65SD4514, NASA-CR-71497, JPL Contract 951112, Jan. 1966, General Electric.

[2] "Central Computer," Vol. IV, Final Report, Voyager Spacecraft Phase B, Task D, Report 67SD4379, NASA-CR-93555, George C. Marshall Space Flight Center Contract NAS 8-22603, Oct. 1967, General Electric.

[3] Larsen, D. P. B. and Skinner, K. L., "Space Packaged Automatic Checkout System," *Transactions of the IEEE*, Vol. AES-2, No. 4, Supplement, July 1966, pp. 777–782.

[4] Massey, H. N. and Smith, W. E., "Summary Report, Implementation Investigations of ZVA/FVA Compression Algorithms," Contract NAS 8-5110, Accession N66-31368, Feb. 1966, Lockheed Missiles & Space Co., Sunnyvale, Calif.

[5] Huffman, G., "Error Control Coding for Interplanetary Spacecraft," Rept. 67SD230, March 1967, General Electric.

[6] Seaman, L. T. and Brown, H. E., "Planetary Approach Guidance for a Martian Mission," presented at AAS Symposium, June 1967, Huntsville, Ala.

[7] Liviakis, G. and Firstman, S. I., "A Survey of Spaceborne Computers," Technical Advisement Memorandum 171-1, PRC D-1237, NASA Accession N66-24705, Planning Research Corp., Los Angeles, Calif.

[8] "Indication of Relative Merit," Voyager Task D Study Milestone Rept. VOY-D3-TM-22, Sept. 1967, General Electric.

[9] "Preliminary Indication of Relative Merit," Voyager Task D Study Milestone Rept. VOY-D3-TM-14, Aug. 1967, General Electric.

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Apollo Guidance and Control System Flight Experience

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The Apollo guidance, navigation, and control system is a complete, integrated, flight management system with a central general-purpose digital processor, multiple sensor information, astronaut command interface and space-to-ground command and data links. The flight experience provides data for an identification of the elements of system design, prelaunch and flight activities that were most influential in achieving success. The prelaunch and flight activities and data reviewed include four unmanned Apollo launches (three command modules and one lunar module) and three manned missions. Comparisons are made between ground measured data and measurements made during missions. The calculated system performance for some guidance phases of the mission has been based upon ground measurements and compared to actual in-flight performances and to system-specified performance. Among the significant factors enabling the system to perform its function successfully were the early recognition of necessary design changes for stable performance, the ability to predict the expected system performance, the discipline imposed by the policy of allowing no unexplained failures, and the ability to diagnose flight operational anomalies.

Introduction

THE Apollo Guidance, Navigation, and Control (GN&C) system[1–7] (Fig. 1), must guide, navigate, and control the spacecraft—command module (CM) and lunar module (LM)—through all phases of the lunar landing mission. It is designed to have a completely self-contained capability. A central element is a general purpose digital computer that contains both flight operational programs and ground checkout programs. The astronaut interface is via the display and keyboard (DSKY). The primary sensors are the inertial measurement unit (IMU) for reference coordinate memory and measurement of the specific force, and the optical subsystem (OSS) for navigation and for reference coordinate alignment of the IMU. In addition, there are radar range measurements for landing, range and line-of-sight direction for rendezvous, hand-controller input commands for manual steering and attitude control, and VHF ranging for rendezvous.

The system design was begun at MIT in October 1961, and the GN&C installation in the first flight spacecraft was completed in September 1965. The first flight program release (Corona) occurred in January 1966, and the first flight was launched on 25 August 1966. During this rather brief period of time, concepts of the lunar-landing-mission operations were changing, and GN&C system requirements were added, subtracted, and modified. The system was designed to be fully integrated with the astronaut as well as to have an automatic capability. The first four flights were unmanned and required the automatic system. The original design intent was to have a completely self-contained navigation system. During the program it was directed that primary navigation would be by the ground-based tracking network. Both means of navigation are accommodated as ground-transmitted spacecraft state vectors.

