
THRST	THRUST

MCDP ABBREVIATIONS AND ACRONYMS	
MCDP ABBREVIATION	NOMENCLATURE
TIE	BUS TIE
TLA	THROTTLE LEVER ANGLE
TMC	THRUST MANAGEMENT COMPUTER
TMSP	THRUST MODE SELECT PANEL
TO	TO/TAKEOFF
TRK	TRACK
TST	TEST
TUNE	TUNER
VERT	VERTICAL
VERT ACC	VERTICAL ACCELERATION
VFY	VERIFY
VLD	VALID
VLV	VALVE
VMO	MAXIMUM OPERATING SPEED
VNAV	VERTICAL NAVIGATION

MCDP ABBREVIATIONS AND ACRONYMS	
MCDP ABBREVIATION	NOMENCLATURE
W/A	WRAP AROUND
WD	WORD
WG	WING
WHL	WHEEL
WRN	WARNING
X-CH, X-CHAN	CROSS CHANNEL
X-FEED	CROSS-FEED
XDCR	TRANSDUCER
?	ANSWER YES/ADV IF TEST DISPLAYED IS TO RUN OR NO/SKIP TO ADVANCE TO NEXT TEST
/	IN FAULT MESSAGES, INDICATES POSSIBLE WIRE PROBLEM BETWEEN LISTED LRUs
#	USED IN DOCUMENT TO INDICATED L, R, C AS APPROPRIATE
-	IN FAULT MESSAGES, INDICATES "OR"

D. Configuration
  (1) SAS 050-051, 150-155, 162-165;
      MTH 275-276;
      MCDP -304 is installed.
  (2) SAS 052-149, 156-161, 166-999;
      MTH 277-999;
      MCDP -305 is installed.

E. Maintenance Control Display Panel Interfacing Systems (Fig. 2)
  (1) The primary MCDP interfaces are with each FCC, FMC, and TMC. Secondary interfaces with the MCDP consist of FCC, FMC, and TMC individual system sensors. Both primary and secondary interfaces are monitored by the MCDP during ground test functions.
  (2) Primary Interfaces
    (a) The left, center, and right flight control computers use ARINC 429 data buses for transmitting interface fault data to the MCDP and receiving ground test data from the MCDP. Analog discrete lines are used for ground test control.



Maintenance Control Display Panel Interfacing Systems
Figure 2

145052

(b) The left and right flight management computers use ARINC 429 data buses to transmit interface fault data to the MCDP. The MCDP can only interrogate the FMC for fault data. The MCDP ground tests do not affect the FMC systems.

(c) The thrust management computer uses an ARINC 429 data bus for transmitting interface fault data to the MCDP and receiving ground test data from the MCDP. An analog discrete line is used for ground test control of the TMC by the MCDP.

(3) Secondary Interfaces

(a) The mode control panel supplies control signals and receives status data from each FCC, FMC, and TMC on ARINC 429 data buses.

(b) Each FCC supplies analog control signals to its control servos and receives analog servo position signals. Each FCC receives sensor data on ARINC 429 data buses from the associated Inertial Reference Unit (IRU) and air data computer.

(c) The TMC receives control signals on an ARINC 429 data bus from the thrust mode select panel. The TMC also transmits status data to the thrust mode select panel on an ARINC 429 data bus. The TMC supplies analog control signals to and receives analog position signals from the autothrottle servomotor generator.

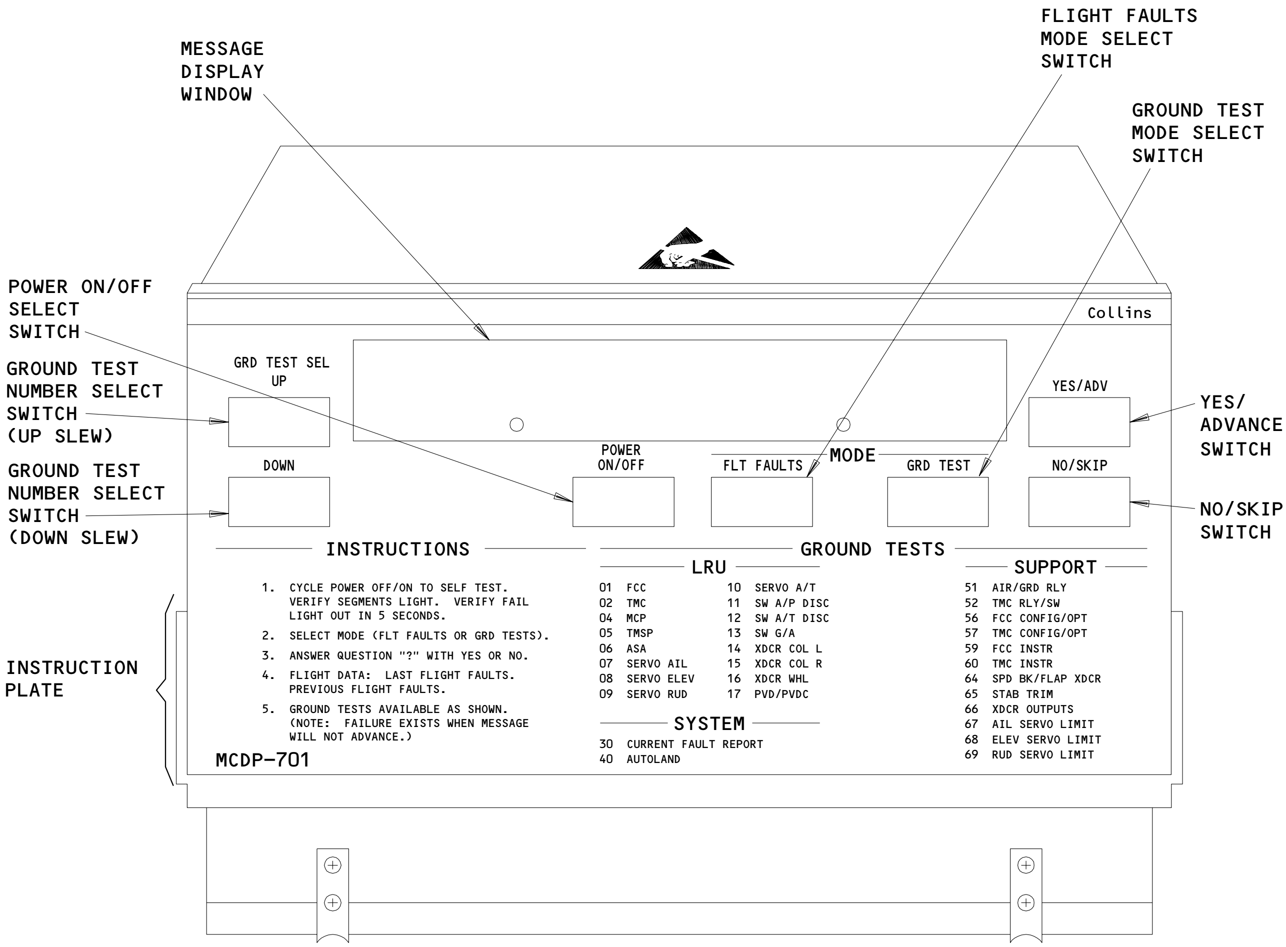
2. Component Details

A. Maintenance Control Display Panel (Fig. 3)

(1) The MCDP automatically obtains and stores fault data after each landing from each FCC, FMC, and TMC. Maintenance personnel can then interrogate the MCDP to obtain last and previous flight fault data. They can also select and conduct MCDP controlled ground tests.

NOTE: Do not transfer the MCDP directly from another airplane. Historical flight data is valid only for the original airplane installation. The flight data is recorded in the MCDP non-volatile memory (EAROM) and can only be erased using shop procedures.

(2) The MCDP size is 6 MCU per ARINC 600. Weight is approximately 15 pounds. It uses bottom to top forced air cooling when installed on shelf E1-2 in the main E/E equipment center. The MCDP uses 115 vac, 400 Hz, single phase power from a circuit breaker on panel P11.



(example MCDP)

Maintenance Control Display Panel
Figure 3

(3) The display window contains 32 characters arranged in 2 lines of 16 characters each. Each character, including decimal point, consists of 17 segmented gas discharge lamps. The display uses an English language format.

(4) The two LEDs on the bottom edge of the display window indicate the Keep-Alive-Power (KAP) circuit is powered. Lights are on when there is power to the MCDP, whether the MCDP is on or off. These lights are covered, so are not very visible.

(5) Control Switches

(a) There are seven front panel control switches. Three of the switches are lighted switches with front panel replaceable lamps. All switches are momentary switches.

(b) The ON/OFF switch is a push ON/OFF light/switch. The switch initiates a power-up routine in the Central Processing Unit (CPU). The power-up routine initially turns on the FAIL message in the ON/OFF switch. If the MCDP passes the self-test routine, the FAIL light turns off after 5 seconds.

(c) The FLT FAULTS (flight faults) switch is a push-on light/switch. The lamp turns on when the function is engaged. The FLT FAULTS switch commands the CPU to enter the LAST FLIGHT FAULT ROUTINE. The FLT FAULTS MODE switch/light is also used to display the flight fault diagnostic code of the non-interface flight fault displayed.

(d) The GRD TEST (ground test) switch is a push-on light/switch. The lamp turns on when the function is engaged. The GRD TEST switch commands the CPU to enter the ground test mode and initialize to the lowest available test number. The GRD TEST MODE switch/light is also used with ground test 30, CURRENT FAULT REPORT to display the ground fault diagnostic code of the non-interface ground fault displayed.

(e) The UP (slew up) switch increases the test number in a sequential loop each time the switch is pressed. Holding the switch on slews the test numbers at the rate of four test numbers per second. Unused test numbers are automatically skipped.

(f) The DOWN (slew down) switch reduces the test number in a sequential loop and slews at the same rate as the UP switch.

(g) The YES/ADV (yes/advance) switch is used to respond to messages ending either with a question mark (?) or word -ADV. Pressing the YES/ADV button in response to the question engages the displayed flight fault mode or ground test. The advance is used to advance to the next flight fault or ground test step. The operator must press YES/ADV to advance to the next test step if -ADV is displayed at end of display message whether test step passes or not. If -ADV is not displayed at end of display message and the test does not automatically advance to the next test step, then that test step has failed and the operator must push YES/ADV to continue the test.

(h) The NO/SKIP switch is to respond to messages ending with a question mark (?) when the operator does not want to display the particular flight faults or run the particular ground test. If the NO/SKIP switch is pressed in response to a question, then the MCDP will skip to the next test question. If the question is part of a sub-routine of a test then the MCDP will skip to the next sub routine within that test if there are any. If NO/SKIP is pressed in response to a ground test message that does not end with a question mark, the MCDP will exit that ground test and display the next test listed on the front of the MCDP. The exit function of the NO/SKIP switch will not work if the message displayed is a fault message.

NOTE: The YES/ADV and NO/SKIP switch will not respond when the MCDP displays in test 40, SET R BUS TO ISOL FOR 15 SEC.

(6) The operating instruction plate is on the lower left side of the front panel. It contains the instructions for applying power and initializing the MCDP. It tells how to select each mode and respond to interrogative messages.

(7) The GROUND TEST Listing is located on the lower right and center of the front panel. It contains test numbers and corresponding names that can be selected. The ground tests are listed as LRU, SUPPORT, and SYSTEM groups.

(8) Internal circuits in the MCDP consist of six functional elements on ten printed circuit boards.

(a) The power supply module consists of a power supply control card and a power supply card. The power supply converts the ac input to six dc voltages. It provides a separate unswitched 5 vdc (keep-alive-power) for the on/off logic circuitry. It has enough storage capacity to maintain its power output and normal activity during brief power interruptions of less than 200 milliseconds.

(b) The ARINC I/O (input/output) card receives data on six low-speed ARINC 429 buses (3 FCC, 2 FMC, 1 TMC). The card provides the buffering, level changing, and temporary storage of the input data. It transmits data on one high-speed and three low-speed ARINC buses. The high-speed bus transmits data to the optional remote display panel. The low-speed buses transmit data to the three flight control computers and the thrust management computer. The ARINC I/O card provides the timing and drive capability for the ARINC input and output buses.

(c) The I/O card provides the interface to the microprocessor central bus. The card accepts 16 strapping inputs including four spares, two air/ground discretes, and seven panel switch inputs. It also has eight functional test outputs and three annunciator outputs.

(d) Program memory is supplied by two cards that provide up to 56K bytes of Ultra Violet Erasable PROM (UV EPROM). The EPROM contains the programming that is controlled by the executive routine in the central processing unit (CPU). The programming manages the power-up initialization, power interrupt management, mode switching, periodic functions management, and power-down routines. The mode switching routine controls the auto-on, power-up, self-test, flight faults, and ground test routines. During the power-up self-test, the MCDP does not accept flight fault data. It only accepts or displays flight faults after the self-test is passed.

(e) The processor card contains the CPU, scratch pad memory, and Electrically Alterable ROM (EAROM). The CPU's internal executive routine controls the programming in the program memory cards.

(f) The display module consists of a display control card and a 32 character alpha-numeric gas discharge display. The display control stores the display data received from the CPU in its internal 96 x 8 bit RAM. The display control also contains the seven front panel switches and three annunciators that allow maintenance personnel to interface with the MCDP.

(g) There are two interconnect boards, rear and sideboard. The rear interconnect board routes incoming and outgoing signals between the ARINC 600 connector and the internal elements of the MCDP. The sideboard interconnect board contains the central bus. The central bus transfers the data between the internal elements of the MCDP and the front panel control board or the rear interconnect.

(9) The option pins allow a standard MCDP design to be customized by selecting applicable programming. The selecting is done by grounding (logic one) or leaving open (logic zero) the appropriate discrete option pins. These pins are located in the main connector on the back of the MCDP and are controlled with airplane wiring. Three pins are used to select the airplane configuration, one pin to select single or dual FMC option, and another pin for dual or triple FCC option. Five pins are reserved for customer options. A parity pin is used make the total option pins set to logic one an odd number.

(a) Airplane Configuration Pins 1, 2, and 3 are all set to zero for 767 airplanes.

(b) The SINGLE FMC option pin set to logic zero indicates the dual FMC option.

(c) The DUAL FCC option pin set to logic zero indicates the three FCC option.

(d) Customer Option Pins:

1) ADC SWITCHING (option 1) pin set to logic zero indicates to the MCDP that ADC switching on the pilots instrument select panel does not affect the ADC inputs to the FCCs.

2) FMC SWITCHING (option 2) pin set to logic zero indicates FMC switching on the pilots instrument select panel does not affect the FMC inputs to the FCCs.
3) TMS A/P DISCONNECT (option 3) pin set to logic zero indicates to the MCDP that the airplanes autopilot disconnect switches and autothrottle disconnect switches operate independent of each other.
4) FMC FAULT REPORTING (option 4) pin set to logic one allows MCDP to display FMC flight faults and current ground faults.
5) MCDU/FMC SWITCHING (option 5) pin set to logic one indicates FMC switching on the pilots instrument select panel is a three position rotary switch and does affect FMC inputs to the FCC's.
6) PVD SYSTEM (option 6) pin set to logic zero indicates the PVD system is not activated.

(10) The MCDP is an electrostatic sensitive unit that requires special handling using authorized procedures (Ref Chapter 20).

