MASSACHUSETTS INSTITUTE OF TECHNOLOGY

APOLLO

GUIDANCE, NAVIGATION AND CONTROL

Approved: Ralph R. Ragan Date: 29 Apr 69
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Presented at the National Space Meeting of the Institute of Navigation, April 22-24, 1969, Houston, Texas.

E-2411

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A PROGRESS REPORT

by

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APRIL 1969

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ACKNOWLEDGMENT

This report was prepared under DSR Project 55-23870, sponsored by the Manned Spacecraft Center of the National Aeronautics and Space Administration through Contract NAS 9-4065 with the Instrumentation Laboratory of Massachusetts Institute of Technology in Cambridge, Massachusetts.

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The status of certain aspects of the Apollo navigation, guidance, and control systems in the command module and lunar module are examined on the basis of experience with the first eight development flights. Covered in this paper are facets of the inertial, optical, and computer hardware operation. The application of these hardware subsystems to the digital autopilots, rendezvous navigation, midcourse navigation, and entry are examined. The systems are judged to be fully ready to help a crew of astronauts land on the moon.

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ABSTRACT

The status of certain aspects of the Apollo navigation, guidance, and control systems in the command module and lunar module are examined on the basis of experience with the first eight development flights. Covered in this paper are facets of the inertial, optical, and computer hardware operation. The application of these hardware subsystems to the digital autopilots, rendezvous navigation, midcourse navigation, and entry are examined. The systems are judged to be fully ready to help a crew of astronauts land on the moon.

INTRODUCTION

At the time this paper is written the Apollo program stands on the threshold of completing its stated goal: a manned landing on the moon. As the exciting conclusion of this program approaches, we examine here the status of the onboard navigation, guidance, and control systems of the Apollo Command Module (CM) and Lunar Module (LM). A detailed description of these systems is available elsewhere*. This paper will make a qualitative examination of the significant results of Apollo flight test experience as it substantiates the conclusion that these systems have been demonstrated to be ready to support the manned lunar landing.

NAVIGATION, GUIDANCE, AND CONTROL FUNCTIONS

The function of navigation, guidance, and control systems is to manage the various spacecraft motions involved in accomplishing the mission. Functions of concern include the onboard measurement of rotational and translational motion, the processing of these measurements for display to the crew and ground control, the acceptance from the crew or ground control of desired spacecraft maneuver instructions, and the execution of the defined maneuvers to change the spacecraft motion by modulating the firing of the various rocket propulsion systems provided for that purpose. As used here, the concepts of navigation, guidance, and control are defined as follows:

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*References will be cited in the body of the paper.

Navigation is the measurement and computation necessary (1) to determine the present spacecraft position and velocity; (2) to determine where the present motion is sending the spacecraft if no maneuver is made; and (3) to compute what maneuver is required to continue on to the next step in the mission.

Guidance is the continuous measurement and computation during accelerated flight to generate necessary steering signals so that the position and velocity change of the maneuver will be as that required by navigation computations.

Control is the management of the rotational aspects of spacecraft motion; the rotation to and the stable maintenance of the desired spacecraft attitude during free fall coasting flight and during powered accelerated flight. The function of control here is that of an autopilot which must accept steering direction from guidance or the astronaut and achieve the commanded vehicle attitude.

SYSTEM DESCRIPTION

The navigation, guidance, and control equipment in the command module and and lunar module is described in detail in references 1, 2, 3, and 4. The following general description will suffice here.

Command Module System

The navigation guidance and control system in the command module is shown schematically in Figure 1. The onboard digital computer plays a central role in the system operation. It receives and transmits data and commands appropriately from and to the other components and subsystems shown. Starting clockwise at the bottom of Figure 1, the computer receives data and instructions from the ground by radio telemetry and sends back to the ground formats of data of interest for mission control. The astronaut with his hand controllers can command the computer to execute rotational and translational commands. The inertial measurement unit, the IMU, provides a measure of spacecraft attitude with respect to an inertial frame defined by the alignment of the inner gyro stabilized member. In addition, accelerometers on this stabilized member measure the linear acceleration components being experienced by the spacecraft due to engine thrust and aerodynamic drag. Rigidly mounted to the base of the IMU is the articulating optical subsystem which the astronaut uses visually to measure direction to stars for IMU alignment and to measure the present direction from the spacecraft to navigation features of the earth and moon for determining spacecraft position and velocity.

During rendezvous exercises with the Lunar Module when the LM is returning to the orbiting command module from the lunar surface, the communication system between the spacecrafts measures the range of the LM from the CM to help in the CM backup of the rendezvous.