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Prelaunch Operation

An Apollo GN&C system on the launch pad at Kennedy Space Center (KSC), has had approximately 12 months of system testing. After a final verification of flight readiness, the countdown operations begin. The average lunar module

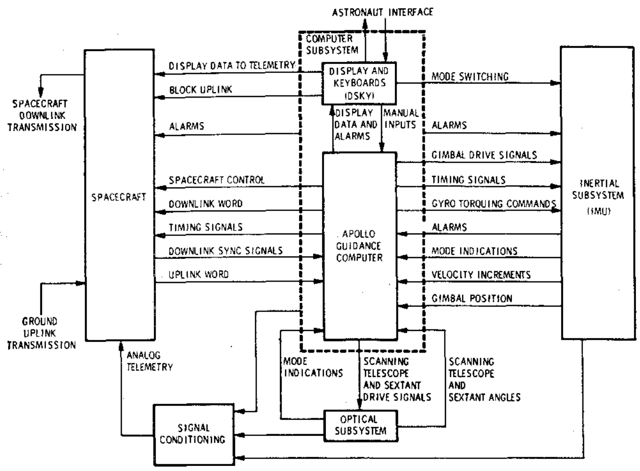

Fig. 1 Apollo Guidance Navigation and Control System block diagram.

GN&C system will have been checked out several weeks before the scheduled flight. The computer erasable memory is loaded for flight and the system turned off except for IMU temperature control. The system is not activated again until it is in space. The average CM GN&C system is operated fifty hours in support of the countdown. The system is exercised through automatic operational checks and a final calibration test. The initial conditons for the mission are loaded into the computer erasable memory and the IMU commanded to start the automatic platform alignment by gyrocompassing. About two weeks prior to launch the alignment of the IMU is verified by the astronaut using the optical system space sextant to sight on illuminated targets two miles from the launch vehicle. The launch vehicle has been demonstrated to be stable enough so that optical verification is now not required in the final countdown.

In the control room, which is 12 miles from the launch site, the serial digital data from the spacecraft are processed by the ACE (acceptance checkout equipment) computers which in turn display this information to the test engineers as meter and oscillograph readings, event lights or CRT (cathode ray tube) displays. In addition to standard data the telemetry transmitted from the flight computer to the ground is processed to produce a CRT display analogous to the onboard DSKY display the astronauts are monitoring. The K-START (Keyboard Sequence to Activate Random Testing) command system duplicates the keyboard section of the onboard computer DSKY. The keyboard entry is paralleled with a tape reader allowing for automatic, rapid, error-free command sequences from the control room to the onboard computer. The capability for monitoring and commanding the GN&C system remotely is exploited in the design of the prelaunch test procedures to enable parallel testing of spacecraft subsystems.

The guidance computer is programed to compensate the system for the predominant instrument errors. The objective of the prelaunch calibration testing is to provide best estimates of the present values of the error coefficients for use as compensation and to provide data for determining the uncertainties to be expected. The known amplitude of gravity is used to calibrate the accelerometers. The gyro drift calibration is based on the detection of the vector rotation of gravity by the accelerometers. The drift information must be separated from accelerations caused by launch-vehicle acceleration due to sway and from noise due to quantization in the pulsed integrating pendulous accelerometer (PIPA). The velocity quantum size for the CM is 5.85 cm/sec and for the LM is 1 cm/sec. The information is separated from the noise by a simplified optimum linear filter which includes in its state vector estimates of launch vehicle disturbances.[3]

The measurements made on the launch pad are usually used as reconfirmations of the selected compensation values. The compensation parameters are accelerometer bias and scale-factor errors for the three accelerometers, and gyro bias drift and two acceleration-sensitive drift terms for the three gyros, for a total of fifteen terms.

Prelaunch and Flight Error Analysis

The prelaunch system performance data have specified tolerances. In the cases where the specified tolerances were exceeded, the flight worthiness of the system was evaluated on the basis of the probable mission effect of the deviating parameter. As an example, shifts of gyro drift parameters beyond specified limits during prelaunch tests occurred on Apollo 3, 4, 5, 6. In a case indicating possible catastrophic failure in flight,[4] the Inertial Measurements Unit was replaced (Apollo 6). In the other cases, where the test data showed a performance degradation, determination of the mission effect was required. This determination required the development of error analyses that relate variations of each of the measurable parameters to the mission.

Each mission in the Apollo program is unique. The mission performance requirements were defined early in the Apollo program based upon a typical lunar landing. Because of the variety of missions and mission objectives, it is necessary to have a separate error analysis for each mission. For all missions except Apollo 5 the segmented mission phase approach to error analysis using a linearization technique is entirely adequate and was pursued. An error analysis is conducted using both the specification values, as well as the demonstrated values, for the GN&C system. Comparisons for selected mission phases are presented in Table 1, and one set is illustrated in Fig. 2.