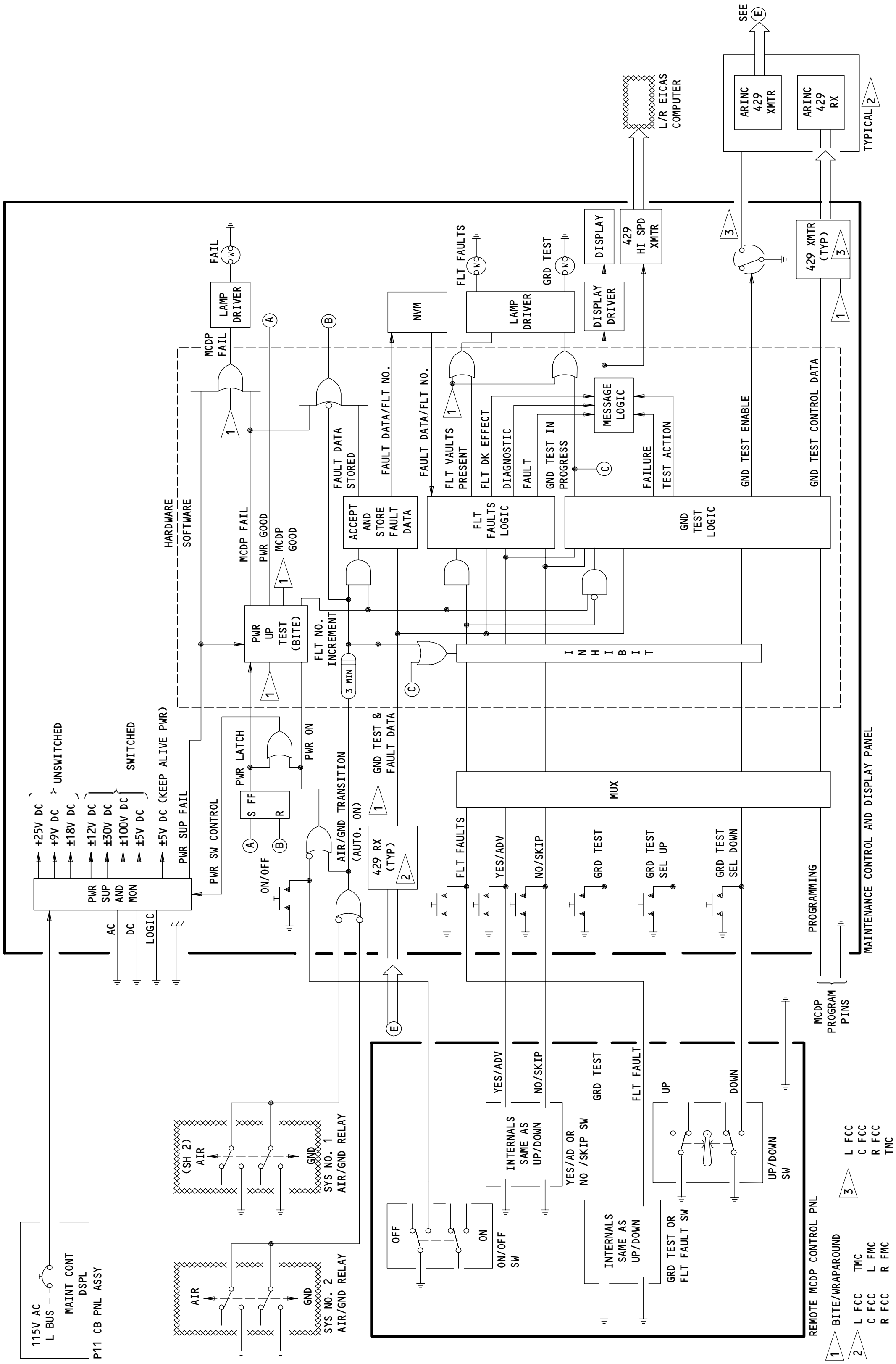
3. Operation

A. Functional Description

(1) Maintenance Control Display Panel (Fig. 4)

(a) The CPU in the MCDP initiates an automatic power-up and self-test routine after each landing when an on-ground condition is sensed by either air/ground relay. The 5 vdc keep-alive-power on the CPU enables the power-up routine. A three minute timer is started at the same time the power-up routine is initiated. A successful self-test starts the flight faults routine. The flight faults routine requests fault data from the six computers (3 FCC, 2 FMC, 1 TMC). A failed self-test causes the CPU to power-down the MCDP. No flight fault data is transferred. The CPU normally powers-down the MCDP after all flight faults are received or after three minutes.

(b) The manual power-up and self-test routines are initiated by momentarily pressing the ON/OFF switch on the MCDP front panel. The FAIL light in the ON/OFF switch is turned on during the self-test routine. It turns off after five seconds if the self-test is successfully completed. During the self-test, a display test pattern is provided for three seconds so the operator can verify that all the display light segments are working. After the FAIL light is turned off, the MCDP automatically enters FLT FAULT Mode and displays LAST FLT FAULTS?.



Maintenance Control Display Panel Schematic
Figure 4

202412

(c) With LAST FLT FAULTS? displayed after a successful self-test, pressing the YES/ADV switch causes a fault from the last flight to be displayed. Each time the YES/ADV switch is pressed, a different fault is displayed until all last flight faults have been displayed. After all last flight faults have been displayed, the display reads ALL LAST FLT FAULTS READ. If there are no flight faults, then the display reads NO LAST FLT FAULTS after the YES/ADV switch is pressed the first time.

(d) With LAST FLT FAULTS? displayed after a successful self-test, pressing the NO/SKIP switch causes PREV FLT FAULT? to be displayed. Then pressing the YES/ADV switch causes the previous flight faults to be displayed. Each previous flight fault is prefaced by the associated flight number. The flight number represents the number of flights since the displayed fault occurred. Up to 99 flights can be counted, but only 10 fault flights are stored. Flight numbers are skipped if no fault is stored for that flight. The message ALL PREV FLT FAULTS DSPLY is displayed after all previous flight faults have been displayed. If there are no previous flight faults, the message NO PREV FLT FAULTS is displayed.

(e) If fault message is displayed, pressing FLT FAULTS MODE switch/light displays its fault message diagnostic code. Alternate pressing YES/ADV switch and FLT FAULTS switch/light to displays each fault message and diagnostic code. Repeated pressing of the FLT FAULTS switch/light also causes the flight faults and diagnostic codes to be alternately displayed. If interface faults are displayed, the YES/ADV switch must be pressed to display the next fault. Interface faults do not have diagnostic codes.

(f) Diagnostic code can be decoded as follows:

```
DATA SOURCE
-----------
FCC=1     |
FMC=2     |
TMC=3     |       INTERMITTENT BIT SET OR A SENSOR FAULT
                  --------------------
          |         |               *4 (SSFDS) OCCURRED DURING
          |         |                  AUTOLAND AFTER APP
          X X X X X X                  SELECTED
              -----
            |   |                         1=YES
            |   |                         0=NO
            |   |
            |  DECIMAL CODE ASSIGNMENT (FIM 22-00-02, Fig 104
               -----------------------           or Fig 104A)
            |                          AUTOFLIGHT BITE
            |                          PROCEDURES)
            |  CHANNEL
           ----------
                L=1
                R=2
                C=3

FLIGHT FAULT DIAGNOSTIC DISPLAY FORMAT
```

(g) After a successful self-test, pressing the GRD TEST switch causes self-tests to be run on all six computers (3 FCC, 2 FMC, 1 TMC) and interface check data to be collected from all valid computers. This data is stored in the MCDP for use in the ground tests. While the self-tests are being run, an IN PROGRESS message is displayed. After the self-tests are completed, the MCDP displays the first test, 01 FCC TEST?. If the test number displayed is the test desired, pressing the YES/ADV switch engages the test. If the test number displayed is not the desired test, pressing the ground test select UP switch or the NO/SKIP switch causes the test number to increase in a sequential loop. Pressing the ground test select DOWN switch causes the test number to decrease in a sequential loop. The MCDP front panel contains a list of available tests.

(h) Engaging the selected test causes the MCDP CPU to set the ground discrete line. This signals the selected computer(s) (FCCs and/or TMC) that a ground test is to be run. The selected computer checks that its servos are engaged, its in-flight flag indicates an on-ground condition, and it has no faults stored as in-flight fault data. The computers send the CPU their status and set the test required wrap-around bit if the IN-AIR bit, servos engaged, and GRD fault bits are set. The set wrap-around bit indicates to the CPU that the selected computer is ready for the ground test. The CPU then sends the selected computer the test routine number and the necessary parameters for the test. It completes the test routine and sends the results to the CPU. If there are no faults, the CPU sends the next test routine number and parameters. During the test routine, the MCDP displays XX IN PROGRESS when operator interaction is not required (XX denotes the test number being performed). If operator interaction is required, the MCDP displays the appropriate message.

(i) The test routine stops and a fault message is displayed when a fault condition occurs. A fault during an operator response is indicated by the MCDP not advancing to the next step. Pressing the YES/ADV switch causes the CPU to continue the test. An XX TEST COMPLETE message is displayed when the test routine is completed. Pressing the YES/ADV or NO/SKIP switch causes the CPU to exit the test routine. Pressing the FLT FAULT switch interrupts the test routine that is in progress, and puts the MCDP into the in-flight mode. However, all information related to the interrupted test is lost.

(j) Diagnostic codes for ground tests are available for Test 30, CURRENT FAULT REPORT. If GROUND TEST 30 CURRENT FAULT REPORT is run, pressing GRD TEST MODE switch/light displays indicated faults diagnostic code. Alternately pressing YES/ADV switch and GRD TEST MODE switch/light displays each current fault and diagnostic code. Repeated pressing of the GRD TEST MODE switch/light also causes the current faults and diagnostic codes to be alternately displayed. The YES/ADV switch must be pressed to display the next fault, if an interface fault is displayed. Interface faults do not have diagnostic codes.

(k) Diagnostic code can be decoded as follows:

```
DATA SOURCE
-----------
FCC=1     |
FMC=2     |
TMC=3     |
          X X X X X
              -----
            |   |
            |  DECIMAL CODE ASSIGNMENT (FIM 22-00-02, Fig 104
            |   -----------------------          or Fig 104A)
            |                          AUTOFLIGHT BITE
            |                          PROCEDURES)
            |  CHANNEL
           ----------
               L=1
               R=2
               C=3
```

GROUND FAULT DIAGNOSTIC DISPLAY FORMAT *

* NOTE: Ground Fault diagnostic display is the same as flight fault format except the intermittent bit is omitted.

(l) The MCDP remote control panel switches are in parallel with the switches on the MCDP. The MCDP display is duplicated on the EICAS display when the CONFIG/MCDP switch is selected on the EICAS Maintenance Panel on the P61 panel. Operation and display using the remote control panel and EICAS display is the same as the standard MCDP control and display.

(2) MCDP Power-Up and Self-Test (Fig. 4)

(a) The MCDP power supply requires a 115v ± 10 vac, 400 ± 30 Hz, single phase power. It receives power from the left ac bus through a circuit breaker on panel P11. The MCDP has individual ac, dc, logic, and chassis grounding. The MCDP power supply is isolated from its ac source by a transformer. The ac to dc power conversion is accomplished with full wave, center tapped, rectifier circuits with noise suppression. Filter circuits limit output voltage ripple to approximately 5 percent.

(b) The dc outputs consist of ± 12 vdc, -30 vdc, ± 100 vdc, 5 vdc, and separate 5 vdc Keep-Alive-Power (KAP). All dc supply levels, except the 5 vdc, are monitored so that an out-of-tolerance voltage causes the FAIL annunciator to turn on. All dc supplies are switched, except for the 5 vdc KAP that is on whenever there is power on the airplane. The KAP is center tapped directly off the transformer and is regulated with a zener diode. The KAP enables the ON/OFF control logic, which causes full-power voltage regulators operation. The second 5 vdc supply is a regulated switched supply. An unregulated 25 vdc supplies its regulator. A 25 vdc monitor detects impending power loss and generates an output interrupt signal to the CPU when unregulated voltage drops below 18 vdc. The switched 5 vdc voltage level is monitored so when the level falls below 4.75 vdc a HOLD signal is sent to the CPU. Operation of the CPU is then inhibited.

(c) Automatic Power-Up and Self-Test

1) Automatic power-up and self-test is initiated by an on-ground indication from either air/ground relay. This triggers two pulse circuits in the control logic. One pulse circuit turns on the power supply and instructs the CPU executive module to do the power-up and latch routine. The other pulse circuit starts the three minute timer and generates the AUTO ON signal to the self-test manager module.

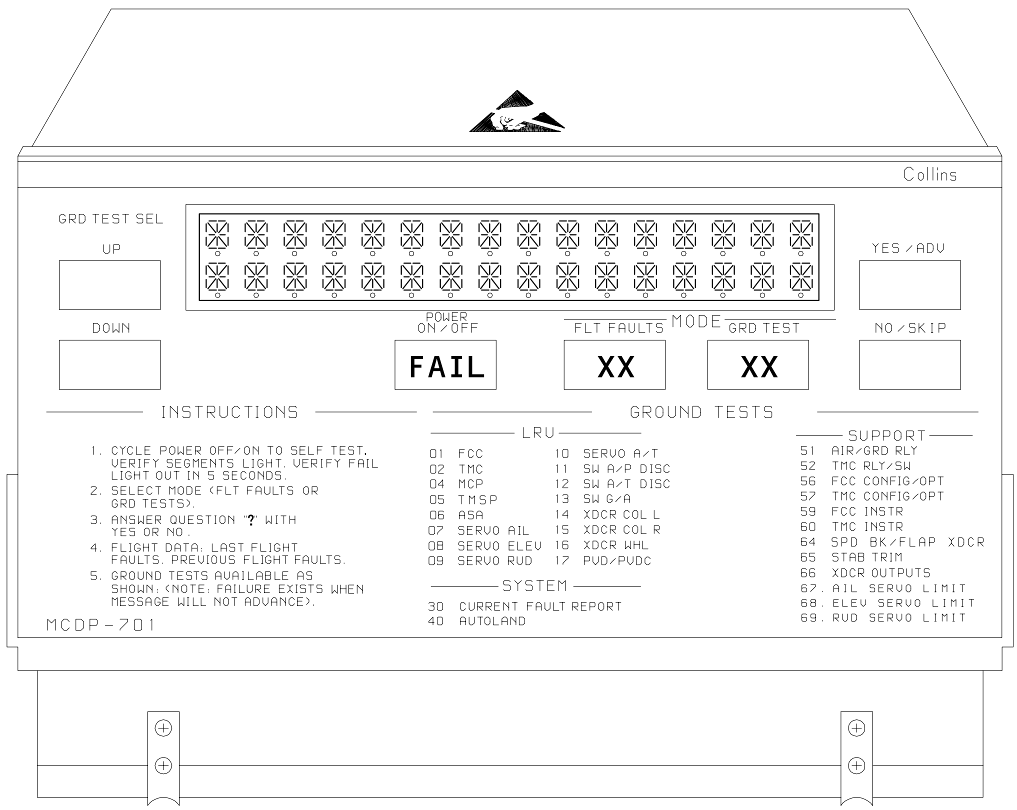
2) The executive module (through the power-up and latch routine) checks that the regulated 5 vdc supply has a minimum level of 4.75 vdc. If the voltage is within limits the power latched-on flip-flop is set, and provides a POWER LATCHED ON signal to the self-test module.

3) The CPU reads the option pins to determine airplane configuration (757 or 767), number of FMCs and FCCs, customer options (ADC or FMC switching, etc), and parity. The option pin data is stored for use in program routines.

4) The POWER LATCHED ON signal to the CPU enables the self-test routines. The AUTO ON signal selects which self-test routines are performed. When the AUTO ON SIGNAL is enabled, only the scratch pad memory check and program memory check are performed. The other five self-tests require operator interaction or cannot be accomplished when the airplane is in the air or rolling down the runway. A self-test failure causes the CPU to send a reset signal to the power latching flip-flop. The reset flip-flop signal removes the power regulators control signal. This switches the regulators off and removes operating power. A successful self-test allows the three minute timer to expire. This removes the AUTO ON signal and triggers a pulse circuit that sends a reset signal to the flip-flop.

5) During an auto power-up, the displays PROM sequencer is in a standby state. This standby mode of the display reduces power consumption approximately 15 percent.

(3) Manual Power-Up and Self-Test (Fig. 5)
   (a) The manual power-up and self-test is initiated by pressing the ON/OFF lightswitch. This sends a pulse signal to the power supply switching control and instructs the CPU to call up the power-up and latch routine.
   (b) The power-up and latch routine checks that the regulated 5 vdc power level has a minimum level of 4.75 vdc. If the level is within limits, the power-up and latch routine triggers a pulse circuit that sets the power regulators latching flip-flop, and provides a POWER LATCHED ON signal to the self-test manager module.
   (c) The option pins are read by the CPU to determine the airplane configuration, FMC, FCC, customer options, and parity. The option pin data is then stored for later use in the program routines.
   (d) The POWER LATCHED ON signal to the CPU enables the self-test routines. The absence of an AUTO ON signal tells the CPU that a manual power-up has been done. Therefore, all of the following seven self-test functions are to be performed: 1) random access memory, 2) read only memory, 3) low-speed ARINC wrap-around, 4) high-speed ARINC wrap-around, 5) discrete wrap-around, 6) display, 7) lights. The self-test of the display and lights consists of turning on the annunciator lights and all segments of the 32 character display for approximately three seconds for operator verification. If a fault is found during the self-tests, the CPU generates an MCDP failure signal that holds the FAIL annunciator on. The FAIL annunciator stays lighted until a manual power-off is done. A successful self-test causes the CPU to enter FLIGHT FAULTS MODE. This is indicated by the FLT FAULTS annunciator lighting and a LAST FLT FAULTS ? message being displayed.
   (e) A manual power-down signal is generated by pressing the ON/OFF lightswitch. This causes the CPU executive module to generate a TURN POWER OFF pulse. The pulse resets the power-on latch flip-flop to off. This enables the power-down routine which turns off the power supply switching control.



- INITIATE MANUAL POWER UP BY PRESSING ON/OFF BUTTON
- A SELF-TEST IS CONDUCTED AND THE ABOVE TEST PATTERN SHOWN
- AFTER 5 SECONDS, A SUCCESSFUL TEST IS INDICATED BY THE FAIL AND GRD TEST LIGHTS EXTINGUISHING
- THE FLIGHT FAULTS MODE IS ENTERED

MCDP Manual Power Up and Self-Test
Figure 5

(4) Remote MCDP Operation (Fig. 6)

(a) Remote MCDP operation uses a hand held MCDP remote control panel that is connected behind the P6-4 panel in the flight compartment. The remote panel switches are connected in parallel with the control switches on the MCDP.

(b) The remote MCDP operation displays MCDP data on the EICAS display in the P-2 panel. The EICAS system accepts MCDP data when the CONFIG/MCDP switch is selected on the EICAS Maintenance Panel located on the P61 panel. The EICAS maintenance panel is disabled in flight through an air/ground relay.