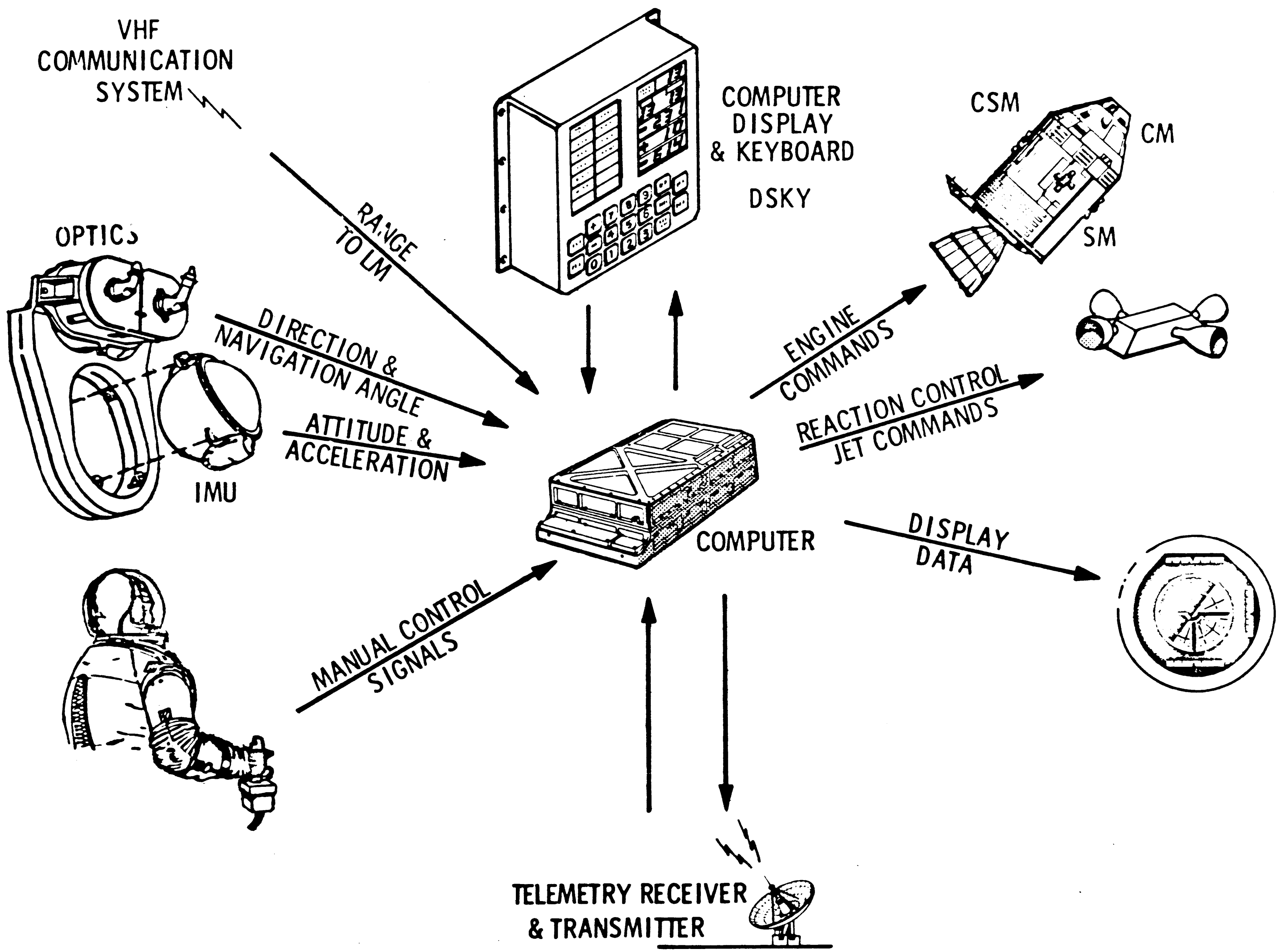

Figure 1 The Command Module GN & C System Schematic

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The computer display and keyboard, DSKY, is the primary communication interface between the astronaut and the computer. By use of the keyboard the astronaut can call up the programs, routines, and displays he desires and insert the data the computer requires. The numerical display presents data of interest to him so that he can monitor the progress and results of the computations and so that the computer can request data or instructions.

During powered maneuvers the computer sends the service propulsion engine on and off signals, and during the engine burning commands the angles of the engine gimbals. Each of the 16 reaction control jets on the service module (SM) and the 12 jets on the command module can be commanded on and off separately by signals from the computer so as to achieve spacecraft torque and translation force as required.

In the command module the only display from the system not on the DSKY is the total attitude and attitude error appearing on the main panel flight director's attitude indicator or "eight ball".

LUNAR MODULE SYSTEM

The system in the LM is shown in Figure 2. The IMU, computer, and DSKY in the LM are physically identical to those in the CM except for the accelerometer scaling in the IMU and the flight program in the computer. The LM optical system is a simple periscope to measure star direction for IMU alignment.

During the lunar landing, the landing radar measures local altitude above the lunar terrain, altitude rate, and components of horizontal velocity for the computer's use in navigating and guiding the landing maneuvers. On the lunar surface and during rendezvous in orbit, the rendezvous radar tracks the CSM orbiting above and provides the LM computer with direction, range, and range rate.

Besides commanding LM descent engine gimbal angle and engine thrust on and off signals, the computer must also command thrust level to this throtteable engine in accordance with the guidance laws being used in the landing. The ascent engine is not gimballed. Control torques during both the descent and ascent powered flight are provided by the 16 reaction control jets on the ascent stage in all three axes in addition to their use during non-powered flight for rotational and small translation maneuvers.

System driven displays in the LM include the DSKY and the attitude and attitude error display eight ball similar to that in the CM. In addition the LM computer drives computed altitude rate, and horizontal velocity component displays during the landing, and CM range and range rate displays during the ascent and rendezvous.