The unmanned Apollo 5 flight was such that known initial conditions for each thrusting phase were not available. As the system guided the vehicle based upon its actual set of initial conditions, the guidance errors could not be treated with linearized perturbations. The resulting position and velocity errors became more nonlinear as the mission progressed. The mission was scheduled for nine Earth orbits, and the small-angle assumptions usually used with gyro drift were no longer applicable. The only solution was to conduct a large number of Monte Carlo error analyses of the complete mission.

Some interesting examples of how error analysis helped resolve operational problems occurred on the early flights. The flight plan for AS-202 called for a suborbital flight of approximately $\frac{3}{4}$ of an orbit with a maximum entry range coupled with a maximum heat-rate input to the heat shield. The original requirements called for an entry-angle uncer-

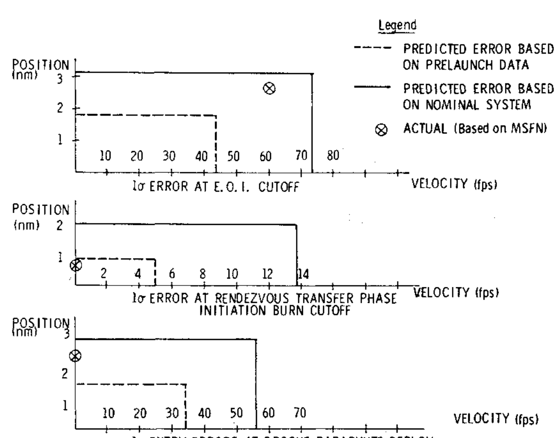

Fig. 2 Comparison of Apollo predicted and actual system errors.

Table 1 Guidance and navigation system performance

Mission and parameters[a]	1-σ Uncertainty based on Specifications	Preflight data	Best estimate error
Apollo 4 (SA501)			
P, re-entry start, naut miles[a]	2.75	3.15	7.5[b]
V, re-entry start, fps	26.6	51.5	140[b]
P, splash, naut miles	22.5	18.6	7.4[b]
Apollo 5 (LM1)			
Altitude at perigee after APS cutoff, naut miles	16.59	17.95	U[a]
P, SIVB cutoff, naut miles	5.6	4.22	0
V, SIVB cutoff, fps	132.5	100	2
Apollo 6 (AS502)			
P, re-entry start, naut miles	2.8	2.75	2.7[c]
V, re-entry start, fps	58	57	10.2[c]
P, re-entry end, naut miles	14.2	7.2	NA[d]
Apollo 7 (AS205)			
P, EOI cutoff, naut miles	3.1	1.8	2.6
V, EOI cutoff, fps	73	43	60
P, rendezvous TPI burn, naut miles	1.95	0.7	0.51
V, rendezvous TPI burn, fps	13.7	5	U
P, drogue deploy, naut miles	2.8	1.4	2.2
V, drogue deploy, fps	56	33.7	U
Apollo 8 (AS503)			
P, EOI cutoff, naut miles	4.3	3.9	0.016
V, EOI cutoff, fps	70.7	66	1
P, TLI cutoff, naut miles	1.25	1.1	1.9
V, TLI cutoff, fps	12.2	10	18
Perilune after LOI (3σ), naut miles	0.31	0.23	0.15
Apolune after LOI (3σ), naut miles	4.7	2.2	1.46
P, drogue deploy (CEP), naut miles	1.92	0.96	0.815

[a] P = position; V = velocity; U = unavailable.
[b] NASA-5-68-454.
[c] MSC-PA-R-68-9.
[d] Because of failure of the SIVB to reignite, the re-entry trajectory was not as planned; therefore, the entry error is not applicable.

tainty specification of $\frac{1}{2}°$. This was an easy achievement with the ground giving a state-vector update. During the checkout phases of the vehicle it was learned there were phases in the mission program when an update should not be sent because of onboard software deficiencies. This resulted in a condition where a back-up system would be required for guidance. As checkout proceeded it was clear that inertial performance could, with a 3σ uncertainty, not exceed $\frac{1}{3}°$. However, near the flight readiness test the performance requirement was voiced to be 0.05° 3σ uncertainty. The system would not make it without update and might not with update. However, near launch the requirement of $\frac{1}{2}°$ was reimposed and no update was attempted. Post-flight analysis showed the entry angle error to be 0.12.°