(c) The EICAS displays MCDP data the same as the MCDPs display. In addition to the normal display, the EICAS also displays the following:

1) MCDP OFF if CONFIG/MCDP switch is selected and the MCDP is off.

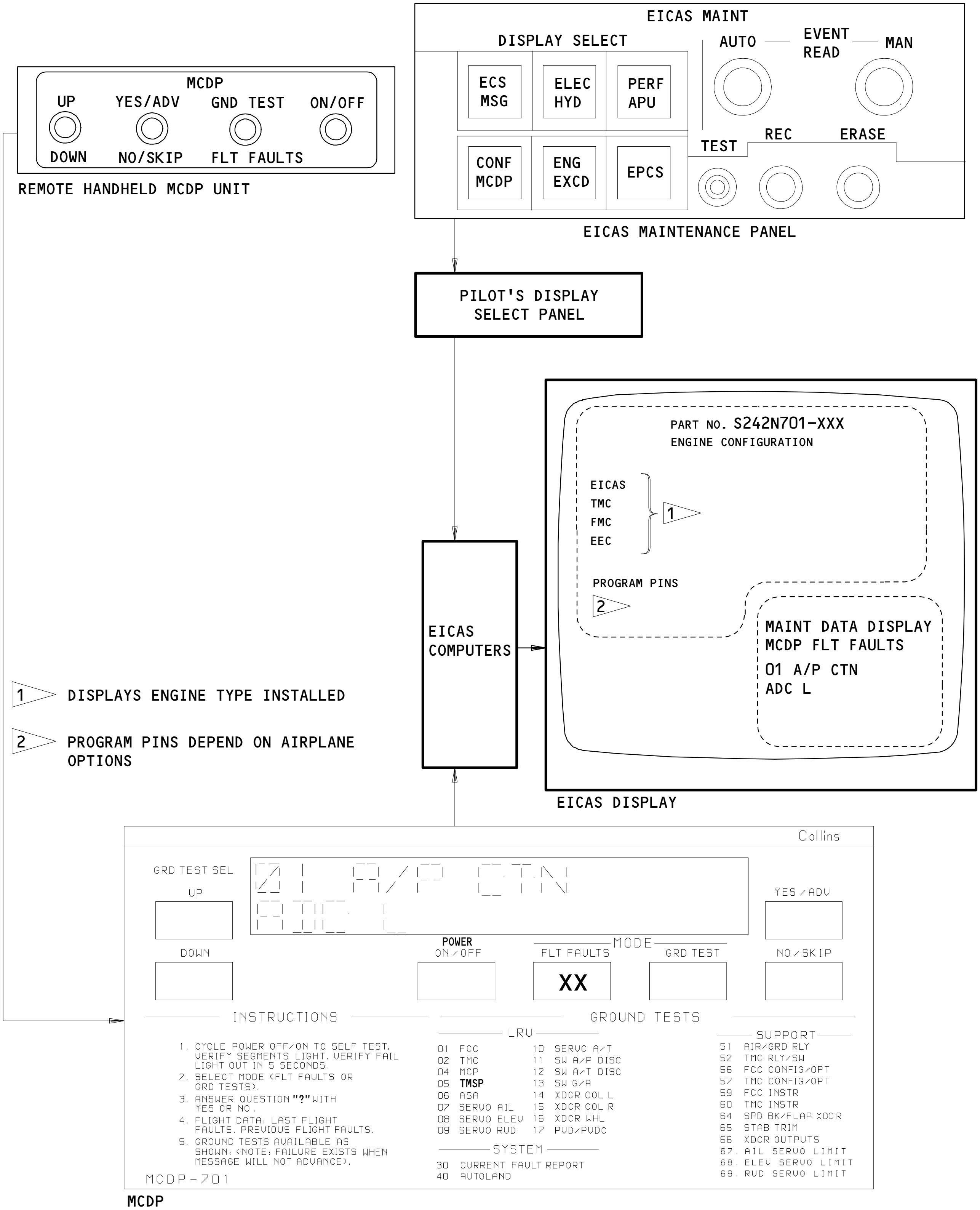
2) MAINT DATA DSPLY to indicate that maintenance data is being displayed. The MCDP mode message MCDP FLT FAULTS or MCDP GRD TEST is displayed below the MAINT DATA DSPLY message.

(5) MCDP Flight Faults Mode (Fig. 4)

(a) The MCDP receives flight fault data from six computers (3 FCC, 2 FMC, 1 TMC). The data is received on ARINC 429 data buses. Automatic transfer of flight fault data to the MCDP occurs when the following conditions are met: 1) either air/ground relay indicates an on-ground condition, 2) a successful automatic power-up and self-test of the MCDP is completed, 3) the Inertial Reference Unit (IRU) senses a ground speed of less than 40 knots. The IRU cannot be in the functional test mode or have the in-flight flag set.

(b) Each computer transmits flight fault data to the MCDP on an ARINC 429 data bus in a block of eight 32-bit words. The data block from each computer is initially stored in a Random Access Memory (RAM) until all valid computers have transmitted their fault data. After all the fault data is received, it is run through a fault consolidation routine and stored in non-volatile memory (EAROM). The MCDP can store up to 10 fault flight segments for a total of 300 flight faults. Each word in the data block is labeled either 356 for fault data and ground test data or 357 for interface fault data. The label 357 interface fault data is only acquired during ground test.

(6) MCDP Flight Fault Input Data Block (Fig. 7)



Remote MCDP Operation
Figure 6

(a) The label 356 word format consists of a block of eight words. All eight words must be present for the MCDP to accept the data.

1) Word one is a general information word that informs the MCDP the computers status and what to do with the incoming data.

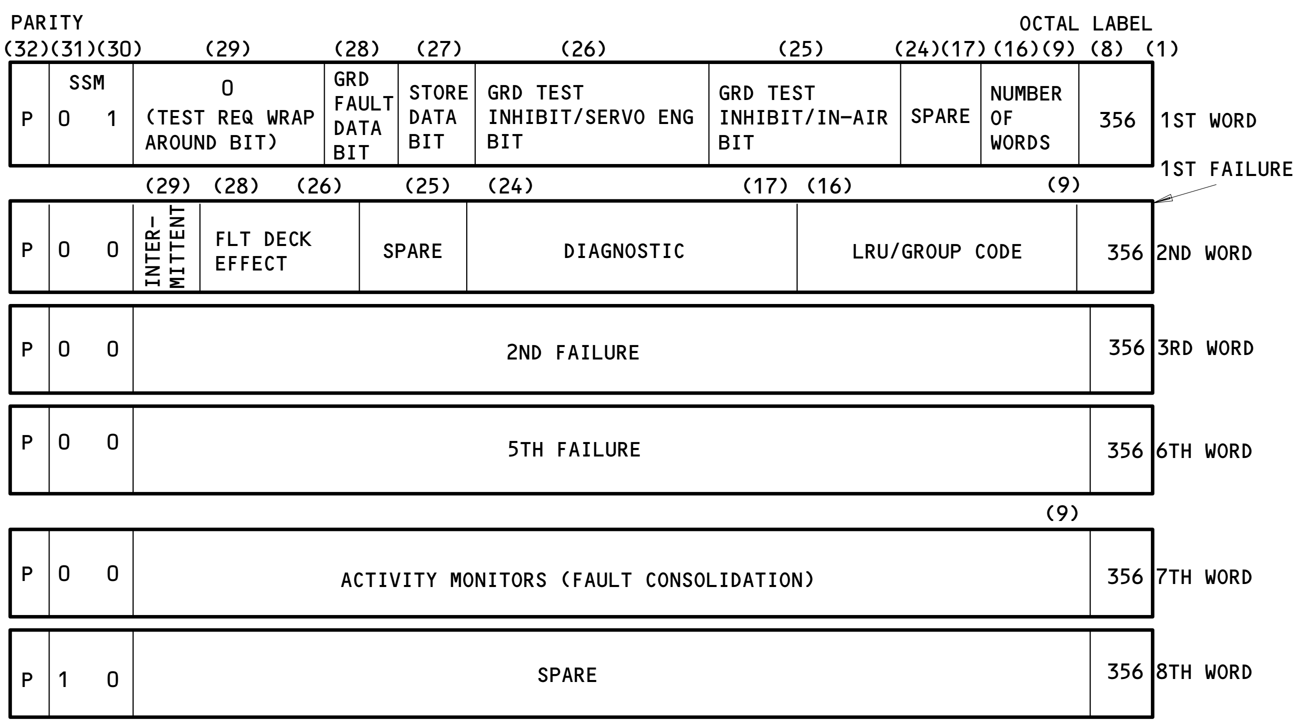
2) Words two through six contains one fault per word in LRU group code in word bits 9 through 16. A total of five faults per computer per flight can be transmitted to the MCDP. Each word also contains the flight deck effect and diagnostic code.

3) Word seven contains the status of all the activity monitors.

4) Word eight is not presently used.

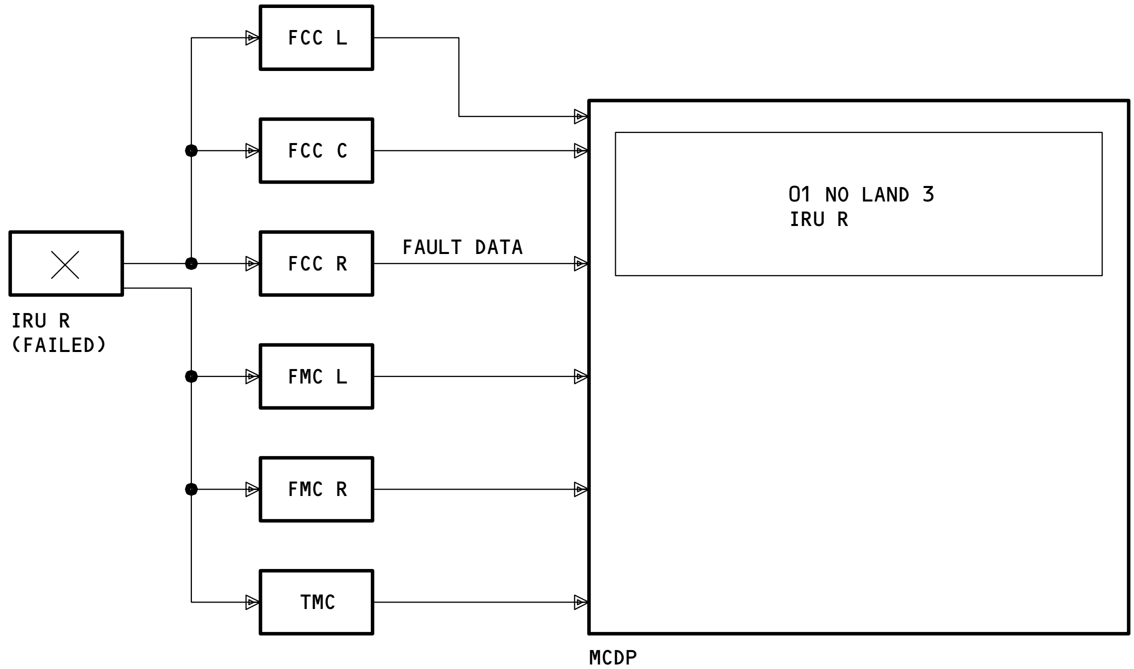
(7) LRU Fault Consolidation (Fig. 8)

(a) The Fault Data Management (FDM) calls the AUTO ON MANAGER routine when the AUTO ON MANAGER ENGAGE logic is generated. The engage logic is generated when the following conditions are met: Automatic Power-On mode is enabled; a successful limited self-test; the AUTO ON signal is provided after the initialization of the three minute timer. The AUTO ON MANAGER routine performs three functions: a Data Input Routine, a Flight Consolidation Routine, and a Store Fault Data Routine.



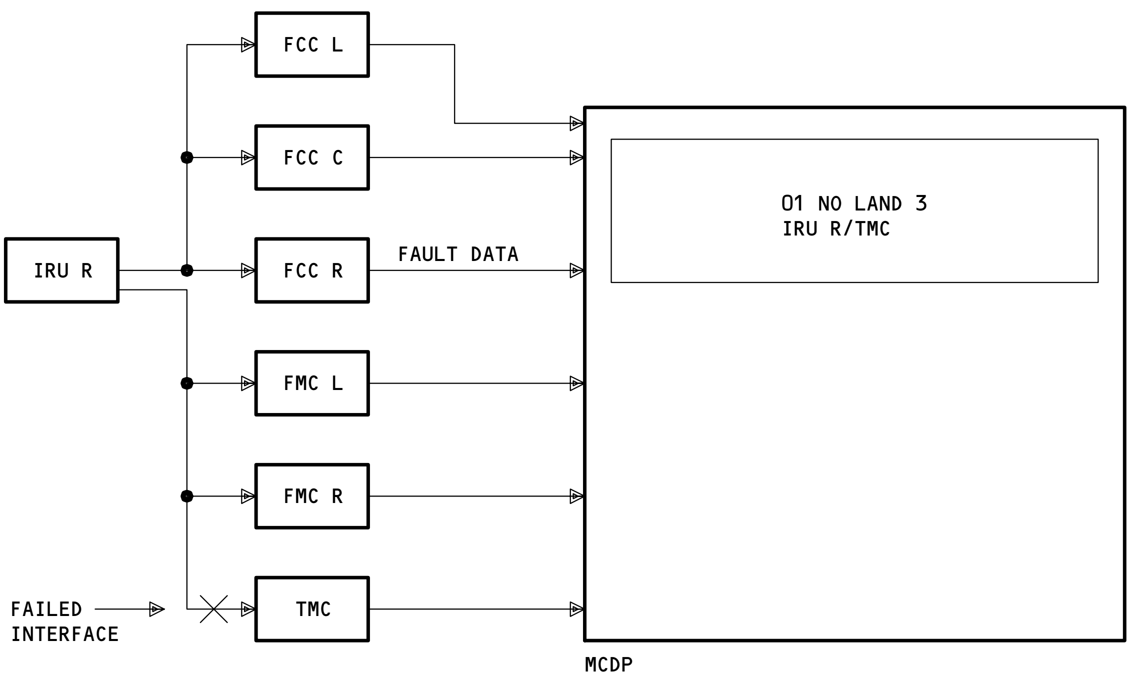
NOTE: CODES SHALL BE 2'S COMPLEMENT BINARY DATA WITH MSB ON LEFT

MCDP Flight Faults Input Data Block
Figure 7



$(\mathrm{FCC\ L} + \mathrm{FCC\ C} + \mathrm{FCC\ R}) \bullet (\mathrm{FMC\ L} + \mathrm{FMC\ R} + \mathrm{TMC}) = \underline{\mathrm{IRU\ R}}$ (SINGLE FAULT)

EXAMPLE - RIGHT IRU FAILURE



$(\overline{\mathrm{FCC\ L}} + \overline{\mathrm{FCC\ C}} + \overline{\mathrm{FCC\ R}}) \bullet (\overline{\mathrm{FMC\ L}} + \overline{\mathrm{FMC\ R}} + \mathrm{TMC}) = \underline{\mathrm{IRU\ R/TMC}}$ (GROUP FAULT)

EXAMPLE - GROUP FAULT

LRU Fault Consolidation
Figure 8

(b) Data Input Routine monitors the option pins to determine the computer configuration (center FCC and right FMC). It checks for fault data from any installed computer that does not have the desired ARINC label and store data bit set. It continues the check for fault data until all the store data bits are set or the AUTO ON three minute timer expires.

(c) Flight Consolidation Routine determines the flight deck effect message and LRU fault message associated with faults received from the six computers. The flight deck effect code is bits 26 through 28 of ARINC label 356 words 2 through 6. The flight deck effect message displayed on line one of the MCDP display is based on priority grouping of the messages (Ref to Fig. 11, Message Priorities Graph). The flight deck effect code is stored in the affected computer. The LRU group code is contained in bits 9 through 16 of ARINC label 356 words 2 through 6. The LRU message is displayed on line two of the MCDP display. The LRU message is based on 20 activity logic routines that monitor the incoming data. The activity logic routines determine when a fault is received from two or more computers. This prevents the fault from being displayed more than once per fault flight.

(d) Store Fault Data Routine is used to store the flight faults in non-volatile memory (EAROM). The store routine monitors word one of the ARINC word group for STORE DATA bit to be set. When the bit is set, the fault data, LRU GROUP CODE, and flight deck effect for that computer is stored in EAROM. The store routine controls the flight segment index by updating the previous flight faults segment number to include the latest flight. The segment numbers are assigned 00 to 99. The flight segment after 99 goes to 00 and the fault data that was in segment 99 is erased. The completion of the store routine resets the power flip-flop signal. This causes a power-down routine. If the store routine does not reset the flip-flop within three minutes, the three minute timer causes a power-down.