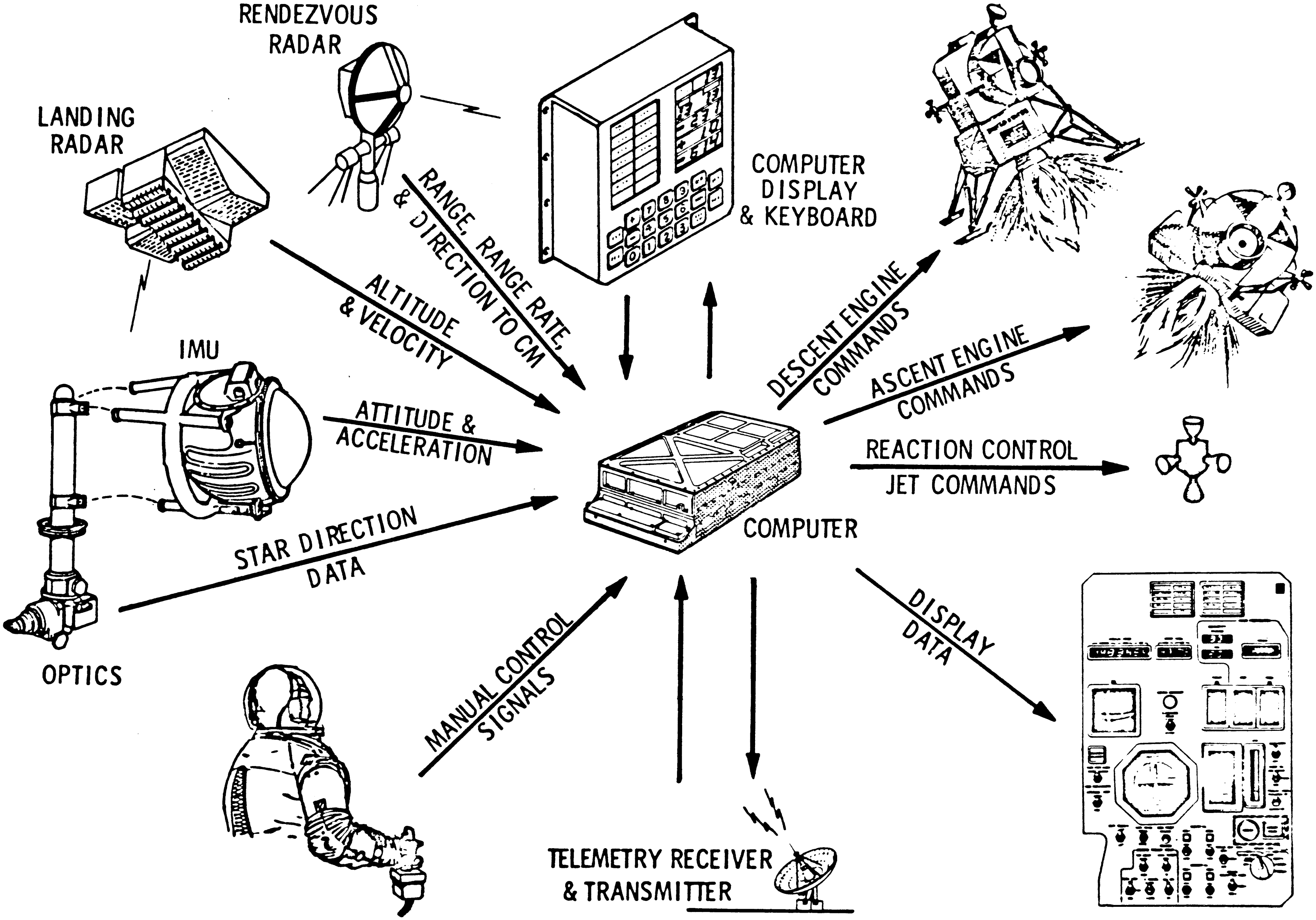

Figure 2 The Lunar Module GN & C System Schematic

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FLIGHT EXPERIENCE

Through March 1969, the navigation, guidance, and control systems have supported eight spacecraft flight tests: three unmanned CSM flights, one unmanned LM flight, three manned CSM flights, and one manned LM flight in missions Apollo 3 through Apollo 9. All functions designed into these systems to support the lunar landing have been checked successfully during 420 hours (17-1/2 days) of equipment operations in space flight including 22.5 hours of equipment flight operation in the LM. In this experience and the experience of many days of launch countdown activities there has been only one equipment failure. A tiny pin dropped out of a backup counter in the CM telescope during the Apollo 9 mission and made the motion sticky. The navigator freed up the stickiness with a hand tool and had no further problem. Since the counter information is displayed on the DSKY the loss of it did not interfere with mission objectives.

A summary of Apollo development flight experience and prognosis is shown in Table I.

THE INERTIAL MEASUREMENT UNIT

The inertial measurement unit or IMU is a three-degree-of-freedom gimballed platform isolating three single-degree-of-freedom gyros and three single-axis accelerometers from the spacecraft attitude. The orientation of the platform and the direction of the sensitive axes of the accelerometers are held inertially fixed by the gyro error signals feeding the platform drive servos. The orientation is held to that attitude determined by alignment to the stars. The IMU provides the computer with information of spacecraft attitude by readout of the IMU gimbal angles. During powered flight and earth atmospheric entry this attitude information is supplemented by indication from the accelerometers of the linear motion arising from the rocket propulsion or aerodynamic forces.

Gyro or Accelerometer Failure Prediction

The orientation stabilization and acceleration measurement of the Apollo IMU is perhaps more accurate than needed to support the Apollo mission. The use of high performance gyros and accelerometers does result in efficient maneuvers and savings in propellant, but the more significant purpose for high performance gyros and accelerometers is that they are more reliable, can degrade without severe mission penalty, and their degradation can be a forecast of outright failure before the system is committed to a mission[15].

During subsystem and spacecraft testing over many months prior to launch, the various parameters indicating the performance of the inertial units are measured and their signature of normal performance noted. Based on these data, the flight computer is loaded with compensation values for 15 coefficients describing the non-perfect behavior of the IMU. These coefficients are quite stable and repeatable in a good IMU.

Table I Primary Navigation, Guidance, & Control Systems (Apollo Development Flights)

	Launch Date	Launch Vehicle		Mission
Apollo 3 CSM	25 Aug 1966	AS-202	Unmanned	Ballistic Suborbital
Apollo 4 CSM	9 Nov 1967	AS-501	Unmanned	Orbital, High Apogee, Simulated Lunar Return
Apollo 5 LM	22 Jan 1968	AS-204	Unmanned	Lunar Module Earth Orbital
Apollo 6 CSM	4 Apr 1968	AS-502	Unmanned	Orbital, High Apogee Simulated Lunar Return
Apollo 7 CSM	11 Oct 1968	AS-205	Schirra, Eisele Cunningham	First Manned Earth Orbit
Apollo 8 CSM	21 Dec 1968	AS-503	Borman, Lovell Anders	Manned Lunar Orbit
Apollo 9 CSM/LM	3 Mar 1968	AS-504	McDivitt, Scott Schweickart	First Manned LM Earth Orbit
Apollo 10* CSM/LM	May 1969	AS-505	Stafford, Young Cernan	LM Operations Lunar Orbit - No Landing
Apollo 11* CSM/LM	July 1969	AS-506	Armstrong, Collins, Aldrin	First Lunar Landing Attempt

*Yet to be flown as of April 1969.

Six of these 15 compensated coefficients which represent the IMU attitude drift and accelerometer output during zero acceleration can be measured directly during free-fall coasting flight. Deviation from expected behavior is a cause for concern. This has happened. The residual bias acceleration indication term of an accelerometer measured for the first time in flight had in each case changed from the prelaunch value but by an amount fully acceptable from a performance point of view. The real question was, were the units deteriorating or were the changes a result of electrical transients during launch operations affecting the magnetic state of the accelerometer torque generator, and, consequently, affecting the subject error term?

The first case occurred in Apollo 7. During free fall the Y accelerometer indication differed from its small pre-launch value to a value which was for all intents equal to zero. The long coasting period after the boost into orbit with no apparent output from the accelerometer was disconcerting .. so much so that a small translation maneuver along the Y axis was executed with the reaction control jets just to see if the accelerometer was alive, even though the failure detection logic for the accelerometers had indicated the unit was normal. The accelerometer responded precisely to the special maneuver and the problem disappeared.

In the second case, an accelerometer performance change which must have occurred about the time of liftoff of Apollo 9 left the spacecraft indication of orbital perigee lower than that indicated from the ground tracking and the Saturn guidance. The small change in accelerometer performance was measured in flight and the onboard computer compensation changed appropriately by telemetry signals from the ground.

Accelerometer Performance

The ability to measure this accelerometer error component in flight is a result, of course, of the spacecraft during free-fall coasting flight being in an excellent near zero acceleration test environment as long as: (1) the spacecraft average rotational rates are sufficiently small; (2) the IMU is sufficiently close to the spacecraft center of gravity; (3) the spacecraft is not dumping wastes or venting gases; and (4) the reaction control jet activity forces holding attitude in a limit cycle are sufficiently balanced. These conditions are met with sufficient tolerance enough of the time to use the accelerometer output as an indication of its value with zero input.

The flight of Apollo 8 to the moon gave an excellent opportunity to watch this zero input accelerometer performance because the IMU was left running the whole 147 hours and, unlike earth orbital flights, the spacecraft stayed in sight of the telemetry receiver stations for so much of the time. The complete set of data obtained are shown in Figure 3. The indicated outputs for zero input for each of the three accelerometer units are shown as a function of time when the check was made. In addition, the compensation values determined from prelaunch testing