Another operational consideration where the error analysis was used concerned notification to the GN&C system that launch vehicle lift-off had occurred. This discrete command to be given to the spacecraft guidance computer was to change the mode of operation from gyrocompass to boost monitor. Three methods were used to achieve this: 1) Approximately 5 sec before lift-off, a discrete command was given called guidance reference release (GRR). 2) At lift-off, the same hard-wire discrete that went to the launch-vehicle guidance system was also sent to the Apollo GN&C system when the vehicle actually lifted off. 3) A backup lift-off command could be sent to the computer either by the astronaut or by an uplink command from the mission control center, Houston. At $T - 15$ sec the Saturn vehicle countdown proceeds automatically, monitored and progressed by a digital computer. Holds had occurred after $T - 5$ sec and it was common practice to recycle back to $T - 15$ min, thus creating a possible problem. Should a sequence like this occur, the guidance system would be released and proceed to monitor the boost. Should recycle occur, there would be insufficient settling time to re-establish orientation of the GN&C system by gyrocompassing. The error analysis results indicated that the GN&C system would navigate and monitor boost properly even if it were released well ahead of lift-off. Due to program considerations, it was decided to remove the GRR signal and to launch with only two methods of indicating lift-off.

Checkout History and Experience

Table 2 summarizes the history of systems to date. The average number of operating hours accumulated in checkout is 2460 hr during an average 10.45-month spacecraft testing period. The success in meeting schedules and establishing the flight worthiness of all the hardware was due to early recognition of the importance of considering checkout problems in the design, to minimization of equipment removals by carefully reviewing all anomalies for their flight impact, and to the discipline imposed by allowing no unexplained failures.

Early spacecraft testing revealed that there was a high probability of applying and/or removing spacecraft power to the GN&C system in an incorrect sequence. The first system design did not incorporate protective features for making the system tolerant of incorrect power sequencing. During checkout several instances occurred where, due to faulty procedures, power to the system was inadvertently applied or removed in an incorrect sequence. This resulted in performance shifts. The design was changed to provide internal protection to incorrect power sequencing. That design change saved many hours of retest and stabilized the performance data obtained in spacecraft testing.

Another example involves ground potential changes in docked test configuration. The possibility of reverse potential on the system was not considered in the initial design. When spacecraft tests indicated that reverse voltages could exist due to grounding configurations, the

Table 2 Guidance and navigation hardware installation history

System	Spacecraft contractor's plant: Installation completed	Spacecraft contractor's plant: System replaced	Spacecraft contractor's plant: Shipped to KSC	Kennedy space center: System replaced	Kennedy space center: Launch date	G&N system: Months in spacecraft	G&N system: Operation hr
Apollo 3, AS202, G&N 17	1/6/66	None	4/16/66	None	8/25/66	8.7	2192
Apollo 4, AS501, G&N122	8/29/66	None	12/22/66	None	11/9/67	14.3	2907
Apollo 5, LM1, G&N603	11/12/66	IMU 12/66	6/23/67	Computer 6/67 IMU 7/67	1/22/68	14.3	2626
Apollo 6, AS502, G&N123	1/3/67	IMU 6/67	11/23/67	IMU	4/4/68	8.6	2669
Apollo 7, AS205, G&N204	12/16/67	None	5/30/68	None	10/11/68	10.0	2345
Apollo 8, AS503, G&N208	4/1/68	None	8/12/68	None	12/21/68	8.6	1905
Apollo 9, AS534, CM104, G&N209	5/2/68	DSKY	10/5/68	IMU		10.0	2233
LM3, G&N605	10/7/67	None	8/14/68	IMU twice	3/3/69	17.0	1703

GN&C system electronics design was changed to tolerate reverse voltages.

When the prelaunch checkout shows any discrepancy, positive action is taken to eliminate possibility of failure in flight. An example of this was the failure of the GN&C system to accept an entry mode change command once during checkout of the AS-202 system. Even though the problem was never duplicated, the relays that could have caused this single malfunction were replaced. Another example involves the computer in the same mission. While one of the computers was undergoing inspection at the factory, it was discovered that one of the vibration isolation pads was missing from the oscillator module. Subsequent examination of other available modules revealed that, on the basis of the sample examined, there was about a 20% chance that one of the vibration isolation pads was missing in the computer in the spacecraft. The decision taken 30 days prior to flight was to remove the computer and inspect. It was rapidly done and verified that the pad had been installed.

The early GN&C system operations were plagued by the occurrence of unexplained restart.[6]‡ Noise susceptibility in test connectors was discovered and corrected by a shorting plug. Software errors were discovered and corrected by new software. Procedural errors were discovered by means of ACE playbacks and laboratory verification. The solution involved hardware changes, software changes, procedural changes and, above all, education and understanding on the part of all GN&C system operation personnel. The successful operation of the hardware during the Apollo flights was due primarily to this careful disciplined engineering that examines all facets of the situation and leaves no area uncorrected.