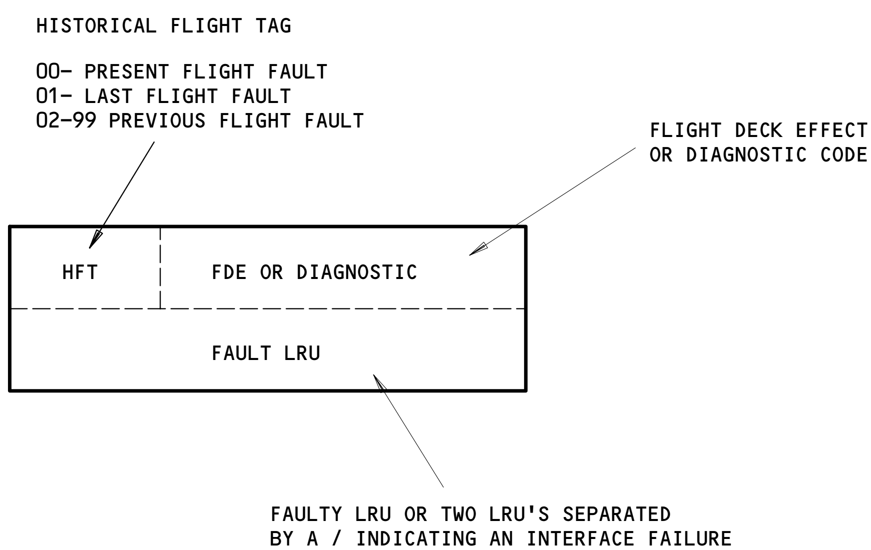
(8) MCDP Flight Fault Display Message Format (Fig. 9)

(a) The flight fault mode displays three groups of flight faults: LAST, PREVIOUS, and PRESENT. The PRESENT flight fault function is usable when the EICAS Maintenance Panel Flight Test Bypass Switch, located in the E/E bay, is positioned to BYPASS. The bypass switch is used for test flights only. A remote MCDP panel is required to operate the MCDP inflight.

(b) The flight fault review manager routine controls the flight fault display. The executive routine enables the review manager routine when it receives the FLT FAULT REVIEW ENABLE signal. The signal is generated after a manual power-up or flight fault mode is selected while in the ground test mode. The flight fault review enable is held high when the following conditions are present: MCDP power OK, ground test not required, ground test not enabled, FLT FAULTS required selected, or power latched ON flip-flop set, and auto ON not engaged. The flight fault mode is disabled when the power is turned off or the ground test mode is selected.

(c) Last Flight Faults

1) After the flight fault review routine is initialized, the FLT FAULT light/switch is turned on and the message LAST FLT FAULT? is displayed. The operator pushes the YES/ADV button to select the sub-mode being offered. Pushing the NO/SKIP pushbutton causes the MCDP to exit the LAST FLT FAULT routine and enter the PREVIOUS FLT FAULT routine.



MCDP Flight Fault Display Message Format
Figure 9

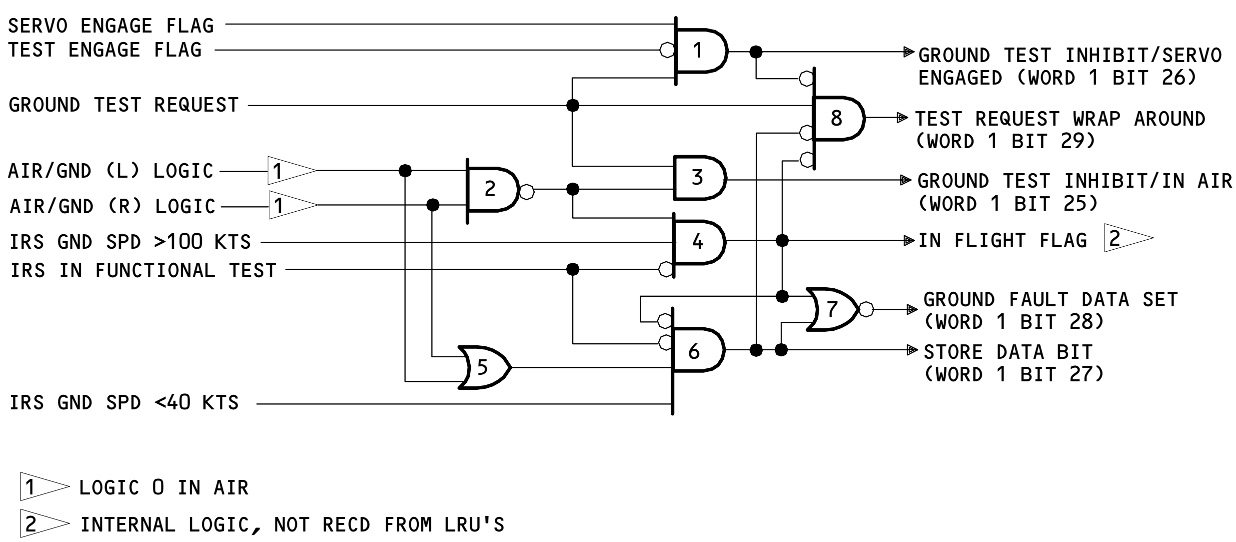
2) Before displaying the last flight faults, the routine checks all ARINC 429 ports for valid interface data. The MCDP displays NO INFC + computer and position for each inactive port. Each computer interface fault message is displayed by pushing the YES/ADV button to progress to the next fault.
3) After all interface faults have been displayed, the Last Flight Faults of the valid computers are displayed. The YES/ADV pushbutton is used to review them. The message ALL LAST FLIGHT FAULTS DSPLY is displayed after all faults have been displayed. Exiting the Last Flight Fault mode, before the ALL LAST FLIGHT FAULTS DSPLY message is displayed, is done by pushing the NO/SKIP pushbutton. Pushing the NO/SKIP or YES/ADV pushbutton when the ALL LAST FLT FAULTS DSPLY message is displayed causes the routine to advance to the Previous Flight Fault routine. The GRD TEST switch selects the ground test mode only when the aircraft is on the ground. The ON/OFF pushbutton causes the MCDP to power-down.
4) The Last Flight Fault messages have the flight segment 01 preceding the flight deck effect message on the first line of the display. The flight fault is displayed on line two.

(d) Display Previous Flight Faults
1) The operation of the Previous Flight Fault routine is similar to the Last Flight Fault routine with the exception that the flight segment number ranges from 02 to 99. The flight segment number indicates how many flights ago the fault occurred.
2) The MCDP displays NO PREV FLT FAULTS message when there are no previous flight faults within the 02 to 99 flight segments. When all the previous flight faults have been displayed, the MCDP displays ALL PREV FLT FAULTS DSPLY. While the MCDP can store faults for up to 99 flight segments, it can only store the last 10 fault flights (flights with recorded faults).

(9) Flight Fault Control Logic (Fig. 10)
(a) The store data bit (word 1, bit 27 of the fault data block format) must be set to logic one before the MCDP stores fault data in non-volatile memory. The store data bit is set when: the inflight flag set to logic zero, the IRU is not in functional test mode, at least one air/ground relay indicating an on ground condition, and an IRU is indicating a ground speed of less than 40 knots.

(b) A test request wrap-around bit is used to check the interfaces between the MCDP and FCC, FMC, and TMC. To set the test request wrap-around bit to logic one (word 1, bit 29 of the fault data block format), the following conditions must be present: autopilot servos not engaged, ground test request analog discrete signals is received by the FCC, FMC, or TMC, the store data bit is set to logic zero, and the inflight flag set is to logic zero. An interface fault is a fault that disables the interconnecting bus of two or more units.

(10) Flight Deck Effect Message Priorities (Fig. 11)



Flight Fault Control Logic
Figure 10

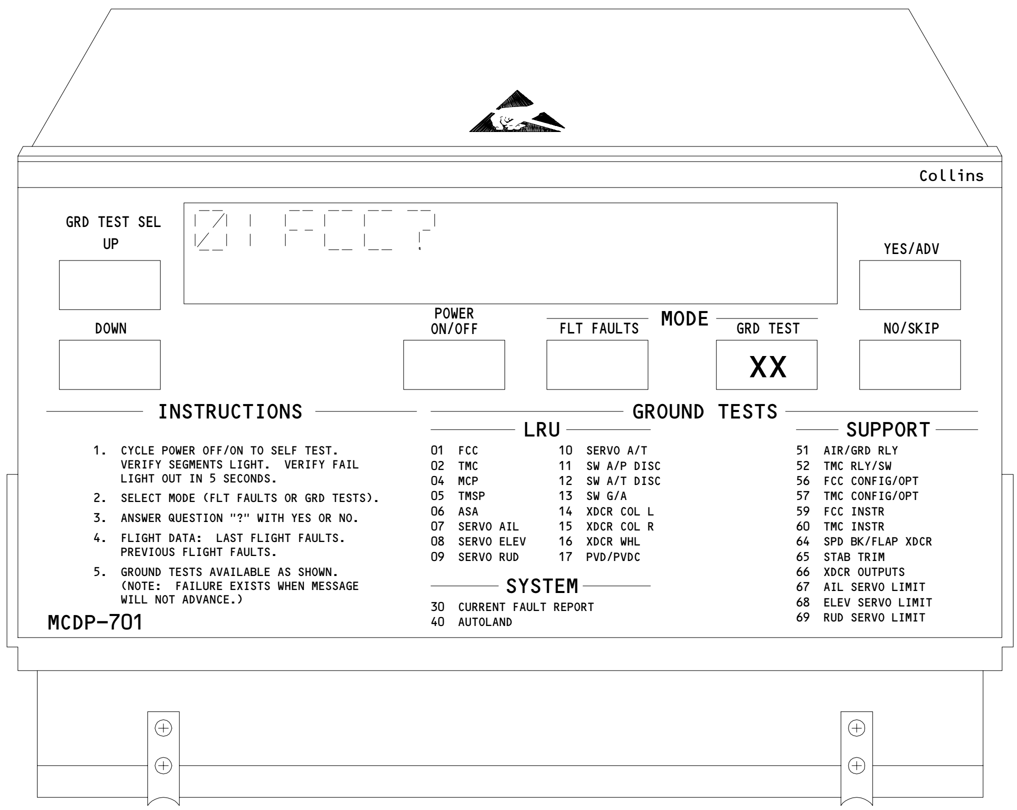
(a) The flight deck effect messages have a priority code from one to seven for each FCC, FMC, and TMC. Priority code one has the highest priority. The computers also have a priority order. The FCC has the highest priority; the FMC the lowest. The autopilot disconnect (A/P DISC) for an FCC has the highest priority and no flight deck effect display (NO FDE DSPLY) the lowest. Flight faults are listed in order of priority (highest to lowest). The FCC flight faults and flight deck effects are listed first, the TMC second, and FMC last.

(11) Ground Test Mode (Fig. 12)

(a) The ground test mode augments the flight fault capability by providing tests to further isolate and verify flight faults. The ground test mode also verifies system operation after maintenance. The three ground test groups (LRU tests, system tests, and support tests) complete tests in the Autopilot/Flight Director System (AFDS) and Thrust Management System (TMS). The LRU tests provide tests of individual LRU's and require operator interaction. The system tests are used to test large sections of the AFDS and TMS at one time. The Autoland System test is both automatic and manual. The Current Fault Report System test is completely automatic. The support tests are used to test only functions that interface with the AFDS and TMS.

PRIORITY	FCC	TMC	FMC
0	SPARE	SPARE	SPARE
1	A/P DISC	A/T DISC	FMC FAIL LT
2	A/P CTN	SPARE	V-L NAV FAIL
3	NO LAND 3	TMC DSPLY	CDU DSPLY
4	NO A/L	SPARE	SPARE
5	F/D BAR		SPARE
6	NO FDE DSPLY		
7	SPARE	F/S DSPLY	

Flight Deck Effect Message Priorities
Figure 11



Ground Test Mode
Figure 12

(b) LRU Tests
   1) The LRU tests provide a test armed interrogative message (e.g., 01 FCC TEST?). The test is engaged when the YES/ADV switch is pressed. The selected test first verifies that all prerequisites are available and automatically monitors the interfaces. Any missing prerequisite or interface fault is displayed.
   2) Any missing prerequisite or interface fault ends the test. If all prerequisites and interfaces are present, the test executes the appropriate automatic and manual tests. Automatic tests being run are indicated by an IN PROGRESS message. A fault message is indicated for each invalid test. Manual tests provide action messages, verify responses, and display failures. At completion of the test, a TEST COMPLETE message is displayed. The test is exited when the YES/ADV or NO/SKIP switch is pressed.

(c) System Tests
   1) The system tests provide a test armed interrogative message (e.g., 30 CURRENT FAULT REPORT?). The test is engaged when the YES/ADV switch is pressed. The selected test first verifies that all test prerequisites are available. Any missing prerequisite is displayed, and the test ends. The system test automatically arms and engages the appropriate LRU and support tests. A TEST COMPLETE message is displayed after all system tests have been completed and ground faults displayed. The test is exited when the YES/ADV or NO/SKIP switch is pressed.

(d) Support Tests
   1) The support tests provide a test armed interrogative message (e.g., 52 TMC RLY/SW TEST?). The test is engaged when the YES/ADV switch is pressed. The selected test first verifies that all test prerequisites are available. Any missing prerequisite is displayed and the test ends. A TEST COMPLETE message is displayed after completion of all operator interaction tests. The test is exited when the YES/ADV or NO/SKIP switch is pressed.

(12) MCDP Ground Test Mode (Fig. 4)
   (a) The ground test mode assists in isolating recorded flight faults to specific components. Ground test is used to verify system operation after the removal and installation of components. Ground tests require that all the computers and interfacing systems required for a specific test must be installed before the test can be done. Electrical and hydraulic power sources must be supplied to the required components. Ground test mode can be entered anytime after a manual power up, the FAIL annunciator light is turned off, and both AIR/GRD relays indicating on-ground. Two to three operators may be required for certain ground tests interaction.

(b) Ground Test Entry

1) The ground test mode is entered by pressing the GRD TEST light switch. The closed switch connects a ground that triggers a pulse circuit which generates an interrupt signal to the executive routine (data management). The data management executive routine generates GRD TEST REQ logic that latches GND TEST ENABLE logic when the following conditions are met: Both AIR/GRD relays sense an on-ground condition; MCDP power circuits are operating properly; MCDP is not in automatic-on mode; flight faults are not requested; and ground test is not already engaged. The ground test enabled latch is removed when the flight faults mode is selected; power is removed; or either AIR/GRD relay indicates IN-AIR (relay position does not apply if GRD TEST 51 is enabled).

2) The ground test executive routine turns on the GRD TEST lightswitch and removes existing display message during initialization. The ground test executive routine calls up a ground-test engagement check routine which generates analog ground-test command discretes to all the FCCs and the TMC. During the ground-test engagement check routine, the message 00 SYST ENG VFY// IN PROGRESS is displayed. There is a two second delay after the ground test command is sent. This allows the computers to initialize into their ground-test modes and send ground-test data to the MCDP. The ground test command routine verifies each computer for a correct ground-test engagement response by checking bits 25, 26, 29, and 32 of word one of label 356 data block.

3) Each of the four computers does the following tests upon receiving the analog ground test discrete bit. They check the AIR/GRD sensors for an on-ground condition and if on-ground, sets TEST REQUEST WRAP-AROUND bit 29 to one. A valid response from the computer sets the ARINC INPUT ERROR flag to zero and TEST REQUEST WRAP-AROUND bit 29 to one. An invalid response from the computer causes the MCDP to display a fault message. The fault message is determined by the status of the ARINC INPUT ERROR flag and digital discrete bits 25, 26, 29, and 32. Refer to GROUND TEST ENGAGE FAULT MESSAGE MATRIX graph (Fig. 10) for details.

4) After the completion of the engagement check, the ground-test executive routine calls up the computer self-test routine. While this routine is run, the message 00 COMP TEST// IN PROGRESS is displayed. The computer self-test routine commands computers that passed the engagement check, to execute their built-in self-tests. Two test commands are sent to the FCC's. The first initiates the FCC self-test and the other initiates an interface check between the FCC and its sensors. The TMC has three test commands sent to it. The first test initiates the TMC self-test, the second initiates interface checks between the TMC and its sensor. The third test initiates an interface check between the TMC and the autothrottle servomotor generator. The MCDP computer self-test routine checks the response data from each computer's AIM word label 356. Word one must contain the test routine number (bits 17-24). Word two must have test valid (bit 29) set to one. Self-tests are automatically terminated after five seconds.

5) The completion of computer self-tests causes the microprocessor to collect and store word label 357 fault data for later use during ground testing. The label 357 word format is in an eight word block. Word one contains the number of words to follow. Words two through eight contain the discrete interface data with 21 discretes per word.

6) The interface faults are displayed by the MCDP according to the GROUND TEST ENGAGE FAULT MESSAGE MATRIX (Ref Fig. 10). During the self-test routine, the MCDP displays the messge 00 INFO DATA IN// IN PROGRESS. After all interface data has been stored and associated fault data messages displayed, the ground test routine transfers control of the microprocessor to the test select routine. The transfer is indicated when 01 FCC TEST? is displayed.

(c) Test Selection Process

1) The operator selects the desired test from the list on the instruction panel by pressing the UP or DOWN ground test select switches (incremental or fast slewing). The NO/SKIP switch increments the test selections up. During the selection routine, each test is displayed with a question mark. The operator presses the YES/ADV switch to engage the test displayed. The engaged test causes the test select routine to display any required computers that failed the ground entry checks and display interface faults for the functioning computers.