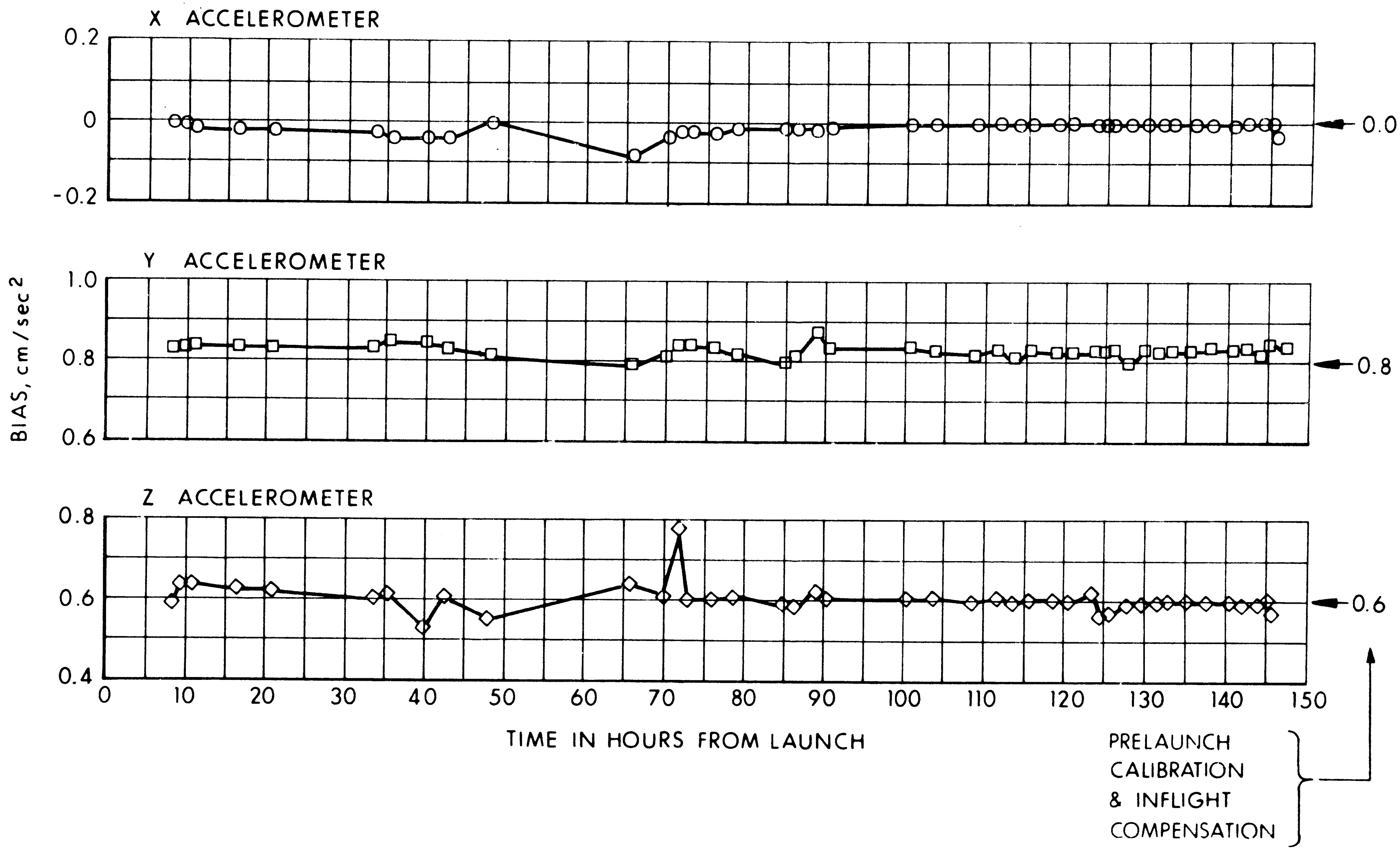

Figure 3 Uncompensated Accelerometer Bias Measured during Apollo 8
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which were loaded into the computer are indicated. The data show the predictability and consistency of this accelerometer performance term.

Gyro Performance

The IMU gyro drift term under near zero acceleration environment can also be measured during flight by dividing the angle change which the IMU realignment process requires by the time period since the last alignment. These results are displayed in Figure 4 for the Apollo 8 flight. Also shown in this figure are the time intervals between the alignments upon which the drift is based. If this time period is short the error in IMU alignment angle change will dominate and add significant error to the indicated drift measurement. The roughness in drift data around 80 hours is due to the shorter period between alignment (once per lunar orbit or about 2 hours) so that alignment errors become significant. Assuming the true drift curves would be smooth, it is possible to infer the actual alignment error which would cause the indicated roughness. In this case, one can infer that IMU star alignment uncertainty during the 10 lunar orbits of Apollo 8 had rms values of 41, 31, and 58 arc seconds about the IMU X, Y, and Z axes.

IMU Gimbal Lock

The final topic in this examination of IMU status is that of IMU gimbal lock. This writer was thoroughly enmeshed in the early design studies leading to the present three-degree-of-freedom gimbal IMU configuration and cannot be accepted as a disinterested observer. The original decision to use only three isolation gimbals for the stable platform and accept the operational nuisance of gimbal lock was based upon the weight, power, reliability, and performance advantages of avoiding the fourth gimbal. The many facets of these arguments for the Apollo requirements were documented in early 1963.(6) The decision was made with the operational knowledge and constraints known at the time and no compelling reason was voiced to cause a change. As a result, attitude maneuvers in Apollo must avoid moving the thrust axis of the vehicle nearer than perhaps 10 degrees of the inertially aligned inner member axis. This is the only constraint. By proper choice of IMU alignment, every phase of the Apollo mission can avoid gimbal lock attitudes. The forbidden zones are small and conceptually simple. Clear signals are given as these attitudes are approached. Moreover, if the IMU attitude is lost due to gimbal lock or any other cause, the IMU is not damaged, a clear warning signal is given, and the procedures for emergency or normal realignment are straightforward. Further attitude cues are available from the backup system and from the scene through the windows.

Flight experience has uncovered an aspect of gimbal lock not anticipated. The crews of both earth orbital flights, Apollo 7 and Apollo 9, noticed in some flight regimes a tendency for aerodynamic torquing or gravity gradient torquing to rotate the spacecraft towards gimbal lock when the IMU had particular alignments. Once each in Apollo 7