Flight Operations

During a mission the GN&C operation is monitored by computers in the real time control center (RTCC) in Houston. The digital data generated by the onboard computer consist of lists of two-hundred 14-bit computer words transmitted once every 2 sec. The contents of the lists are designed to provide information relevant to the mission activity. The data are used to drive displays on the guidance officer's console and numerous other support consoles. The amount of data from the guidance computer is limited by the word size and transmission rate. The design of the program selects the quantities to be transmitted and is used to make up for this deficiency. The data used for the real-time displays are selected prior to the mission, based on the flight controller's experience and operational requirements. In real time the data format is quite inflexible.

The control of the system is accomplished in the same computer complex. The data transmission parallels the onboard keyboard-entry capability. The data transmitted consist for the most part of updates of the spacecraft position and velocity which are determined by ground tracking stations and converted into the proper format by the Houston RTCC. The controller has the capability of commanding the spacecraft computer through an analogous keyboard with the same codes as the astronauts.

Review of the data obtained from the flight monitoring indicates that the ground calibration enables accurate error compensation. Review of the anomalies in flight operations indicates that there is a reasonable amount of time available during the mission for troubleshooting and diagnosis of problems. The only cases that could not be diagnosed in real time involved inadequate real-time data.

Guidance System Monitoring During a Mission

The monitoring of the guidance-system performance during the mission consists of comparing navigation data from other sources (ground tracking, Saturn V guidance, LM backup for CM, CM backup for LM), computing accelerometer output with no input at zero gravity, and determination of the quality of the inertial reference by successive inflight optical realignments of the IMU. These successive realignments are performed several hours apart so that the rotations of the IMU stable member required to realign it are mostly due to gyro drift with the fixed errors reduced inversely proportional to this time interval. There are also operational techniques utilizing star and planet horizons for checking the commanded attitude prior to a velocity-change maneuver.

The onboard measurement of the available IMU performance parameters can be used to further improve the performance. The compensation parameters can be modified through the keyboard, either onboard or from the guidance officer's console in Houston.

The guidance-system monitoring is designed to provide the flight controllers with data upon which a prediction of the future operation of the system is made. The flight controllers have preprogrammed decision points enabling the continuation of the mission with a backup system in control, or with a new mission plan, if their data indicate that the primary system may not perform adequately during the next critical mission phase.

The data telemetry from the spacecraft and the ability to predict future operation are limited. The limits set for the various parameters are selected on the basis of the worst performance experienced during design evaluation tests and prelaunch tests, excluding catastrophic failures.

The onboard measurements to date have indicated that excellent performance should be predicted and excellent performance has followed. The only onboard measurement available for the unmanned missions (Apollo 4, 5, 6) is ac-

‡ A restart is an internal protective mechanism that enables the computer to recover from random program errors, operator errors, and from environmental disturbances. Restart attempts to prevent the loss of any operating functions.

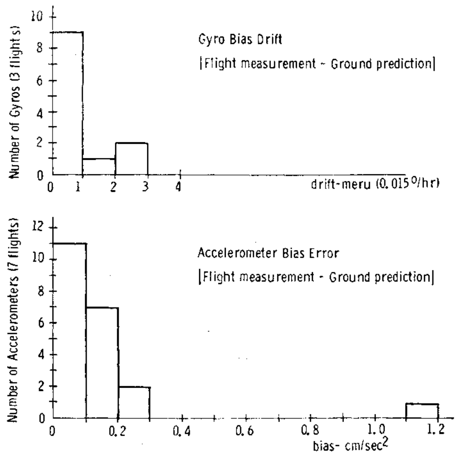

Fig. 3 Gyre Bias Drift (NBD) and accelerometer bias flight data.

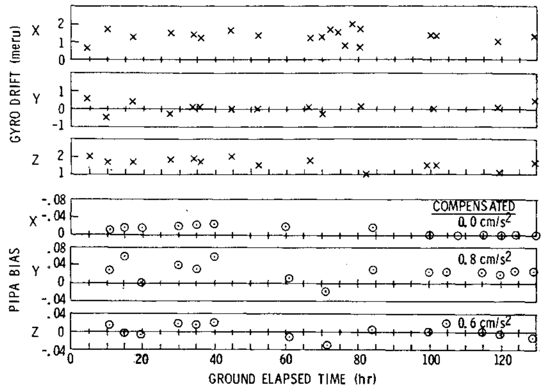

Fig. 4 Gyro drift and PIPA bias.

celerometer output at zero gravity (a_b). The manned missions also include inertial platform drift at zero gravity (NBD).