2) If at least one of the required computers and its interface is valid, the test number select routine for the front panel test is called. This loads the test routine number and test parameters into the MCDP output data block. The data block is transmitted to the test computer on the ARINC 429 data bus. The test computer responds by setting the test routine number in progress bit, and system status bits in word 356 (Ref to Fig. 15, MCDP Input Test Data Block).

(13) Ground Test Engage Fault Message Matrix (Fig. 13)

(a) The ground test engage executive routine checks the status of the ARINC INPUT ERROR FLAG, WRAP-AROUND bit, and SERVO ON bit. This determines whether there is a test engage fault. The ground test is engaged when wrap-around bit 29 is set to one and the ARINC INPUT ERROR flag is set to zero. The ARINC INPUT ERROR flag is set to one if any of the following fault conditions are present:
1) No activity is on the bus for a specific label.
2) The specific label is received but not the data block.
3) The word parity is incorrect.
4) The number of words is incorrect (more or less than eight).

(b) If the error flag is set to one, the fault message NO INFC (computer name)// MCDP is displayed.

ARINC INPUT ERROR FLAG	DIGITAL DISCRETE			FAULT MESSAGE
	W/A BIT 29	IN AIR BIT 25	SERVO ON BIT 26	
1	X	X	X	NO INFC (*) // MCDP
0	0	1	X	IN AIR-NO GRD TEST (*)
0	0	0	1	SERVO ON-NO GRD TEST (*)
0	0	0	0	MCDP / (*) TEST CMD FAIL
0	1	0	0	NO FAULT MESSAGE

* DENOTES AFFECTED COMPUTER

X INDICATES VALUE OF BIT NOT USED IN CONSIDERATION TO DISPLAY FAULT MESSAGE

Ground Test Engage Fault Message Matrix
Figure 13

(c) Ground test inhibit, in-air bit 25 (label 356) is set to one when the computer receives an in-air indication from the air/ground relays. The fault message IN AIR-NO GRD TEST (computer name) is displayed, provided ARINC INPUT ERROR flag and WRAP-AROUND bit 29 are set to zero.

(d) Ground test inhibit, servo engaged bit 26 (label 356) is set to one when the analog ground test command discrete is generated due to an engaged autopilot control servo. Fault message SRVO ON-NO GRD TEST (computer name) is displayed, provided the ARINC INPUT ERROR flag, WRAP-AROUND bit 29, and IN AIR bit 25 are set to zero. Fault message MCDP/(computer name) TEST CMD FAIL is displayed when the ARINC INPUT ERROR flag and all three digital DISCRETE bits are set to ZERO. This is an indication that the analog ground test command discrete was not received by the affected computer.

(14) MCDP Output Test Data Block (Fig. 14)

(32)	(31)(30)	(29) 5 BITS (25)	(24) 8 BITS (17)	(16) 8 BITS (9)	FCC LABEL 8 BITS (8) (1)	TMC LABEL 8 BITS (8) (1)	
P	SSM	(P2) RESERVED	SIGN | PARAMETER P1	TEST ROUTINE NUMBER	OCTAL L 001, R 005, C 011	015	WORD 1
P	SSM	SIGN | (P5)	SIGN | (P4)	SIGN | (P3)	L 002, R 006, C 012	016	WORD 2
P	SSM	SIGN | (P8)	SIGN | (P7)	SIGN | (P6)	L 003, R 007, C 013	017	WORD 3
P	SSM	SIGN | (P11)	SIGN | (P10)	SIGN | (P9)	L 004, R 010, C 014	020	WORD 4

NOTE: PARAMETERS P1 THRU P9 & TEST ROUTINE NUMBER ARE BINARY DATA WITH MSB ON LEFT.
SIGN BIT IS THE LEFT MOST BIT IN EACH PARAMETER.

MCDP Output Test Data Block
Figure 14

(a) The MCDP output test data block informs the FCC's and TMC which test routine to perform and what parameters to use. Test routine numbers are stored in MCDP CPU memory. Each test number is associated with a selected ground test. The test routine number is assigned to bits 9 through 16 of word one of each output test data block. The data block structure consists of four 32-bit words for each FCC and TMC. Each FCC and TMC is assigned a non-ARINC label to bits 1 through 8 for each word address in its data block as follows:
1) 001, 002, 003, 004 address left FCC
2) 005, 006, 007, 010 address right FCC
3) 011, 012, 013, 014 address center FCC
4) 015, 016, 017, 020 address TMC

(b) The test routine numbers and Parameters (P) are binary coded with the Most Significant Bit (MSB) on the left and the sign codes are word bits 16, 24, and 29. The output data block uses the standard AIM Sign/Status Matrix (SSM) code. Test parameter bits and word assignments are as follows:
1) P1, bits 17 through 24 of word 1
2) P2, bits 25 through 29 of word 1 (reserved)
3) P3, bits 9 through 16 of word 2
4) P4, bits 17 through 24 of word 2
5) P5, bits 25 through 29 of word 2
6) P6, bits 9 through 16 of word 3
7) P7, bits 17 through 24 of word 3
8) P8, bits 25 through 29 of word 3
9) P9, bits 9 through 16 of word 4
10) P10, bits 17 through 24 of word 4
11) P11, bits 25 through 29 of word 4

(15) MCDP Input Test Data Block (Fig. 15)

(a) The MCDP INPUT test data block sends the test results back to the MCDP. The data block consists of eight 32-bit words with each word containing ARINC label 356 in word bits 1 through 8. The SSM code in bits 30 and 31 uses standard AIM data coding. Bit 32 checks parity. The specific bit definitions for each word are as follows:
1) Word one - bits 9 through 16 specify the number of words (always 8) in the data block. Bits 17 through 24 contains the test routine numbers from the MCDP. Bits 25 through 28 are spares. Bit 29 contains the test request wrap-around used during ground test entry functions.
2) Word Two - bits 9 through 27 contain the ground test comparison results. The results are discretes that indicate whether the following are within tolerance: left and right column null, left column forward and aft, right column forward and aft, control wheel null, left and right control wheel rotation, flap 0, 10, and 30 degree position, elevator detent. Bit 28 contains the test in progress status. Bit 29 contains the test valid status.

3) Word Three - Bits 9 through 27 contain Binary Numerical Representation (BNR) data. The BNR data indicates position-data in response to the test number sent in word two. Bits 25 through 29 are spares.
4) Words Four through Seven - Bits 9 through 29 are discrete data defining the test response conditions.
5) Word Eight - Bits 9 through 29 are spares.

(16) Ground Test Description (Fig. 16)

(a) LRU Tests Description

1) 01 FCC (Left, Right, and Center)
   a) Displays any FCC self-test faults and FCC interface faults from the associated system interfaces.
2) 02 TMC
   a) Displays any TMC self-test faults and TMC interface faults from associated system interfaces.
3) 04 MCP
   a) Displays any faulted FCC and TMC self-test faults. Displays FCC and TMC interface faults related to MCP.

   NOTE: In later MCP test steps failure of FCC L and FCC C, or FCC R will cause half lighting of lower half of MCP switch/light indicators. Failure of TMC will prevent EPR/N1/THR and SPD switch/lights from lighting.

(32)	(31)	(30)	(29)	(28)	(27)	(26)(25)	(24) ... (17)	(16) ... (9)	LABEL (8) ... (1)	
P	0	1	TEST REQ WRAP AROUND	SPARE	SPARE	SPARE	TEST ROUTINE NUMBER	NUMBER OF WORDS	356	1ST WORD

			(29)	(28)	(27) ... (9)		
P	0	0	TEST VALID BIT	IN PROGRESS BIT	TEST COMPARISON RESULTS	356	2ND WORD

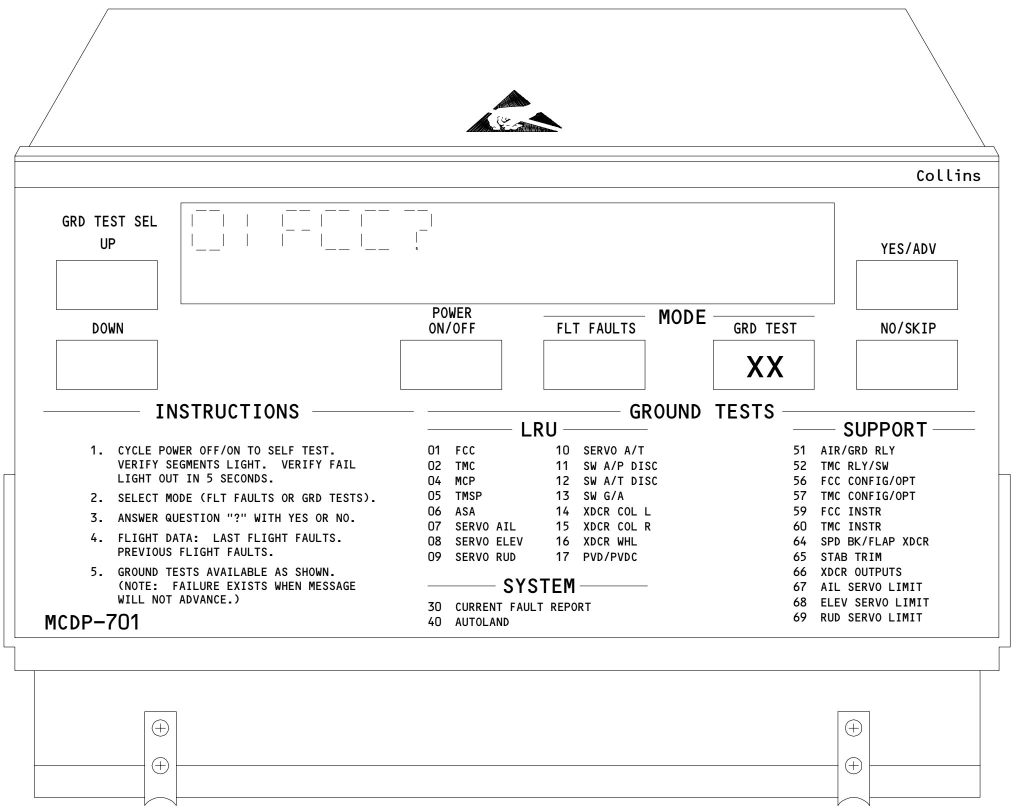
			(29) ... (25)	(24) ... (9)		
P	0	0	SPARE	BNR DATA (TRANSDUCER OUTPUT PARAMETER)	356	3RD WORD

P	0	0	DISCRETE DATA	356	4TH WORD

			(9)		
P	0	0	DISCRETE DATA	356	7TH WORD

P	1	0	SPARE	356	8TH WORD

MCDP Input Test Data Block
Figure 15



Ground Tests Description
Figure 16

b) SAS 050, 051, 150-157, 162-167 WITH SB 22-42, AND SAS 052-149, 158-161, 168-999; MTH 275-280 WITH SB 22-42, AND MTH 281-999;
Displays MCP A and B processor failures.
c) Checks the status of MCP program pin wiring for customer selected options.
d) Displays the interaction message for checking the lower lights in the MCP lightswitches.
e) Displays interaction messages for checking the ON and OFF position of the captain's flight director switch.
f) Displays interaction messages for checking the ARM and OFF position of the autothrottle ARM switch.
g) Displays the interaction message for checking the command function of the indicated airspeed/mach select switch.
h) Displays the interaction message for checking the command function of heading select switch.
i) Displays the interaction message for checking the ON and OFF positions of the first officer's flight director switch.
j) Displays interaction messages for checking the down and up position of the disengage bar.
k) Displays interaction messages for checking the command function for each position of the bank limit switch.
l) Displays interaction messages for checking the control function of the indicated airpseed/mach display rotary control.
m) Displays interaction messages for checking the control function of heading display rotary control.
n) Displays interaction messages for checking the control function of the vertical speed display rotary control.
o) Displays interaction messages for checking the control functions of the altitude select display rotary control.

4) 05 TMSP
a) Displays any TMC self-test faults related to TMSP.
b) Displays TMSP-to-TMC interface faults
c) Does a TMSP SW/MODE DISPLY test routine which checks the mode display and command function of continuous, climb, and cruise mode switching. Displays an interactive message at end of the test.
d) Does a TMSP TEMP SELECT test with interaction messages to check control function of temperature select control display.

5) 06 ASA
a) Displays any FCC self-test faults related to ASA (Autoland Status Annunciator).
b) Displays FCC interface faults related to ASA.
c) Checks all ASA displays and their reset function.

6) 07 SERVO AIL
   a) Displays any failed FCC self-test faults and FCC interface faults related to aileron servo.
   b) Provides interaction message to verify hydraulic pressure on to both wing and tail.
   c) Provides interaction message to verify aileron trim set to zero.
   d) Provides interaction message to push MCP CMD switches. This action automatically synchronizes the aileron and elevator servos to the present control surface position. Servos will not engage until servos synchronize.
   e) Wheel centers, then moves clockwise and then counterclockwise and back to center. Inboard ailerons are driven to a plus and minus 10 degree position. FCCs check that each position is within limits.
7) 08 SERVO ELEV
   a) Displays failed FCC's and autopilot elevator servo to FCC interface faults.
   b) Provides interaction message to verify hydraulic pressure on to both wing and tail.
   c) Provides interaction message to verify elevator trim set to zero.
   d) Provides interaction message for operator to verify elevator feel pitot-static pressure is zero.
   e) Provides interaction message to push MCP CMD switches. This action automatically synchronizes the aileron and elevator servos to the present control surface position. Servos will not engage until servos synchronize.
   f) Column centers, then moves aft (trailing edge up), then forward (trailing edge down) and back to center. Elevators are driven to a plus 20 degrees (trailing edge up) and a minus 9 degrees (trailing edge down) and the FCCs check that each position is within limits.
8) 09 SERVO RUD
   a) Displays any FCC self-test faults and FCC interface faults related to rudder servo.
   b) Provides interaction message to verify hydraulic pressure on to both wing and tail.
   c) Provides interaction message to verify rudder trim set to zero.
   d) Provides interaction message to push MCP CMD switches. This action automatically synchronizes the aileron and elevator servos to the present control surface position. Servos will not engage until servos synchronize.
   e) Rudder centers, then moves rudder trailing edge right, then left and back to center. Rudder driven to a plus and minus 20 degrees and FCCs check that each position is within limits.

NOTE: Airplanes with -133 FCCs and option group 1 with a value of 6; if the RUD SERVO X FAIL (X = L, C, R) message shows intermittenly when you do this ground test, ignore this message unless the RUD SERVO X FAIL (X = L, C, R) message is shown in the last flight faults or previous flight faults with the diagnostic code 206.

9) 10 SERVO A/T
   a) Displays any TMC self-test faults and interface faults related to autothrottle.
   b) Initializes the autothrottle to the aft position, drives the throttle forward, then drives the throttle aft. The test verifies that the throttle drives at the test specified rate and indicates the correct position.
10) 11 SW A/P DISC
   a) Displays any faulty FCC self-test faults and FCC disconnect switch-to-FCC interface faults related to A/P DISC switch.
   b) Instructs the operator to close the captain's and first officer's autopilot disconnect switches. The switch output is monitored for autopilot disconnect command response.
11) 12 SW A/T DISC
   a) Displays any TMC self-test faults and interface faults related to A/T DISC switch.
   b) Instructs operator to close the left and right autothrottle disconnect switches. Switch output is monitored for autothrottle disconnect command response.
12) 13 SW G/A
   a) Displays any FCC and TMC self-test faults, and FCC and TMC interface faults related to G/A switch.
   b) Instructs the operator to close the captain's and first officer's Go-Around (G/A) switches. The G/A switch command response is monitored at each FCC and TMC.
13) 14 XDCR COL L
   a) This test does not apply to this aircraft configuration. If test selected, MCDP displays 14 NO TEST THIS A/C CONFIG.
14) 15 XDCR COL R
   a) This test does not apply to this aircraft configuration. If test selected, MCDP displays 15 NO TEST THIS A/C CONFIG.
15) 16 XDCR WHL
   a) This test does not apply to this aircraft configuration. If test selected, MCDP displays 16 NO TEST THIS A/C CONFIG.

16) 17 PVD/PVDC
   a) This test does not apply to this aircraft configuration. If test selected, MCDP displays 17 NO TEST THIS A/C CONFIG.

(b) System Tests Description
   1) 30 CURRENT FAULT REPORT
      a) Displays all LRU and interface faults detected and stored in the FCCs and TMC while the MCDP operates in the ground test mode. Ground fault messages and diagnostic codes are the same as flight fault messages except the flight deck effect and intermittent bit status are not displayed.
      b) Provides interaction message to allow operator to skip Interface Fault messages.
      c) Provides interaction message to allow operator to select Current Ground Fault display.
   2) 40 AUTOLAND
      a) Displays the current ground faults and interface faults related to the autoland system. The autoland related interfaces are automatically monitored. It automatically performs LRU and support tests related to the autoland system that do not require interaction with the operator.
      b) Automatically peforms LRU and support tests related to the autoland system that require no manual preconditioning except for applying electrical power.
      c) Automatically performs LRU and support tests related to the autoland system that require manual preconditions but no hydraulic power.
      d) Performs manual interface checks related to the autoland system.
      e) Performs manual LRU and support tests related to the autoland system.
      f) Performs A/P servo and stab trim tests.