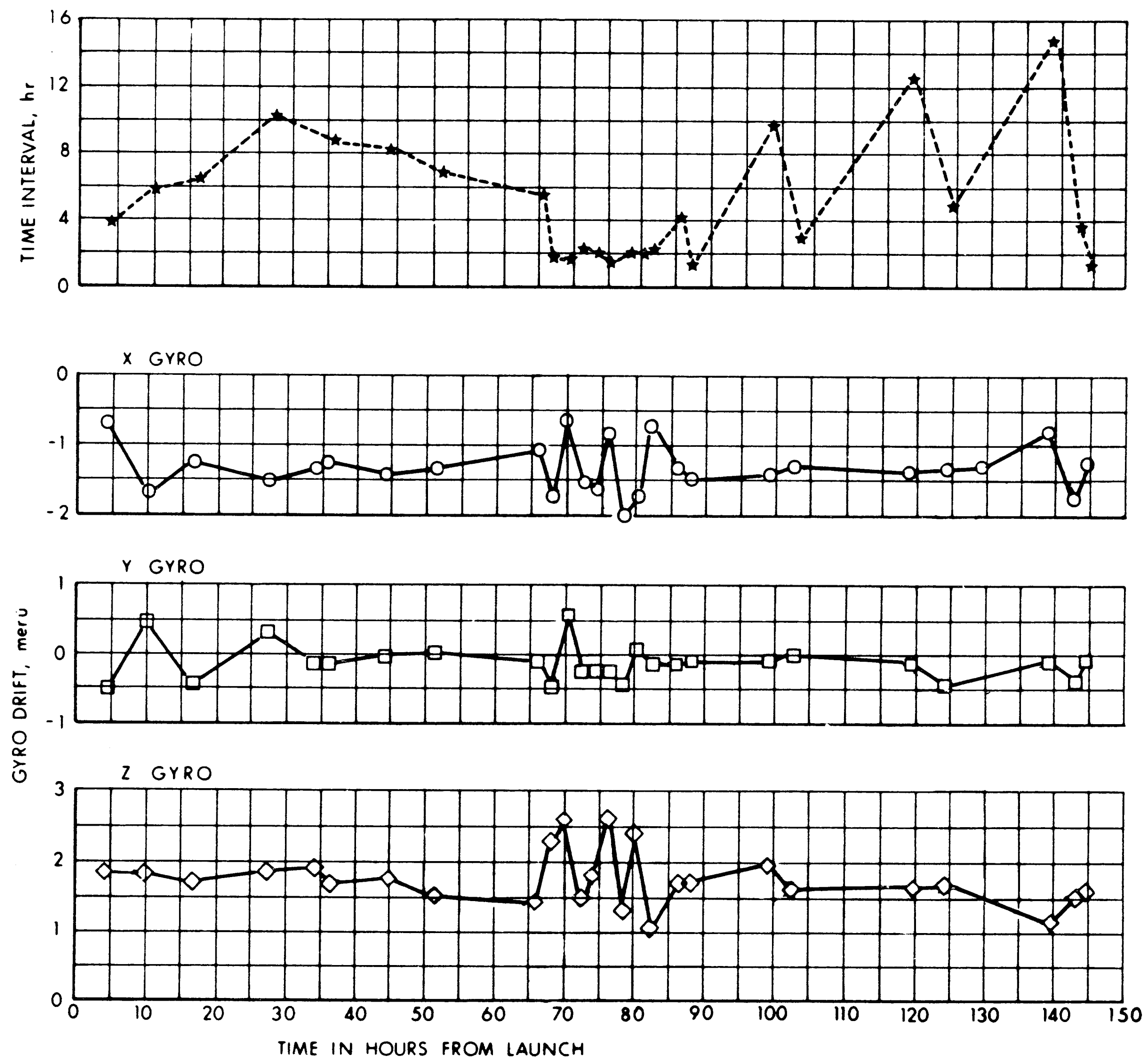

Figure 4 Compensated Gyro Drift Measured during Apollo 8
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and Apollo 9 the spacecraft rotated into gimbal lock while not being watched. Neither crew expressed particular alarm over the event, but they complained about the constraint.

The problem of gimbal lock disappears while the IMU is not on and early mission planning expected that most of the time this would be the case because of spacecraft electrical power considerations. However, the IMU was left running and aligned during the whole of the Apollo 8 mission and it appears that this will be the rule for all lunar trips. At least one crewman was awake at any time and could watch for possible random spacecraft tumble into gimbal lock. The near earth torquing effect mentioned above was absent, of course. Loss of IMU alignment due to gimbal lock did not occur in this lunar mission.

But the attitude constraint imposed by gimbal lock is indeed an awkward nuisance at times. Mission design must take it into consideration and the crew must be aware of it during attitude maneuvers. Although the computer can take over the task of watching for gimbal lock and avoiding it by appropriate schematic maneuvers, this feature has not been authorized for the computer fixed memory program.

An experiment performed in earth orbit on Apollo 9 used the IMU and the erasable memory in the computer to take over attitude control in a wide deadband automatic control (see section on control systems). This was extremely efficient in the use of reaction control fuel. But of significance here, this crew felt that they would be perfectly willing to operate in this mode while all three slept; the control system would keep attitudes well away from gimbal lock.

The lunar landing will be made with the present 3-gimbal IMU. The question as to whether the complexity and performance compromise of the all-attitude 4-gimbal configuration was worth avoiding under the prevailing circumstances at the time of design commitment will not be answered. However, with the present state of the art, an all-attitude IMU would be chosen without a doubt either by the addition of the necessary redundant gimbal or by the use of an appropriate computation algorithm with a direct structure mounted gyro implementation.

THE OPTICAL SUBSYSTEM

The optical subsystem in the command module consists of a two line-of-sight, 28 power, narrow-field-of-view sextant and a single line-of-sight, unity power, wide-field-of-view scanning telescope both mounted on a rigid navigation base with the IMU. The lunar module optical subsystem is a simple unity power periscope mounted on the LM navigation base which supports the LM IMU.

Star Visibility in Scanning Telescope

One of the design problems associated with the Apollo optics has been that of star visibility in the CM scanning telescope when the spacecraft is in sunlight. The problem arises from the particular design constraints of this optical unit. The scanning telescope must have a wide field of view so that constellations can be recognized and the individual navigation stars identified. It must have controlled articulation of its line-of-sight direction to provide sighting angles for the computer and for use in designating targets to the narrow-field-of-view sextant. The physical space available for the optical elements is severely restricted, there must be no protuberances outside the command module skin, and the scene must be transferred from the objective at this external skin, a distance of almost two feet, to the required location of the eyepiece inside the cabin. The restricted space available limited the means of obtaining line-of-sight articulation to a double dove prism which is tilted and rotated appropriately outboard of the objective lens system. Although efficient in use of space, this scheme introduces a large piece of glass just where it can cause the most mischief in scattering light into the instrument. Shielding this prism effectively from direct and indirect sunlight to avoid the scatter has not been possible due to limited space and the wide field of view of optical coverage which must be kept clear. Although the double dove prism allows controlled pointing of the line of sight, it unfortunately also rotates the image, confusing the viewer. The correction is achieved by another prism inside the instrument. The two prisms and the lens trains associated with the long optical transfer to the eyepiece cause about 50% signal light loss as received at the eye.

In summary, the scanning telescope suffers both from light loss and a propensity to undesirable light scatter and the associated washout of the background when the sun or other bright source outside the field of view illuminate the objective. In addition, the necessary wide field of view means, of course, that a random aim of the instrument is likely to find the sun, earth, moon, or part of the spacecraft in view. These bright objects in view either prohibit sighting or severely degrade the eye's accommodation and ability to see stars.

Analysis and test forecast that even the brighter navigation stars would not be visible from space through the scanning telescope within perhaps 70 degrees of the sun. Star visibility and constellation recognition would only be possible with the spacecraft oriented with the optics pointed away from the sun shaded by the spacecraft and with the sunlit earth or moon out of the field of view. The lunar trip of Apollo 8 bore out this forecast. With dark adaption, it was always possible to find and identify constellations and stars in the scanning telescope as would be required when using the computer's spacecraft orientation determination program, P51, as the initial step in an alignment of the IMU from a random orientation. An unplanned test of this occurred. The loss of the IMU alignment on the way back from the moon due to a procedural error required calling up program P51. The scanning telescope was used to identify and acquire the needed stars and the IMU alignment was quickly accomplished without problem.