The inertial component data are presented in Table 3 and Fig. 3.

Apollo 8 afforded an unique opportunity for monitoring the IMU over a long period of continuous operation. The data indicate that stability of inertial operation has been achieved in the design. The entire data history is plotted in Fig. 4.

Diagnosis of Problems During the Mission

The adequacy of all subsystems to continue into the next phase and to complete the mission is reviewed continuously by the flight controllers. It is, therefore, important to diagnose problems in real time in support of the GO/NO-GO decisions. The flight experience shows that there is adequate time available for problem diagnosis and that there is a capability for real-time troubleshooting. There are two types of problems where real-time troubleshooting is of no value: those involving actual hardware failures and those involving incompatibilities due to inaccurate models of the spacecraft being used in the control programs. Examples of problems involving the GN&C system that have been explained in real time illustrate the capability that does exist.

During the APOLLO 4 (AS501) mission it was reported that a large difference existed between the γ indicated by the onboard computer and the γ as compared from radar tracking data, where γ is the angle between the position vector and the velocity vector. Real-time measurement of accelerometers indicated the GN&C system was operating properly. The difference was found to be a ground computation error. The guidance system was allowed to continue in control of the mission.

During the APOLLO 6 (AS502) mission a divergence was observed between the attitude information supplied by the GN&C inertial reference and the backup body-mounted attitude gyros. The divergence was first attributed to GN&C malfunction. Real-time review of prelaunch data

Table 3 Inertial component performance

	Accelerometer bias, cm/sec[c]			Gyro bias drift less compressation, meru[a]		
	Ab_x	Ab_y	Ab_z	NBDX	NBDY	NBDZ
Apollo 4						
In-flight measurement	0.304	0.23	−0.39			
Compensation	0.41	0.21	−0.28			
Apollo 5						
In-flight measurement	0.1	−0.35	0			
Compensation	0.14	−0.22	0.12			
Apollo 6						
In-flight measurement	−0.83	2.77	1.93			
Compensation	0.64	2.9	2.1			
Apollo 7						
Last prelaunch measurement	0.2	0.24	0.16	1.9	0.4	−0.8
a) In-flight, following boost	0.275	0	0.215	2.2	0.2	0.15
b) In-flight, at 145 hr[b]	0.309	0	0.208	1.4	−0.63	0
Apollo 8[c]						
Expected value from last ground measurement	0	0.845	0.615	0.93	2.2	1.3
a) In-flight, following boost	0	0.83	0.62	1.5	0.62	1.8
b) In-flight, during translunar coast	0	0.83	0.605	1.51	−0.13	1.84
c) In-flight, in lunar orbit	0	0.83	0.60	1.6	0.03	1.97
d) In-flight, during trans-Earth coast	0	0.82	0.59	1.38	0.16	1.6
Apollo 9						
Expected value based on ground measurements	0.38	−0.004	0.002	−1.6	−0.4	2.7
a) LM system after turn-on in orbit	0.32	0.013	−0.008	−3.6	−0.1	3.3
b) CM system after turn-on in orbit	−0.53	−0.34	0.38	−2.3	−0.5	−1.6
Expected value based on ground measurements[d]	0.64*	−0.10*	0.36	−1.2	−0.2	−2.4

[a] One meru is 0.015 deg/hr.
[b] The crew removed power from the guidance system during inactive periods.
[c] The Apollo 8 mission was flown with the guidance system continuously operating. The monitoring of the inertial reference and accelerometer errors provides a large set of data on Apollo inertial system performance in space environment.
[d] The compensation value was changed in orbit.

for the backup system indicated that the drift rates measured on the ground accounted for the divergence. The GN&C system remained in primary control for a successful mission.

During the APOLLO 7 mission a procedure for using the landmark-tracking navigation program for navigation sightings on the horizon was determined. The procedure did not work in the spacecraft. The computer was programed with the reasonable assumption that landmarks would be on the surface of the earth. The attempt to use the program for horizon sightings above the Earth's surface rather than the landmarks resulted in the attempt to compute the square root of a negative number. This resulted in a restart. The error in the procedure was quickly determined by ground tests.

A computer restart also occurred when the astronaut did not select a star when the computer requested a star selection for navigation. The computer interpreted the selection of "no" star as star number 0; the navigation star catalogue, however, started with star number 1. The result was a computer restart due to accessing a memory location address which did not "exist." The restart was diagnosed from real-time displays.