(c) Support Test Description
   1) 51 AIR/GRD RLY
      a) Displays any FCC and TMC self-test faults, and any interface faults related to air/ground relays.
      b) Interaction test of the landing gear system 1 air/ground relay and its interface to the FCCs and TMC.

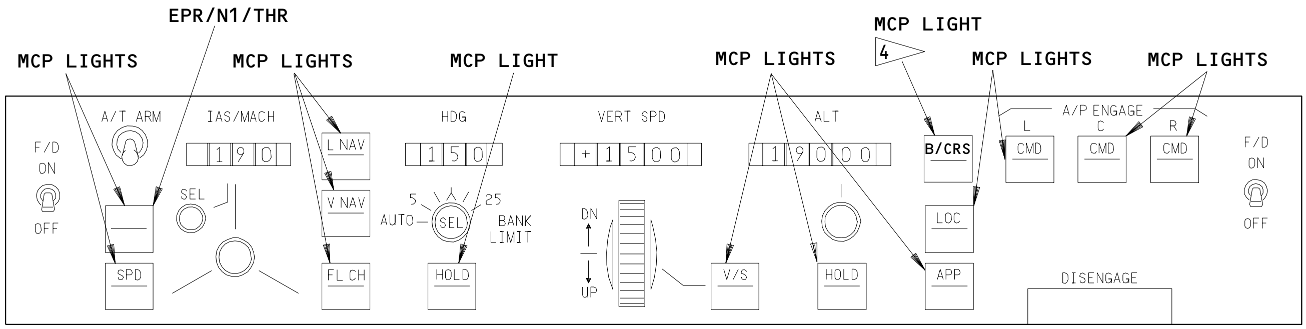
c) Interaction test of the landing gear system 2 air/ground relay and its interface to the FCCs.

2) 52 TMC RLY/SW
   a) Displays any TMC self-test faults and TMC interface faults related to Environmental Control System (ECS) relay.
   b) Tests for discrete output signals from the Electrical Systems Card File #1 (P-50 Panel), left and right Electronic Engine Controls (EEC) discrete card assemblies, and captain's FMC alternate select switch.
3) 56 FCC CONFIG/OPT
   a) Displays any FCC self-test faults and FCC interface faults related to parity.
   b) Checks the status of the FCC program pin wiring for airplane and hardware configurations and customer selected options.
4) 57 TMC CONFIG/OPT
   a) Displays any TMC self-test faults and TMC interface faults related to parity.
   b) Checks status of TMC program strapping for engine and airplane configuration, autothrottle, and thrust limit customer selected options.
5) 59 FCC INSTR
   a) Displays any FCC self-test faults.
   b) Sends test routine command and parameters to FCCs, and displays operator interaction message.
   c) Tests operation of captain's and first officer's MCP flight director switches and the ability of FCCs to move the pitch command bar 10 degrees. Checks each valid FCC separately.
6) 60 TMC INSTR
   a) Displays any TMC self-test faults.
   b) Sends test routine number and parameters to TMC and displays operator action message.
   c) Tests the ability of the TMC to control the captain's and first officer's EPR instruments target index. EPR target index driven to a test value of 1.00.
7) 64 SPD BK/FLAP XDCR
   a) Displays any TMC and FCC self-test faults and SLAT SW and FLAP XDCR related interface faults.
   b) Displays a VFY HYD ON message.
   c) Provides interaction messages to check the FCCs sensing of the speedbrakes down, armed, and up.
   d) Displays action messages to set flap lever to 25, 15,1, and 0.
   e) Tests operation of slat position sensing switch, flap position sensing transducer, and switch/transducer-to-FCC and TMC interfaces. The flap switch and flap transducer operation is checked when the flap lever is positioned to 25, 15, 1, and 0 units.

8) 65 STAB TRIM
   a) Displays any FCC self-test faults and FCC interface faults related to stab trim.
   b) Provides action message VFY SAM ON and VFY HYD ON.
   c) Tests the ability of the FCC to command the Stabilizer Trim and Aileron Lockout Module (SAM) to drive the horizontal stabilizer UP and DOWN. It checks the FCC-to-SAM interface.
9) 66 XDCR OUTPUTS
   a) Displays any FCC and TMC self-test faults, and interface faults related to the transducers.
   b) Displays position of horizontal stabilizer, aileron servo, aileron surface, elevator servo, elevator surface, rudder servo, rudder surface, and flaps to the FMC and TMC.
10) 67 AIL SURF LIMIT
   a) Displays any FCC self-test faults and FCC interface faults related to aileron servos.
   b) Provides action messages VFR HYD ON and A/P ENG LCR TO CMD (for valid FCC's only) (LCR = Left, Center, Right).
   c) Drives aileron to its positive limit if operator responds YES/ADV to action message AIL POS LIM?.
   d) Displays aileron surface position in degrees.

      AIL SURF DEG
      ±XX.X ±XX.X ±X.X, where XX.X = decimal number
11) 68 ELEV SURF LIM
   a) Elevator surface limit test same as aileron limit test.
12) 69 RUD SURF LIM
   a) Rudder surface limit test same as aileron limit test.

(17) Ground Test - LRU - 04 MCP (Steps 1-5) (Fig. 17)
   (a) The MCP test checks the function of all lightswitches, controls, and displays on the AFCS mode control panel using a preprogrammed test routine with operator interaction. The tests are divided into three sections: testing lighted switches, testing unlighted switches, and testing displays. A TEST COMPLETE message is displayed after all test steps are completed.
      1) The lighted switches are tested by the MCDP sending a test command to the FCC's on ARINC 429 data buses. The FCC's send necessary data (on ARINC 429 data buses) to the MCP's lightswitches. This turns on all switch/lights. The operator presses each lighted switch; this is sensed by each FCC, and they send back a command to turn the light off.



STEP	MCDP TEST REQUIREMENT	ACTION/INFORMATIVE MESSAGE	FAIL MESSAGE
1.	PROVIDE TEST ARMED MESSAGE ⇨	04 MCP TEST?	
2.	DISPLAY ALL FAULTED FCCs SEQUENTIALLY. ADVANCE TO NEXT STEP AUTOMATICALLY FOR NO COMPUTER FAULTS OR WHEN ALL FAULTS DISPLAYED ⇨	04 IN PROGRESS	04 FCC [1] FAIL 04 TMC FAIL
3.	DISPLAY MCP INTERFACE FAULTS SEQUENTIALLY. ADVANCE TO NEXT STEP AUTOMATICALLY FOR NO INTERFACE FAULTS OR WHEN ALL FAULTS DISPLAYED ⇨	04 IN PROGRESS	MCP BUS IN A/P DISC SW MCP A/P ENG DISC MCP A/P ARM IN FCC TO MCP BUS TMC BUS IN
3A.	[2] DO A TEST FOR MCP A PROCESSOR. ADVANCE TO NEXT STEP AUTOMATICALLY IF NO FAILURE OCCURS OR FAULT MESSAGE IS DISPLAYED	04 IN PROGRESS	04 MCP FAIL/MCP A FAIL
3B.	[2] DO A TEST FOR MCP B PROCESSOR. ADVANCE TO NEXT STEP AUTOMATICALLY IF NO FAILURE OCCURS OR FAULT MESSAGE IS DISPLAYED	04 IN PROGRESS	04 MCP FAIL/MCP B FAIL
3C.	[3] DECODE AND DISPLAY MCDP CONFIGURATION DATA	04 MCP CONFIG L/R 03 03	
4.	PROVIDE ACTION MESSAGE ⇨	04 MCP MAN TEST?	
5.	PROVIDE ACTION MESSAGE AND ADVANCE AUTOMATICALLY WHEN IN PROGRESS BIT CHANGES FROM 1 TO 0 OR ⇨	04 PUSH OFF MCP LIGHTS-ADV	

⇨ ADVANCE TO NEXT STEP FOR YES/ADV

[1] DENOTES LEFT, RIGHT, OR CENTER COMPUTER

[2] AIRPLANES WITH -304 AND SUBSEQUENT MCDP

[3] AIRPLANES WITH -304 SUBSEQUENT MCDP AND MCP OPTION GREATER THAN 01

[4] AIRPLANES WITH BACKCOURSE SWITCH INSTALLED

Ground Test - LRU - 04 MCP (Steps 1-5)
Figure 17

2) The testing of the unlighted switches is accomplished by the operator responding to instructions displayed by the MCDP. The test routine monitors AIM word 356 supplied from the FCC and detects the switch or control action. The test is automatically advanced to the next step after each correct test result. The test stops if the test result is incorrect. Pressing the YES/ADV switch continues the test after the test routine stops.
3) The MCP displays are tested by the operator responding to MCDP display messages and verifying that the display is in accordance with the MCDP message. Failure of the display to respond in accordance with the MCDP message indicates that the display is faulty. No fault messages are displayed on the MCDP. The operator must press the YES/ADV switch after each test, whether the result is correct or not, in order to continue the test.

(b) MCP Test Steps 1-5 Description

1) Step 1 - The MCDP CPU test select routine monitors the GRD TEST SEL UP (or NO/SKIP) and DOWN switches for a command to select another test until the required test arm message 04 MCP TEST? is displayed. The operator presses the YES/ADV switch to engage the 04 MCP TEST.
2) Step 2 - The MCDP displays 04 IN PROGRESS while checking its memory to see if any faulty FCCs and TMC were detected during the ground test entry function. Each fail message, 04 FCC (L, R, or C) FAIL, is displayed in sequence by pressing YES/ADV for the next fault. If there are no faulty FCCs, the test automatically advances and a checks for interface fault data stored in memory during the ground test entry functions. The 04 IN PROGRESS message is displayed while the CPU is checking for interface fault data.
3) Step 3 - An interface fault message is displayed for each faulty FCC-to-MCP interface. Subsequent interface faults are displayed by pressing the YES/ADV switch. The MCDP can display any or all of the following interface fault messages:
   a) The MCP BUS IN message indicates that the ARINC 429 data bus from the MCP to the FCC is faulty.
   b) The MCP A/P ENG DISC message indicates that the 28 vdc servo power engage interface from the FCC to the MCDP is faulty.
   c) The A/P DISC SW message indicates that the control wheel A/P disconnect switch or interface is faulty.
   d) The MCP A/P ARM IN message indicates that the +28 vdc servo power arm interface from the FCC to the MCP is faulty.
   e) The FCC TO MCP BUS IN message indicates that the ARINC 429 data bus from the FCC to the MCP is faulty.
   f) The TMC BUS IN message indicates that the ARINC 429 data bus from the TMC to the FCC is faulty.

4) AIRPLANES WITH -304 AND SUBSEQUENT MCDP;
These steps occur:
a) Step 3A - If there are no interface faults, the test routine automatically advances and displays 04 IN PROGRESS while checking for MCP A processor failure. The message 04 MCP FAIL/MCP A FAIL is displayed if a failure is detected. If there is no failure, the test automatically advances.
b) Step 3B - The MCDP displays 04 IN PROGRESS while checking for MCP B processor failure. The message 04 MCP FAIL/MCP B FAIL is displayed if a failure is detected. If there is no failure, the test automatically advances.

5) Step 3C - If there are no faults, the test routine automatically advances to decode MCP configuration data. 04 MCP CONFIG L/R is displayed on the top line with a numerical readout of the MCP customer option pin configuration displayed on the second line. Pressing the YES/ADV switch advances to the MCP manual test.

6) Step 4 - If there are no interface faults, the test routine automatically advances to the MCP test and displays 04 MCP MAN TEST?. Pressing the YES/ADV switch enables the 04 MCP manual test.

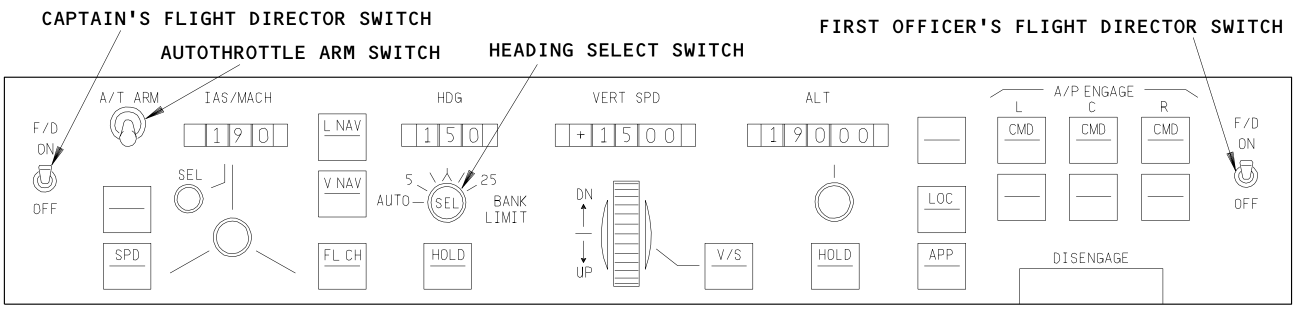
7) Step 5 - The 04 manual test starts by testing all lighted switches. The MCDP commands the FCCs to do the MCP lighted switch test. The FCC turns on all lightswitches that are controlled by the FCCs. The MCDP then displays action message 04 PUSH OFF MCP LIGHTS and the operator pushes each MCP lightswitch. If the lightswitch pushed is functioning, the FCC senses that the switch has been pressed and sends a signal back to the MCP to turn that lightswitch off. If pressing the lightswitch does not cause the light to turn off, it is an indication that the lightswitch circuit is faulty. No fault message is displayed. The test advances to the next step when the FCC senses that all switches have been tested, or the YES/ADV switch is pressed.

(18) Ground Test - LRU - 04 MCP (Steps 6-12) (Fig. 18)

(a) MCP Test Steps 6-12 Description

1) Test steps (6-20) are verified by the CPU automatically advancing to the next test step. The test does not advance if the CPU does not verify that the switch tested is in the correct position. To continue the test past a fault, the operator presses the YES/ADV switch. There is no fault message for unlighted switch tests.

2) Step 6 - Action message 04 SET CAPT F/D SW ON instructs the operator to position the captain's flight director switch to ON. The CPU monitors AIM label 356, word 6, bit 9 to be set to logic one. Bit 9 set to one indicates to the CPU that the captain's F/D switch is in the ON position. The CPU then advances to the next test step.



STEP	MCDP TEST REQUIREMENT	ACTION/INFORMATIVE MESSAGE	FAIL MESSAGE
6.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF CAPT F/D SWITCH VERIFIED OR ⇨	04 SET CAPT F/D SW ON	1
7.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN OFF STATE OF CAPT F/D SWITCH VERIFIED OR ⇨	04 SET CAPT F/D SW OFF	1
8.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF A/T SWITCH VERIFIED OR ⇨	04 SET A/T SW ARM	1
9.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN OFF STATE OF A/T SWITCH VERIFIED OR ⇨	04 SET A/T SW OFF	1
10.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF HDG SEL SWITCH VERIFIED OR ⇨	04 HOLD DN HDG SEL SW	1
11.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF F/O'S F/D SWITCH VERIFIED OR ⇨	04 SET F/O F/D SW ON	1
12.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN OFF STATE OF F/O'S F/D SWITCH VERIFIED OR ⇨	04 SET F/O F/D SW OFF	1

⇨ ADVANCE TO NEXT STEP FOR YES/ADV

1 NO MESSAGE PROVIDED, TEST DOES NOT ADVANCE FOR FAULT CONDITION

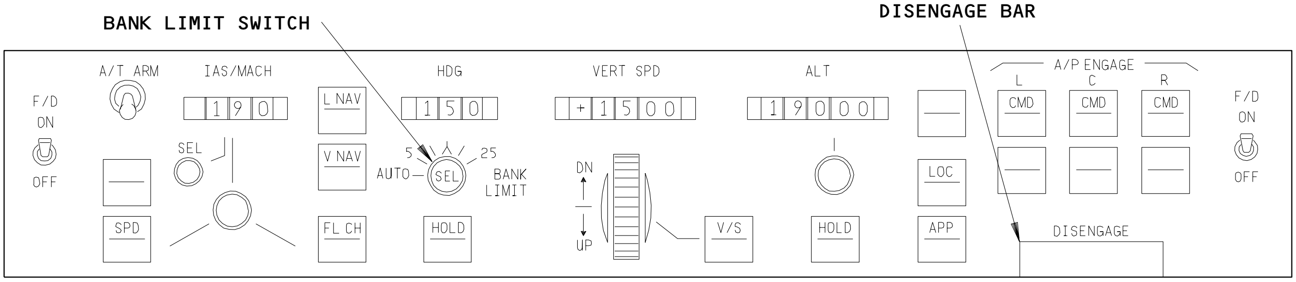
Ground Test - LRU - 04 MCP (Steps 6-12)
Figure 18

3) Step 7 - Action message 04 SET CAPT F/D SW OFF instructs the operator to switch off the captain's flight director switch. The CPU monitors AIM label 356, word 6, bit 9 to be set to zero. Bit 9 set to zero indicates to the CPU that the captain's flight director is turned off. The CPU advances to the next step.
4) Step 8 - Action message 04 SET A/T SW ARM instructs the operator to position the autothrottle switch to A/T ARM. The CPU verifies that the switch position is A/T ARM when it senses that AIM label 356, word 6, bit 10 is set to logic one. When the switch position is verified, the CPU advances to the next test step.
5) Step 9 - Action message 04 SET A/T SW OFF instructs the operator to position the autothrottle switch to off. When the CPU verifies that AIM label 356, word 6, bit 10 has been set to logic zero, the CPU advances the test routine to the next step.
6) Step 10 - Action message 04 HOLD DN HDG SEL SW instructs the operator to press the heading select switch. The CPU verifies that AIM label 356, word 6, bit 24 is set to logic one. The CPU advances to the next test step.
7) Step 11 - Action message 04 SET F/O F/D SW ON instructs the operator to position the first officer's flight director switch to ON. The CPU verifies that the AIM label 356, word 6, bit 13 is set to logic one. The CPU advances to the next test step.
8) Step 12 - Action message 04 SET F/O F/D SW OFF instructs the operator to turn off the first officer's flight director switch. The CPU verifies that the AIM label 356, word 6, bit 13 is set to logic zero. The CPU advances to the next test step.