This experience with Apollo 8 provided no information on the degradation in visibility that light scatter from a docked Lunar module would cause. This docked configuration was flown in earth orbit on

Apollo 9 but the considerable source of light from the sunlit earth directly below does not represent conditions in cislunar space. Tests reveal that star recognition in the scanning telescope while docked to the lunar module will require tight constraints on spacecraft attitude.

Of course, light scatter in low orbit is not a problem in the use of the scanning telescope in the nighttime shadow of the earth or moon. Abundant stars are clearly visible. Also, the other major use of the scanning telescope in earth or lunar orbit to track landmarks for navigation data does not suffer from problems of visibility.

Star Recognition in Sextant

The sextant articulating star line of sight makes the precision measurement of star direction for the IMU alignment. For the entire Apollo 8 mission the IMU was scheduled to remain operating continuously. Periodic realignment was performed 30 times with the sextant each time using the automatic star pointing acquisition of program P52. The navigator reported he never had any doubt that he had the right star in the two degree field of view. The correct star was the only bright star in the center of the field. For added confidence, after marking on two stars, the computer program provides a check of the star angle difference displayed to the operator. This is the difference of the measured angle between the two stars used for the alignment from that angle calculated from the computer's catalog of star coordinates for the identified stars. The average difference displayed was 0.007 degree (24 arc seconds) for the 30 alignments. In addition, several times a check was made by asking the computer to point to a third star. Each time the requested star came up in the cross hairs to the satisfaction of the navigator.

The sextant clearly does not suffer from the low light transmission and the light scatter problem of the scanning telescope. This was as expected. The narrow field of view, the use of a simple mirror rather than a complex prism to point the line of sight, the better light shielding possible, and the magnification makes the sextant's visibility of stars superior.

Particles and Deposits

Not forecast, however, was the confusion caused by the debris cloud from the SIVB after separation. Confusion of particles and stars was so bad that it was not even possible to calibrate the sextant by superimposing star images on top of themselves while near this debris cloud.

The scanning telescope was particularly susceptible to confusion from particles glinting in the sunlight. For the first day or so of Apollo 8 particles, probably dust, were emitted in fewer and fewer amounts into the field of view every time the optic's shaft was rotated. This did not occur in Apollo 9. Particles caused by dumping fluids from the spacecraft also generated confusing false star images. No fluid should be dumped just before or during optics use.

During the design of the Apollo optical systems, there was much concern over the potential problem of deposits on the external glass surfaces degrading optical performance. The threat would be from particles or film arising from the residue of reaction control jet firing or waste dumping into space. No degradation of visibility as flight time progressed was observed in any instrument on any mission. A special examination was made late in the flight of Apollo 7 by removing the eyepiece so that scatter or glare on the objective system from deposits would be visible. No glare existed. It appears that these surfaces will keep themselves sufficiently clean in space.

LM Alignment Optical Telescope

The earth orbital flight of Apollo 9 provided the first opportunity to evaluate the IMU alignment optical telescope in the lunar module. The LM optics are of unity power and wide field of view like the CM's scannng telescope. But the LM's alignment telescope does not scan and thus avoids the light scattering of double dove prism. Also, the LM, unlike the CM, does not limit space for the installation of effective light shields. The alignment telescope was tried while docked to the CM. Light reflected off the CM was a bother as expected so the final alignment was scheduled for after separation from the CM. When free of the CM, the LM IMU alignments went smoothly using the alignment telescope.

Visibility through this instrument while the LM sits on the surface of the moon has been simulated and found workable after suitable light scatter shielding was incorporated. Final evaluation must await the lunar landing mission itself. Several alternative alignment schemes on the lunar surface are available which do not require star visibility.

Other Features of the Optics

Apollo 9 also demonstrated other methods of IMU alignment with the optics. Data in the form of tables were carried aboard with which the celestial coordinates of planets could be obtained as a function of time in the mission. CM IMU alignment was exercised using the coordinates of Jupiter as a non-catalog star inserted into the computer's memory. Automatic optics pointing was called and Jupiter appeared in view with a fine display of her moons. Marking on Jupiter and another star, the P52 program was used in a demonstration of the use of a planet for IMU alignment.

A sun filter is provided to make it possible to use the sun as an alignment target and was exercised for the first time on Apollo 9. The filter worked well functionally -- even sunspots could be counted -- but the actual alignment was not performed.

Also in the Apollo 9 mission, several days after the activities with the LM were completed and it had been sent away alone into a high apogee orbit, the LM trajectory parameters were sent up to the CM computer. Using the P20 rendezvous program, the computer pointed the sextant at the LM and the navigator picked it up in the eyepiece 2700 nautical miles away and was able to keep track of it until it was blocked from view.

The automatic pointing capability of these optics to celestial, surface, ororbiting targets suggests the possibility of a camera replacement of the eyepieces. A 16 mm movie camera was adapted to the sextant and carried on Apollo 8. Motion picture film of the lunar surface were obtained from orbit in this manner; however, program constraints in supporting the lunar landing have not yet allowed full exploitation of cameras on the optics.

THE COMPUTER

The Apollo flight computer, one each and identical in the CM and LM, is the heart of the system in each spacecraft. It's computational capabilities are enormous for its size of only one cubic foot and 70 pounds and for its operating power of only 70 watts. It's necessary features include the ability to handle a number of different computational problems simultaneously in real time interleaved in a single central processor on a priority basis. Also of special note is the large number of signal interfaces with which it communicates with other systems throughout the spacecraft.

The use of the system in supporting a manned or unmannned flight, an earth orbital or lunar flight, and a command module or a lunar module, is determined by the computer program written into its hard wired nondestructable memory of over half a million bits. Leading up to the recently completed lunar landing programs, there have been a total of nine significantly different flight programs developed, Table II. The specification, formulation, design, coding, testing, and documentation of these programs has been a major undertaking, involving at least as much effort at the MIT Instrumentation Laboratory as was expended on the system hardware design. The change control activity alone has seen over 750 program change requests from the various government teams and contractors processed through the program change board at MSC, Houston. Extensive testing of the programs is carried out at various facilities at MIT and elsewhere to certify that the program is sufficiently error free for flight and that logical curiosities and anomalies not fixed have suitable work-around procedures designed, advertised, and which appear in the flight crew checklist when appropriate.

Flight experience with the computer has shown the value of the error detection and alarms built into the hardware and software programs and the memory protection offered by these features. During Apollo 6 -- the second Saturn V launch with unmanned spacecraft CSM 020 -- considerable noise arriving from the telemetry receiver appeared at the up telemetry channel to the computer. The error checks and alarm in the decoding in the computer was able to reject the noise when the signals failed the logical test and bad data into the erasable memory was avoided.