Another "failure" was associated with the "mark" button. The computer assimilates line-of-sight data from the optics only upon astronaut command, which consists of an interrupt caused by depressing the mark button on the navigator's control panel. The line-of-sight information is used for rendezvous navigation as well as for inertial-platform realignment. To protect the rendezvous navigation information from being modified by platform alignment sighting data, the computer programers prevent the processing of alignment marks during rendezvous navigation. The problem occurred when the astronaut terminated the rendezvous navigation program in a fashion not expected by the programers. This termination left the computer with the information that no alignment marks were to be processed. The next attempt at re-alignment failed due to an apparent failure in the mark interface to the computer. Ground troubleshooting uncovered the cause and a reselection and proper termination of the navigation program eliminated the problem.

Another problem involved the PIPAs, which use a pendulous mass as a torque-summing element. The accelerometer bias (output with no input) is due to the residual torques in the instrument. During a mission, at zero gravity, the accelerometer is calibrated by monitoring its output. During Apollo 7 the flight controllers noticed that the expected low output at zero gravity decreased to zero. This was interpreted as a possible hardware failure, and an in-flight test was conducted by making a maneuver to thrust along both directions of the accelerometer input axes. The results showed that the instrument was operating properly. The cause for the lack of any output was simply the PIPA reaching an operating region in free-fall where the torque generated by electronic nonlinearities was equal and opposite to the residual electromagnetic torques.

On the Apollo 8 mission, the prelaunch alignment program which is coded 01 was inadvertently selected by the astronaut during the trans-Earth coast. The commanding of the Apollo guidance computer consists mainly in selecting numerically-coded programs and loading the desired number at the time the computer requests the information. The loaded information is redisplayed for confirmation by the astronaut prior to being acted upon by the computer. The astronaut confirms that he indeed wants the displayed program to be executed by depressing a key on the keyboard. The "problem" caused by selecting the prelaunch alignment program was mainly due to the fact that the erasable portion of the computer memory is time-shared. The effect on the contents of the erasable memory of starting program 01 at that time was unknown. The problem was quickly dealt with by the crew and the contents of the memory verified by the ground to be correct.

Other Operational Problems

The problems involving the GN&C system in the Apollo program have been minor. They do provide an object lesson of the types of problems to be expected in a large program with many opportunities for error in design and operation. The operational problems can be categorized to indicate where the operational system is most susceptible to error. The types of problems to date have been the following: 1) Ground flight control errors, 2) operator errors, 3) misinterpretation of design data, 4) misinterpretation of flight telemetry data, 5) new phenomena, and 6) hardware problems. Ground errors include only those which arose in real time. These type of errors can be dealt with by realtime troubleshooting. Some examples have been already described. The selected major real-time "problems" categorized previously as operator errors clearly reflect difficulties in the design of the interactive computer programs and their use under mission conditions. These types of problems also can be easily diagnosed and corrected.

When the spacecraft responds to commands differently than the computer program expects it to respond, a performance degradation can result from either a logic error or incorrect information in the computer. Both have occurred to date. For example, the otherwise-successful suborbital mission AS202 missed the target by 200 miles. The major cause was that the lift-to-drag ratio, L/D, was 0.25 instead of an expected 0.35, with the result that the vehicle had insufficient lift to attain the targeted range. In Apollo 5, the control program for guidance during LM descent propulsion system engine operation monitored the thrust build-up after the engine had been commanded to fire. If the thrust buildup did not occur, the program was designed to turn off the engine and generate an alarm. During the flight the engine thrust buildup for the first descent engine burn did not occur at the rate expected by the program and the computer turned off the engine. The program was designed so that appropriate realtime commands could have restarted the control program but, due to ground tracking considerations, the mission was flown with backup procedures.

The spacecraft telemetry data are processed by a computer complex at Houston to provide realtime displays for the flight controllers. The limitations of that system require that some data not be displayed. The display, therefore, does not given an exact picture of the spacecraft status. The prime example of how the selected displays can cause misinterpretation occurred on Apollo 8. The flight plan called for the power to the GN&C optical subsystem to be left ON throughout the mission. The telemetry for the state of that power was not selected for realtime display. The computer monitors the sextant articulating line-of-sight angles and this information is transmitted as part of the computer down-telemetry. Several times during the mission the computer data indicated that the "trunnion" angle, one of the two data-encoded optics-system angles, changes from the expected 0° to an unexpected 45°. This change was unexplainable from the available data. The system operation, however, indicated that by recycling normal optics-operating procedures the system was not affected. The decision to continue to the moon was based on that fact. Several failure models were invented during the mission to explain the problem. Later, during the astronaut debriefing, it became apparent that the problem was due to switching OFF the optics power. With power removed the change in angle was to be expected each time the power was reapplied. Search through the data which was not processed in real time confirmed that explanation.