(19) Ground Test - LRU - 04 MCP (Steps 13-20) (Fig. 19)

(a) MCP Test Steps 13-20 Description

1) Step 13 - Action message 04 PUSH DISENG BAR DN instructs the operator to position the disengage bar down. The CPU verifies that the AIM label 356, word 6, bit 14 is set to logic one. The CPU advances to the next test step.
2) Step 14 - Action message 04 PUSH DISENG BAR UP instructs the operator to position the disengage bar up. The CPU verifies that AIM label 356, word 6, bit 14 is set to logic zero. The CPU advances to the next test step.
3) Step 15 - Action message 04 SET BANK LIM TO 25 instructs the operator to position the bank limit selector switch to 25. The CPU verifies that AIM label 356, word 6, bit 18 is set to logic one. The test advances to the next test step.
4) Step 16 - Action message 04 SET BANK LIM TO 20 instructs the operator to position the bank limit selector switch to 20. The CPU verifies that AIM label 356, word 6, bit 19 is set to logic one. The test advances to the next test step.



STEP	MCDP TEST REQUIREMENT	ACTION/INFORMATIVE MESSAGE	FAIL MESSAGE
13.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN OFF STATE OF DISENGAGE BAR VERIFIED OR ⇨	04 PUSH DISENG BAR DN	[1]
14.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF DISENGAGE BAR VERIFIED OR ⇨	04 PUSH DISENG BAR UP	[1]
15.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN LIMIT = 25 FROM BANK LIMIT SWITCH VERIFIED OR ⇨	04 SET BANK LIM TO 25	[1]
16.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN LIMIT = 20 FROM BANK LIMIT SWITCH VERIFIED OR ⇨	04 SET BANK LIM TO 20	[1]
17.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN LIMIT = 15 FROM BANK LIMIT SWITCH VERIFIED OR ⇨	04 SET BANK LIM TO 15	[1]
18.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN LIMIT = 10 FROM BANK LIMIT SWITCH VERIFIED OR ⇨	04 SET BANK LIM TO 10	[1]
19.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN LIMIT = 5 FROM BANK LIMIT SWITCH VERIFIED OR ⇨	04 SET BANK LIM TO 5	[1]
20.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN LIMIT = AUTO FROM BANK LIMIT SWITCH VERIFIED OR ⇨	04 SET BANK LIM TO AUTO	[1]

⇨ ADVANCE TO NEXT STEP FOR YES/ADV

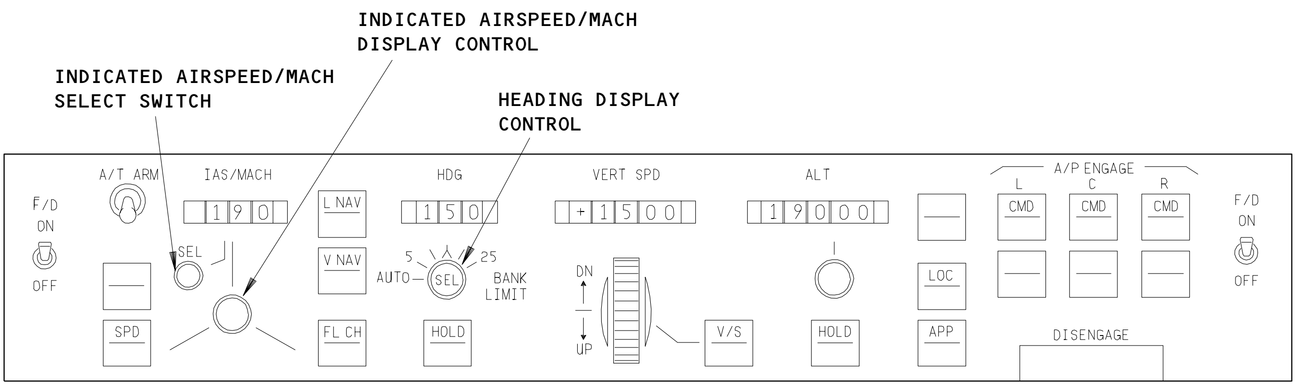
[1] NO MESSAGE PROVIDED, TEST DOES NOT ADVANCE FOR FAULT CONDITION

Ground Test - LRU - 04 MCP (Steps 13-20)
Figure 19

5) Step 17 - Action message 04 SET BANK LIM TO 15 instructs the operator to position the bank limit selector switch to 15. The CPU verifies that AIM label 356, word 6, bit 20 is set to logic one. The test advances to the next step.
6) Step 18 - Action message 04 SET BANK LIM TO 10 instructs the operator to set the bank limit selector switch to 10. The CPU verifies that AIM label 356, word 6, bit 21 is set to logic one. The test advances to the next step.
7) Step 19 - Action message 04 SET BANK LIM TO 5 instructs the operator to set the bank limit selector switch to 5. The CPU verifies that AIM label 356, word 6, bit 22 is set to logic one. The test advances to the next test step.
8) Step 20 - Action message 04 SET BANK LIM TO AUTO instructs the operator to position the bank limit selector switch to AUTO. The CPU verifies that ARINC label 356, word 6, bit 23 is set to logic one. The test advances to the next test step.

(20) Ground Test - LRU - 04 MCP (Steps 21-25) (Fig. 20)
(a) MCP Test Steps 21-25 Description
1) For test steps 21 and on, whether the results are correct or not, the operator must press the YES/ADV switch to advance to the next test.
2) Step 21 - Action message 04 PUSH IAS/MACH SEL SW instructs the operator to push the IAS/MACH SEL switch. The CPU verifies that the AIM label 356, word 6, bit 11 is set to logic one. The CPU advances to the next test step.
3) Step 22 - Action message 04 TURN IAS/MACH CCW - DSPLY DEC? instructs the operator to rotate the indicated airspeed/mach display control counterclockwise, and verify that the display decreases in value.
4) Step 23 - Action message 04 TURN IAS/MACH CW - DSPLY INC? instructs the operator to rotate the indicated airspeed/mach display control clockwise, and verify that the display increases in value.
5) Step 23A - Action message 04 PUSH IAS/MACH - DSPLY BLANK? instructs the operator to push the IAS/MACH SEL switch and verify that the IAS/MACH display window goes blank.
6) Step 24 - Action message 04 TURN HDG CCW - DSPLY DEC? instructs the operator to rotate the heading display control counterclockwise, and verify that the display decreases in value.
7) Step 25 - Action message 04 TURN HDG CW DSPLY INC? instructs the operator to rotate the heading display control clockwise, and verify that the display increases in value.

(21) Ground Test - LRU - 04 MCP (Steps 26-30) (Fig. 21)
(a) MCP Test Steps 26-30 Description
1) Step 26 - Action message 04 TURN VERT SPD UP - DSPLY INC? instructs the operator to rotate the vertical speed control toward UP and verify that the vertical speed display increases in value.

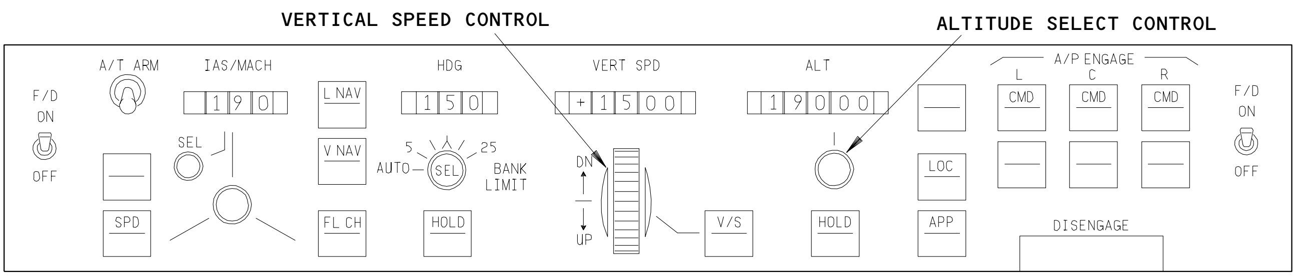


STEP	MCDP TEST REQUIREMENT	ACTION/INFORMATIVE MESSAGE	FAIL MESSAGE
21.	PROVIDE ACTION MESSAGE AND ADVANCE TO NEXT STEP WHEN ON STATE OF IAS/MACH SELECT SWITCH VERIFIED OR ⇨	04 PUSH IAS/MACH SEL SW	1
22.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY DECREASES ⇨	04 TURN IAS/MACH CCW-DSPLY DEC?	
23.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY INCREASES ⇨	04 TURN IAS/MACH CW-DSPLY INC?	
23A.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY BLANKS ⇨	04 PUSH IAS/MACH DSPLY BLANK?	
24.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY DECREASES ⇨	04 TURN HDG CCW-DSPLY DEC?	
25.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY INCREASES ⇨	04 TURN HDG CW-DSPLY INC?	

⇨ ADVANCE TO NEXT STEP FOR YES/ADV

1 NO MESSAGE PROVIDED, TEST DOES NOT ADVANCE FOR FAULT CONDITION

Ground Test - LRU - 04 MCP (Steps 21-25)
Figure 20



STEP	MCDP TEST REQUIREMENT	ACTION/INFORMATIVE MESSAGE	FAIL MESSAGE
26.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY INCREASES ⇨	04 TURN VERT SPD UP-DSPLY INC?	
27.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY DECREASES ⇨	04 TURN VERT SPD DN-DSPLY DEC?	
28.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY DECREASES ⇨	04 TURN ALT SEL CCW-DSPLY DEC?	
29.	PROVIDE ACTION MESSAGE AND VERIFY DISPLAY INCREASES ⇨	04 TURN ALT SEL CW-DSPLY INC?	
30.	PROVIDE TEST COMPLETE MESSAGE	04 TEST COMPLETE	

⇨ ADVANCE TO NEXT STEP FOR YES/ADV

Ground Test - LRU - 04 MCP (Steps 26-30)
Figure 21

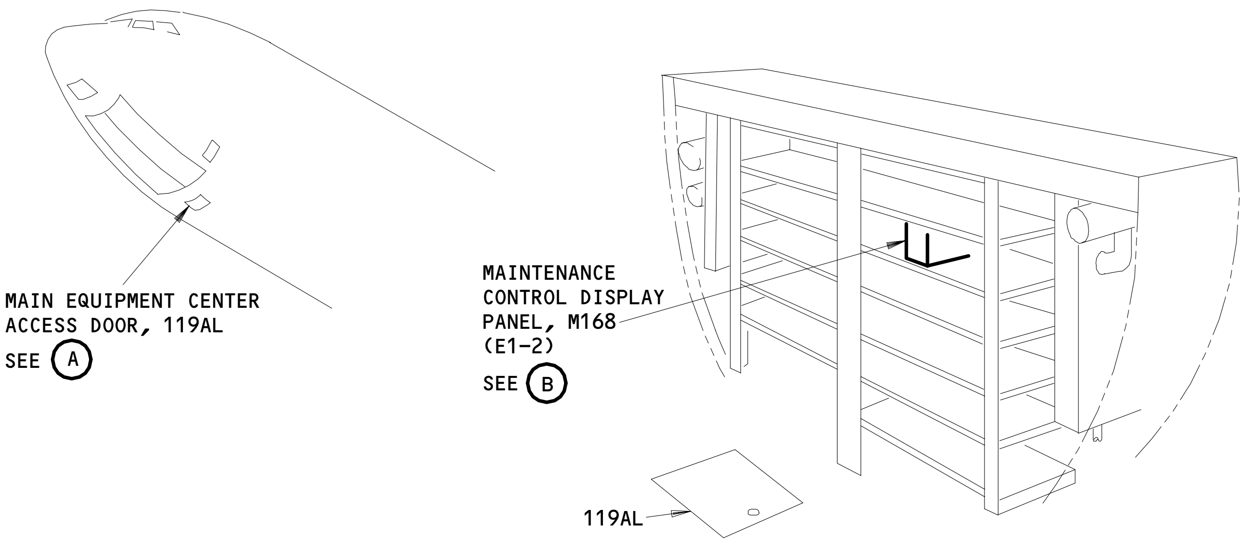
2) Step 27 - Action message 04 TURN VERT SPD DN - DSPLY DEC? instructs the operator to rotate the vertical speed control toward DN and verify that the vertical speed display decreases in value.
3) Step 28 - Action message 04 TURN ALT SEL CCW - DSPLY DEC? instructs the operator to rotate the altitude select control counterclockwise and verify that the altitude display decreases in value.
4) Step 29 - Action message 04 TURN ALT SEL CW - DSPLY INC? instructs the operator to rotate the altitude select control clockwise and verify that the altitude display increases in value.
5) Step 30 - Message 04 TEST COMPLETE is displayed when the 04 MCP test routine has ended. The completed test is exited and enters ground test select when either the YES/ADV or NO/SKIP switch is pressed. On entering the ground test select routine, the display shows the next ground test listed on the front panel.

MAINTENANCE MONITOR

COMPONENT	FIG. 102 SHT	QTY	ACCESS/AREA	AMM REFERENCE
CIRCUIT BREAKER - MAINT CONT DISPLAY, C520		1	FLT COMPT, P11 11U9	*
CONNECTOR, REMOTE MCDP CONTROL PANEL, D1447	2	1	FLT COMPT, P6	*
PANEL - MAINTENANCE CONTROL DISPLAY, M168	1	1	119AL MAIN EQUIP CTR, E1-2	22-41-01
RELAY - (FIM 31-01-36/FIG. 101) AIR/GROUND, SYS NO. 1, K142				
RELAY - (FIM 31-01-37/101) AIR/GROUND, SYS NO. 2, K201				
SWITCH - (FIM 31-41-00/101) MAINT ENABLE BYPASS, S612				

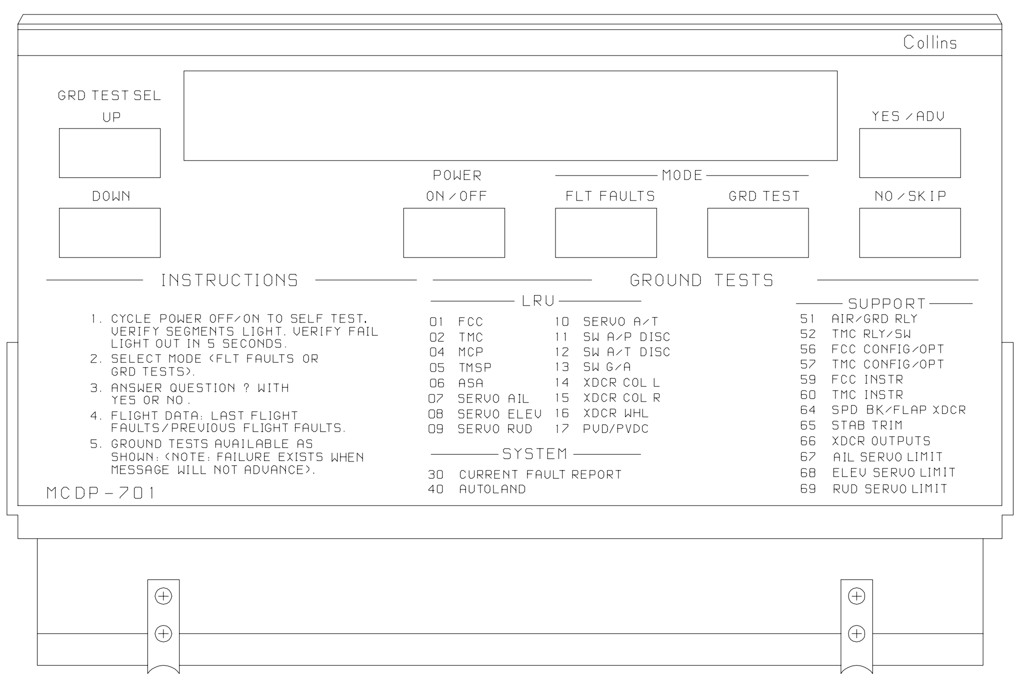
* SEE THE WDM EQUIPMENT LIST

Maintenance Monitor - Component Index
Figure 101



MAIN EQUIPMENT CENTER

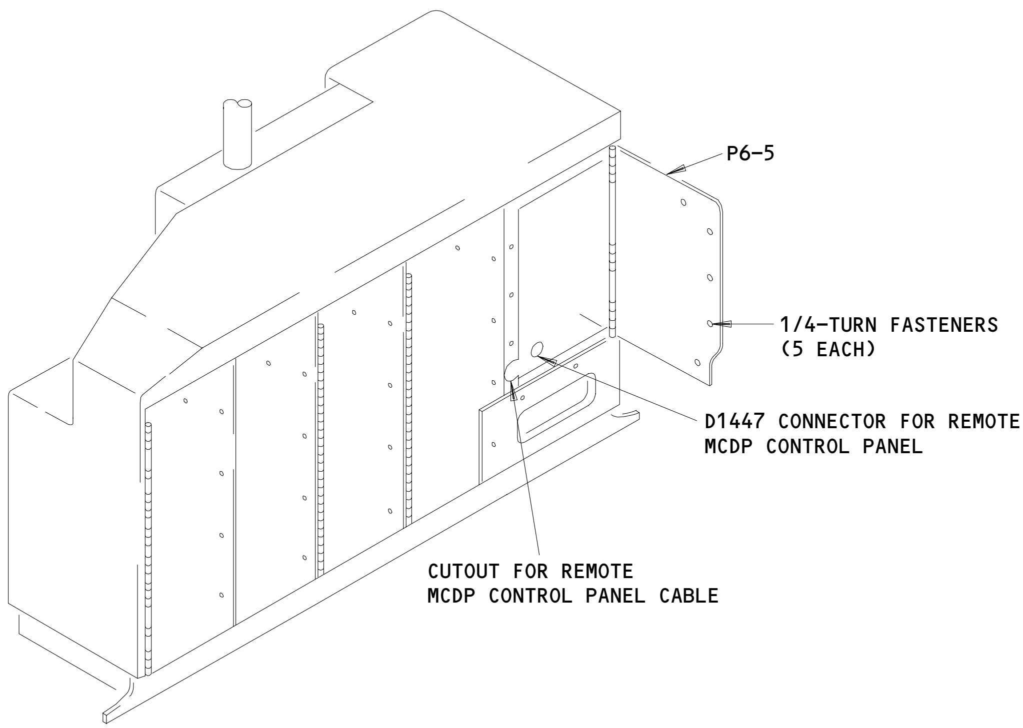
A



MAINTENANCE CONTROL DISPLAY PANEL, M168

B

Maintenance Monitor - Component Location
Figure 102 (Sheet 1)