For many situations, the computer program associated with the keyboard is able to deteect illogical inputs to the DSKY due to mispunched keys. The DSKY will signal "operator error" so that the astronaut can try again. Other error detection has been designed and experienced in flight which causes the computer to "restart" automatically. Restart causes the program to go back a few steps to a point where the computation state was put into memory and then start fresh from that point. This is done so quickly that the astronaut is only aware that it has happened by the fact that the restart light comes on. More serious internal logical problems or procedural errors have raised alarms requiring the astronaut manually to recycle back to the start of the program in progress. In the few cases experienced, the crew have been able to recover without help. In such situations the ground has examined the state of the memory to verify that memory corrections do not need to be telemetered up. The important point is that the computer and its associated operating procedure is generally forgiving of the errors of the type experienced. The few problems which have happened have not occurred during the stressful critical mission phases where training has been intensive and procedures are followed carefully through well tested logical paths in the program.

The use of a digital computer to tie together the measurement, processing, and commanding functions of the system has provided design flexibility of enormous value. Computer program changes are not made easily and can be unsafe without time-consuming retesting. But in many cases changing the program to accommodate hardware problems has nevertheless saved considerable tine, effort, and expense.

THE CONTROL SYSTEMS - DIGITAL AUTOPILOTS

The attitude control of the Apollo spacecrafts have presented a design challenge. These autopilots have had to consider many permutations of several variables.

First are the variations of spacecraft configurations: (1) the CM alone in its atmospheric entry phase; (2) the command module and service module together; (3) the CSM when docked to the LM under CM control; (4) the LM when docked to the CSM under LM control; (5) the descent configuration of the LM; and (6) the ascent configuration of the LM after leaving the descent stage on the lunar surface.

Second are the variations provided for achieving control torques: (1) the arrays of small reaction control thrusters on the CM, on the SM, on the LM ascent stage which provide control forces; (2) the fast responding service propulsion engine gimbals; (3) the slow responding trim gimbal drive of the LM descent engine; and (4) the aerodynamic forces of the entry of the CM.

Third are the variations in the flight regimen from free-fall coasting flight, rocket powered accelerated flight, and aerodynamic influenced atmospheric entry.

And fourth are the wide variations in dynamic properties of each of the spacecraft configurations as fuel is expended, mass and inertia vary, bending frequencies and their damping vary, and fuel slosh modes and coupling vary.

The original attempt at these autopilot designs was a conventional analog system approach. But in 1964, NASA wisely made the decision to incorporate these autopilots into the CM and LM digital computers. It was easily demonstrated that a direct digital equivalent of the signal processing of the analog autopilot design candidates would not work. Sampling rates would have to be too high in relation to the data processing speed of the computer. Success depended upon design approaches adaptable to the nature of digital processing and capitalizing upon the flexibility and nonlinear computations easily and directly available in a digital computer.

All configurations, CM, CSM, and LM, of the digital autopilots in free fall and powered flight have now been flight tested successfully with a wide sampling of flight environments and with excellent performance and efficient use of fuel. About 10% of the memory in the CM and LM computers is devoted to autopilots. During times of high activity, only 20 to 30% of the available computation time is used in autopilot data processing.

Rate Derivation for Control

Essential to control systems is some form of stabilization signal. For analog autopilots this is obtained typically from angular rate indicating gyroscopes mounted to the vehicle structure or other specialized sensors. The Apollo primary system does not have sensors to measure angular rate directly. The digital autopilots derive spacecraft attitude rate by processing available attitude signals. The simple ratio of IMU indicated attitude difference divided by time difference is sufficient and works quite adequately for roll control about the thrust axis of the powered flight autopilots. But this simple formulation is grossly inadequate for the necessarily more complicated wide bandwidth control situations. The attitude signals come from the IMU in quanta steps of about 0.01 degree, and at low rates no new information is available for rate indication until the next angle increment occurs. The system gets around this problem by providing in the computer a model of the spacecraft response to applied torques. This model includes the torque level obtained from firing reaction control jets, the torque obtained from the thrust acting on the engine at the existing engine gimbal angle, and the presently existing spacecraft moments of inertia. Now as attitude jets are fired and engine gimbal angles changed, this model can immediately change the angular rate estimation. Then periodically the integral of this rate estimation can be compared with the IMU orientation angles and the weighted difference applied to the state of the model to bring it back to consistency with the actual indicated time history of spacecraft orientation.

This rate estimation works very well. It is interesting, though perhaps not significant, that while the spacecraft is sitting on the launch pad this estimation produces a clean signal of earth's rotation rate. But what is significant is that a fast-responding low noise indication of spacecraft rate is obtained without the need for special gyroscopes and without the associated weight, power, and reliability penalties.

Autopilot Gain Scheduling

The computation capabilities of the digital computer has also made feasible a very effective gain scheduling for the autopilots. The acceleration measured by the IMU during powered flight allows the computer to estimate spacecraft mass loss as propellant is expended based upon engine specific impulse. With this knowledge of spacecraft mass, predetermined polynomials can be evaluated to give good estimates of spacecraft moments of inertia which directly allow an estimate of the angular acceleration produced by the controlled torques. Using this to adjust the gains within the autopilot, the wide variations in dynamic characteristics of the spacecraft can be accommodated with near optimum response.

Free-Fall Control Systems

The free-fall flight control systems for the CM, CSM, and LM operate using appropriate efficient phase plane logic. Each iteration cycle of the autopilot examines the existing angle error and angle rate error. If either of these are outside chosen deadbands the computer schedules and executes open loop the time history of two reaction jet firings which should bring both angle and rate errors to zero simultaneously. Inside the chosen deadband, the control oscillates slowly in a minimum reaction jet impulse limit cycle determined by the 14 milliseconds minimum firing permitted by the design of the jets. The characteristics of these systems are very efficient in reaction jet fuel expenditure and they minimize the number of on-off cycles the jets must tolerate.

An example of the flexibility with which the digital computer can augment control system design is a last-minute change added to the Apollo 9 flight procedures. The crew were desirous of having the control system in free-fall orbital flight respond to orbital rate calculated within the computer so that the spacecraft would hold attitude with respect to the local vertical without their attention. This feature had been proposed earlier as a program in fixed memory but had not been incorporated. A week before the scheduled launch, a simple procedure to load a program and data into erasable memory by means of a few dozen DSKY keystrokes was transmitted to the astronauts. This provided the crew with the desired feature which operated in orbit as advertised.

LM Ascent Powered Flight Control

The LM ascent powered flight autopilot obtains control torque only by means of the reaction jets. The engine is fixed; it cannot swivel.

This control, then, operates very similarly to the free-fall coasting flight autopilots described above, but with the addition that the system estimates the torque arising from the offset of the main engine thrust from the center of gravity. Controlled limit cycles then will normally operate so that, unless error margins are exceeded, reaction jets will be fired only to oppose the main engine thrust misalignment torque.