In the new phenomena category, it was found that the debris generated by the spacecraft can appear in the optics as stars to make true star identification difficult. The Apollo missions, therefore, have made extensive use of the computer-inertial measurement unit combination to direct the optical line of sight to aid star identification. Another finding was that the size of the Apollo spacecraft resulted in considerable attitude changes in Earth orbit due to atmospheric drag at perigee. This could be costly in fuel for large space-stations.

Because of the careful ground tests and reviews of test results, there have been very few G&N-related hardware problems to date in the Apollo missions. Hardware problems occurring in flight result in use of backup systems. The major problem that involved the G&N was the Apollo 6 ground update problem. Since the unmanned Apollo missions were dependent on ground tracking navigation data to a much greater extent than the manned missions, several navigation updates were planned for Apollo 6. The navigation data or other remote commands to the computer are transmitted in a triple-redundant code, $K\bar{K}K$. The computer will not accept data that does not conform to this code. During the Apollo 6 mission several attempts to send navigation updates were rejected by the computer. The most likely cause for rejecting the data is electromagnetic interference. Review of the interface did indicate a possible problem due to the ground command lines left disconnected at launch and unterminated. These wires were the probable antenna for picking up the noise. The source of the interference was later determined to be an ion pump associated with the fuel cells. The ion pump in the Apollo 7 spacecraft generated the same problem during a ground-test in the altitude chamber. The Apollo 6 ion pump had not been ground-tested in the altitude chamber. Wiring changes were also made in subsequent spacecraft to eliminate the possible noise pick-up in the ground command lines.

Conclusions

1) The system-error model contained in the specification is a good representation of the actual system errors during a mission. There is excellent agreement between the ground and the free-fall initial parameter measurements. Care must be taken in the mission error analysis where the guidance system is in the steering loop to see that mission phases can be treated as separate phases. This can always be done with correct initial conditions.

2) Quality and reliability are built into the equipment. With a well planned and well designed prelaunch checkout, inflight hardware problems are minimized. There is a reasonable amount of time available for inflight problem diagnosis and there exists an ability for troubleshooting and diagnosis both in flight and on the ground.

3) Automatic prelaunch checkout of space guidance, navigation and control systems is mandatory if checkout costs are to be reduced.

4) Missions are designed after the hardware is built; therefore, the hardware must be flexible to accommodate different mission applications.

5) The complexity of the GN&C system, as well as of the total spacecraft, dictates that emphasis be placed on simulation for verification and training.

6) Discipline is necessary to understand, explain and, where required, fix all phenomena associated with checkout. Any problem found must be related (by the use of strict build control) to all possible systems, and the effects evaluated, based upon requirements. The concept of no unexplained failure is the foundation of a discipline that enabled success to be achieved in the Apollo program.

References

[1] Miller, J. E., ed., *Space Navigation, Guidance and Control*, AGARDOGRAPH 105, 1966, Technivision Limited, Maidenhead, England.

[2] Martin, F. H. and Battin, R. H., "Computer-Controlled Steering of the Apollo Spacecraft," *Journal of Spacecraft and Rockets*, Vol. 5, No. 4, April 1968, p. 400.

[3] Schmidt, G. T. and Brock, L. D., "General Questions on Kalman Filtering in Navigation Systems," *Theory and Applications of Kalman Filtering*, edited by C. T. Leondes, NATO-AGARD, Chap. 10, to be published.

[4] Feldman, J. and Miller, J. E., "Gyro Reliability in the Apollo Guidance, Navigation and Control Systems," *Journal of Spacecraft and Rockets*, Vol. 5, No. 6, June 1968, pp. 638–643.

[5] Trageser, M. and Hoag, D. G., "Apollo Spacecraft Guidance System," Rept. R-495, June 1965, Massachusetts Institute of Technology, Cambridge, Mass.

[6] Copps, E. M., Jr., "Recovery from Transient Failures of the Apollo Guidance Computer," Rept. E-2307, Aug. 1968, Massachusetts Institute of Technology, Cambridge, Mass.

[7] Hoag, D. G., "Apollo Navigation Guidance, and Control Systems," Rept. E-2411, April 1969, Massachusetts Institute of Technology, Cambridge, Mass.