MAIN POWER DISTRIBUTION PANEL P6

C

Maintenance Monitor - Component Location
Figure 102 (Sheet 2)

MAINTENANCE MONITOR - ADJUSTMENT/TEST

1. General
   A. The maintenance control display panel (MCDP) is installed on the E1-2 shelf in the main equipment center. This procedure gives the operational test of the MCDP. MM 22-00-02/201 gives the performance of ground tests, and selection of flight faults and their diagnostic codes.

TASK 22-41-00-715-001

2. Operational Test
   A. Equipment
      (1) Remote Control Unit, Maintenance Control Display Panel - A22001-22 (recommended), A22001-15 (optional)
   B. References
      (1) 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
      (2) 22-00-02/201, Autoflight BITE
      (3) 24-22-00/201, Electrical Power - Control
      (4) 27-61-00/201, Spoiler/Speedbrake Control System
      (5) 31-41-00/501, EICAS
   C. Access
      (1) Location Zones
          119/120   Main Equipment Center
          211/212   Flight Compartment

      (2) Access Panel
          119AL   Main Equipment Center

   D. Prepare to do a Test of the MCDP

      S 865-004
      (1) Supply electrical power (Ref 24-22-00).

      S 865-002
      (2) Make sure that these circuit breakers on the overhead circuit breaker panel, P11, are closed:
          (a) 11C30, LANDING GEAR POSITION AIR/GND SYS 1
          (b) 11E8, FMCS CDU LEFT
          (c) 11E9, FMCS CMPTR LEFT
          (d) 11E16, MODE CONT PNL L
          (e) 11E17, FLT CONT CMPTR PWR L
          (f) 11E18, FLT CONT CMPTR SERVO L
          (g) 11E20, FLT CONT CMPTR PWR CENTER
          (h) 11E21, FLT CONT CMPTR SERVO CENTER
          (i) 11E29, FMCS CDU RIGHT
          (j) 11E30, FMCS CMPTR RIGHT

(k) 11E34, MODE CONT PNL R
(l) 11E35, FLT CONT CMPTR PWR RIGHT
(m) 11E36, FLT CONT CMPTR SERVO RIGHT
(n) 11F14, TMC AC
(o) 11F15, TMC DC
(p) 11F16, TMC SERVO
(q) 11U9, MAINT CONT DSPL
(r) 11U15, LANDING GEAR AIR/GND SYS 1
(s) ALL MTH AIRPLANES;
SAS 150-999;
11U23, LANDING GEAR POSITION AIR/GND SYS 2
(t) SAS 050-149;
11U24, LANDING GEAR POSITION AIR/GND SYS 2

S 725-005
(3) Make sure that EICAS operates properly (Ref 31-41-00).

S 015-003
(4) Open the access panel, 119AL, for the MCDP (Ref 06-41-00).

E. Do a Test of the MCDP

S 715-006
(1) Do the initial power-up test without the remote MCDP control panel, as follows:
(a) Push and hold the POWER ON/OFF switch-light until the FAIL light in the POWER ON/OFF switch comes on.
1) Make sure that the FLT FAULTS MODE and GRD TEST MODE lights come on and then go off.
2) Make sure that all the display segments come on.

NOTE: The POWER ON/OFF switch can be pushed off and then on to make sure that the switch-lights and display segments come on.

3) Wait 5 seconds, and then make sure that these steps occur:
a) The FAIL light in the POWER ON/OFF switch goes off
b) The FLT FAULTS MODE light comes on
c) The MCDP display shows LAST FLT FAULT?.

S 715-007
(2) Do the initial power-up test with the remote MCDP control panel, as follows:
(a) Connect the remote MCDP control unit to the remote control connector, installed behind the P6-5 panel.
(b) Push the CONF/MCDP switch on the EICAS maintenance panel, on the right side panel P61.
1) Make sure the EICAS display shows the message MCDP OFF.
(c) Set and hold the ON/OFF switch up until the EICAS display shows IN PROGRESS.
1) Make sure that the EICAS display shows MCDP FLT FAULTS, after MCDP OFF is shown.

2) Make sure that the EICAS display shows LAST FLT FAULT? below MCDP FLT FAULTS.

S 715-008

(3) Do the Ground Tests Mode test that follows:
(a) Push the GRD TEST MODE switch-light.
1) Make sure that the GRD TEST MODE switch-light comes on.
2) Make sure that the MCDP display shows 01 FCC TEST?.
3) Make sure that the EICAS display shows MCDP GRD TEST, after MCDP FLT FAULTS is shown.
4) Make sure that the EICAS display shows 01 FCC TEST, when the remote MCDP control unit is used.
(b) Push the GRD TEST SEL UP or DOWN or NO/SKIP switch.

NOTE: See the front of the MCDP for a list of available tests. The NO/SKIP switch is used to advance past a test that is displayed but not engaged or to exit a test that has been engaged. The UP and DOWN switches display the test numbers at four per second.

1) Make sure that the test numbers increase when the GRD TEST SEL UP or the NO/SKIP switch is pushed.
2) Make sure that the test numbers decrease when the GRD TEST SEL DOWN switch is pushed.

S 715-033

(4) To do a specified ground test, refer to MM 22-00-02/201.

S 715-009

(5) Do the Flight Faults Mode test that follows:
(a) If the FLT FAULTS light is not on, and the MCDP display does not show LAST FLT FAULT?, push the FLT FAULTS switch.
1) Make sure that the MCDP display shows LAST FLT FAULT?.
(b) If you want the MCDP display to show the LAST FLIGHT FAULTS, push the YES/ADV switch.
1) Make sure that the MCDP display shows 01 "flight fault", or NO LAST FLT FAULTS.
(c) Push the YES/ADV switch to show each flight fault.
1) Make sure that the MCDP display shows a different 01 "flight fault" each time the YES/ADV switch is pushed, until ALL LAST FLT FAULT DSPLY is shown.
(d) Push the YES/ADV or NO/SKIP switch to exit the last flight faults (when NO LAST FLT FAULTS or ALL LAST FLT FAULT DSPLY is shown).
1) Make sure that the MCDP display shows PREV FLT FAULT?.
(e) If you want the MCDP display to show the PREVIOUS FLIGHT FAULTS, and the MCDP display does not show PREV FLT FAULT?, push the NO/SKIP switch.
1) Make sure that the MCDP display shows PREV FLT FAULTS?.

(f) Push the YES/ADV switch to show previous flight faults.
1) Make sure that the MCDP display shows XX "previous flight fault", or NO PREV FLT FAULT.

NOTE: XX indicates the number of flights ago that the flight fault occurred.

2) Make sure that the MCDP display shows a different XX "previous flight fault" each time the YES/ADV switch is pushed, until ALL PREV FLT FAULT DSPLY is shown.

(g) Push the YES/ADV or NO/SKIP switch to exit the previous flight faults (when NO PRV FLT FAULTS or ALL PREV FLT FAULT DSPLY is shown).

F. Flight Faults for Test Flights Only test:

S 435-036
(1) Connect the remote MCDP control unit to the remote control connector, installed at the P6-5 panel.

S 865-037
(2) Push the CONF/MCDP switch on the EICAS maintenance panel, on the right side panel P61.
(a) Make sure that the EICAS display shows the message MCDP OFF.

S 865-029

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURIES TO PERSONS OR DAMAGE TO EQUIPMENT.

(3) Do the deactivation procedure for the spoilers (Ref 27-61-00/201) or move all persons and equipment away from the spoilers.

S 865-030
(4) Open these circuit breakers and attach DO-NOT-CLOSE tags:
(a) On the overhead circuit breaker panel P11:
1) AIRPLANES WITH THE "LANDING GEAR POSITION AIR/GND SYS 2 ALTN" CIRCUIT BREAKER INSTALLED AT PANEL GRID LOCATION 11C29;
11C29, LANDING GEAR POSITION AIR/GND SYS 2 ALTN
2) 11C30, LANDING GEAR POSITION AIR/GND SYS 1

3) 767-300;
   11U23, POSITION AIR/GND SYS 2
4) 767-200;
   11U24, POSITION AIR/GND SYS 2

S 215-038

(5) Make sure that the EICAS display does not show TMC configuration data or the message MAINT DATA DISPLAY.

S 865-039

(6) Set the MAINT Enable Bypass Switch, installed on the E8 EICAS rack in the main equipment center, to BYPASS.

S 735-040

(7) Push the CONF/MCDP switch on the EICAS maintenance panel (P6).
   (a) Make sure that the lower EICAS display shows configuration data and the message MCDP OFF.

S 735-041

(8) Set and hold the ON/OFF switch up, on the remote MCDP control unit, until the EICAS display shows IN PROGRESS under MAINT DATA DISPLAY.
   (a) Make sure that the EICAS display shows MCDP FLT FAULTS, after MCDP OFF is shown.
   (b) Make sure that the EICAS display shows PRESENT FLT FAULTS? below MCDP FLT FAULTS.

S 735-042

(9) Set the GND TEST/FLT FAULTS switch to GND TEST on the remote MCDP control unit.
   (a) Make sure that the EICAS display shows MCDP FLT FAULTS.

S 865-043

(10) Set the MAINT Enable Bypass Switch to NORMAL.
   (a) Make sure that the EICAS display does not show TMC configuration data or the message MAINT DATA DISPLAY.

S 865-031

WARNING: DO THE DEACTIVATION PROCEDURE FOR THE SPOILERS OR MOVE ALL PERSONS AND EQUIPMENT AWAY FROM THE SPOILERS. THE SPOILERS CAN RETRACT QUICKLY AND CAUSE INJURIES TO PERSONS OR DAMAGE TO EQUIPMENT.

(11) Do the deactivation procedure for the spoilers (Ref 27-61-00/201) or move all persons and equipment away from the spoilers.

S 865-034
(12) Remove the DO-NOT-CLOSE tags and close these circuit breakers on the P11 panel:
(a) On the overhead circuit breaker panel P11:
1) AIRPLANES WITH THE "LANDING GEAR POSITION AIR/GND SYS 2 ALTN" CIRCUIT BREAKER INSTALLED AT PANEL GRID LOCATION 11C29;
11C29, LANDING GEAR POSITION AIR/GND SYS 2 ALTN
2) 11C30, LANDING GEAR POSITION AIR/GND SYS 1
3) 767-300;
11U23, POSITION AIR/GND SYS 2
4) 767-200;
11U24, POSITION AIR/GND SYS 2

S 735-044
(13) Push the CONF/MCDP switch on the EICAS maintenance panel.
(a) Make sure that the EICAS display shows MCDP DATA DSPLY AUTO ON MODE, IN PROGRESS.

S 865-045
(14) Open and then close this circuit breaker on the P11 panel:
(a) 11U9, MAINT CONT DSPL

S 215-046
(15) Make sure that the EICAS display shows the message MCDP OFF.

G. Put the Airplane Back to Its Initial Condition

S 865-013
(1) Push the POWER ON/OFF switch to OFF if the MCDP is not necessary.

S 085-021
(2) Remove the remote MCDP control unit if it was installed.

S 865-032
(3) Do the activation procedure for the spoilers if you did the deactivation procedure (MM 27-61-00/201).

S 865-028
(4) Remove electrical power if it is not necessary (Ref 24-22-00).

MAINTENANCE CONTROL DISPLAY PANEL - REMOVAL/INSTALLATION

1. General
   A. The maintenance control display panel (MCDP) is installed on the E1-2 shelf in the main equipment center.

TASK 22-41-01-004-001

2. Remove the Maintenance Control Display Panel
   A. References
      (1) 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels
      (2) 20-10-01/401, E/E Rack-Mounted Components
      (3) 20-41-01/201, Electrostatic Discharge Sensitive Devices
   B. Access
      (1) Location Zones
         119/120   Main Equipment Center
         211/212   Flight Compartment

      (2) Access Panel
         119AL   Main Equipment Center

   C. Prepare to Remove the MCDP

      S 864-002
      (1) Open this circuit breaker on the overhead circuit breaker panel, P11, and attach a DO-NOT-CLOSE tag:
         (a) 11U9, MAINT CONT DSPL

      S 014-003
      (2) Open the access panel, 119AL, for the MCDP (Ref 06-41-00).
   D. Remove the MCDP

      S 914-004

      CAUTION: DO NOT TOUCH THE MCDP BEFORE YOU DO THE PROCEDURE FOR DEVICES THAT ARE SENSITIVE TO ELECTROSTATIC DISCHARGE. ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE MCDP.

      (1) Do the procedure for devices that are sensitive to electrostatic discharge (Ref 20-41-01).

      S 024-005
      (2) Remove the MCDP (Ref 20-10-01).

TASK 22-41-01-404-006

3. Install the Maintenance Control Display Panel
   A. References
      (1) 06-41-00/201, Fuselage (Major Zones 100 and 200) Access Doors and Panels

(2) 20-10-01/401, E/E Rack-Mounted Components
(3) 20-41-01/201, Electrostatic Discharge Sensitive Devices
(4) 24-22-00/201, Electrical Power - Control

B. Access
(1) Location Zones

119/120	Main Equipment Center
211/212	Flight Compartment

(2) Access Panel

119AL	Main Equipment Center

C. Install the MCDP

S 914-007

CAUTION: DO NOT TOUCH THE MCDP BEFORE YOU DO THE PROCEDURE FOR DEVICES THAT ARE SENSITIVE TO ELECTROSTATIC DISCHARGE. ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE MCDP.

(1) Do the procedure for devices that are sensitive to electrostatic discharge (Ref 20-41-01).

S 424-008
(2) Install the MCDP (Ref 20-10-01).

NOTE: Do not move the MCDP directly from a different airplane. The MCDP non-volatile memory (EAROM) contains flight data that is correct only for the initial airplane installation. You must use shop procedures to erase this data unless the MCDP will be installed on the same airplane.

D. Do a Test of the MCDP

S 864-009
(1) Supply electrical power (Ref 24-22-00).

S 864-010
(2) Remove the DO-NOT-CLOSE tag and close this circuit breaker on the P11 panel:
(a) 11U9, MAINT CONT DSPL

S 714-025

(3) Do the MCDP Test that follows:

(a) Push the POWER ON/OFF switch-light. This will start a self-test.

(b) Make sure that all of the display segments and all switch-lights come on during the test.

(c) Make sure that the FAIL light goes off after five seconds.

(d) Make sure that the FLT FAULTS switch-light comes on.

(e) Make sure that the MCDP display shows LAST FLT FAULTS?.

(f) Push the YES/ADV switch-light.

(g) Make sure that the MCDP display shows NO LAST FLT FAULTS.

(h) Push the YES/ADV switch-light.

(i) Make sure that the MCDP display shows PREV FLT FAULTS?.

(j) Push the YES/ADV switch-light.

(k) Make sure that the MCDP display shows NO PREV FLT FAULTS.

(l) Push the POWER ON/OFF switch-light if the MCDP is not necessary.

E. Put the Airplane Back to Its Initial Condition

S 414-023

(1) Close the access panel, 119AL (Ref 06-41-00).

S 864-024

(2) Remove electrical power if it is not necessary (Ref 24-22-00).