CSM Powered Flight Control

The powered flight autopilot for the CSM uses a fast-acting main engine swivel to obtain pitch and yaw control torques. The computer compares estimated spacecraft rate with the desired rate being commanded by the guidance programs. The resulting rate error is integrated and shaped by a seventh order difference equation filter, the outputs of which command the engine swivel servos. At the time the design was committed, there was little confidence in the dynamic bending mode and fuel slosh models available, especially for the CSM docked to the LM. Bending modes lower than one cycle per second were reported to be possible. Accordingly, the gains and filter characteristics were chosen conservatively so that a wide margin of tolerance to variations in spacecraft dynamic characteristics was obtained but at a penalty of a slow response time and low static gain. A special loop to calculate thrust misalignment is provided so that the steady state autopilot error resulting from this low loop gain would not be excessive. However, initial thrust misalignment causes a large initial attitude error transient peaking at about 10 seconds after engine ignition at an amplitude depending upon initial conditions of thrust misalignment. Short burns of around 10 seconds while docked to the LM under this control leave fairly high residual out of plane error velocity. However, these errors are within Apollo tolerances. Better models on the docked spacecraft dynamics have since become available after the original design was frozen. Bending frequencies are assured to be higher than 2 Hz. A much higher gain design will appear in Apollo 10 which has captilized on this better information. Residual velocity errors should be reduced to about 10% of those of the earlier design.

Five CSM-docked-to-the-LM burns of various lengths were made in the Apollo 9 mission using the lower gain control system. The last one had an unusual LM mass since the LM had already exhausted most of its descent engine fuel in a previous burn. This is not a configuration of the lunar mission and was not recognized in the control system design. A large residual out-of-plane velocity error at cutoff of almost 12 ft/s occurred, but with the magnitude components expected from simulation.

LM Descent Powered Flight Control

The lunar module descent powered flight autopilot was originally intended to use a slow responding engine swivel to put the thrust vector through the center of gravity with a slow computation loop. Dynamic control torques in all axes would be provided by reaction control jet firings. Although the engine trim gimbal could be commanded at the fixed rate of only 0.2 deg/s on each axis in response to discrete signals from the computer, the challenge to make this provide most, if not all, of the dynamic control was motivated by the reaction control jet fuel savings and the much fewer number of firings which would result. Response to the challenge was ambitious because of the very slow speed with which the engine swivel could be made to effect changes in torque even though 5 times as much torque is obtained at maximum angle as it obtained from firing a pair of jets. A jet torque minimum impulse is obtained within 15 milliseconds of the command while the same impulse requires 400 milliseconds with the engine gimbal. Nonlinear control laws were examined and a formulation for a third order minimum time control was achieved.(10) This autopilot uses computer generated estimates of attitude, attitude rate, and attitude acceleration every 0.1 second upon which to base a policy for commanding the appropriate axis of the trim gimbal in a time sequence of its three possible states: plus 0.2 deg/s, zero, and minus 0.2 deg/s. This time sequence is such that the spacecraft angular acceleration, angualr velocity, and angle error would theoretically all be brought to zero simultaneously in the fastest possible time. Provision was made to fire the reaction jets appropriately if the angle error got past a threshold, but in simulations this logic was rarely exercised.

The benefits of this nonlinear control law became particularly evident when it appeared that the total time the downward firing jets could operte in the descent configuration was severely limited because of the danger of burning of the descent stage by the jet plume. More thermal insulation would have been a serious weight penalty which was avoided by the development of this autopilot design requiring almost no jet activity. But the real test of this LM descent autopilot control system came in Apollo 9. Although the possibility for using the LM descent engine to push the docked CSM was considered as a backup for service propulsion failure, this configuration became essential to the Apollo 9 mission. It was the only safe way of achieving the long descent engine burn to qualify the engine for the lunar landing. But with the LM pushing the CSM another problem appeared. The LM forward firing jets then were severely limited due to their impingement on the docked command module. The solution was to provide the ability to have the computer cease all X axis jet activity forward or aft when so commanded by astronaut DSKY input and fly the LM control system pushing the CSM entirely with the slow moving trim gimbal. This was how the descent engine docked burn of over 6 minutes duration and over 1700 ft/s was controlled in the Apollo 9 flight. The residual cross axis velocity error at the end of the burn indicated only 0.1 ft/s. Attitude error during the burn remained small and no evident excitation of bending or slosh modes was seen.

RENDEZVOUS NAVIGATION

A particularly critical phase of the Apollo mission is the rendezvous of the LM with the CSM following the LM's ascent from the lunar surface. The navigation problem of the rendezvous is to measure the positions and velocities of the two spacecrafts in a common frame and from this to determine the maneuvers required to bring the two spacecraft together in a prescribed fashion. The characteristics of the maneuvers and flight paths followed in manned space flight rendezvous have been those of the so-called "concentric flight plan"(11) which was developed to make backup possible both by the astronauts using simple onboard chart solutions and by ground tracking calculations in case the primary system fails.

Rendezvous Measurements and Calculations

The active partner in the lunar landing mission rendezvous, the LM, makes its navigation measurements with the IMU during maneuvers and with the rendezvous radar during coasting phases. The rendezvous radar, when successfully tracking the CSM, measures directly the range and range rate for the computations. The radar antenna gimbal angles with respect to the spacecraft are sent to the computer, along with the spacecraft orientation measured by the IMU from which the computer can determine the direction to the CSM in the stable coordinate frame to which the IMU is aligned. Since the radar antenna and the IMU are mounted at separate locations in the spacecraft, the processing provides for an unknown but assumed constant misalignment between the two and corrects the line-of-sight calculation for a computed estimate of this misalignment. The estimates of this misalignment and the position and velocity of one of the two vehicles are updated recursively in a Kalman optimum filter as radar measurements are incorporated periodically. The navigation states are used in several programs in the computer to produce the targeting parameters for the various rendezvous maneuvers which are executed with the inertial guidance steering.

While the LM, as the active vehicle, is performing its rendezvous navigation, the sensors and computer in the CM are being used to provide an independent check on the LM rendezvous navigation. It also is prepared to take over the active roll in case the LM gets into trouble and needs rescue. In these functions the navigator in the CM makes LM line-of-sight direction measurements with the CM optical system. In addition, range data measured within the communication system will be available to the computer in future spacecrafts, but for the CM rendezvous navigation in the Apollo 7 and Apollo 9 missions only line-of-sight direction data were used for state estimates in the computer.

CSM Active Rendezvous

A CSM active rendezvous was exercised in earth orbit in the Apollo 7 mission using the Saturn SIVB stage as a target. The onboard computer's rendezvous program, P20, satisfactorily pointed the optics to the SIVB for optics acquisition; the navigator, Don Eisele, was able to do satisfactory tracking and marking on the target, and the computer came up with the critical TPI (transfer phase initiate) maneuver data to put the CSM on a trajectory intersecting the SIVB. This onboard solution matched the ground tracking generated solution within a fraction of a foot per second. The onboard solution was used to target and guide the 18 ft/s maneuver using the reaction control thrusters. Further optical tracking following this was performed for a midcourse correction which was only about 2 ft/s. No other midcourse corrections were needed. Braking was done visually by the crew.

LM Active Rendezvous

The LM was active with the CSM the target in the earth orbital rendezvous on Apollo 9. This rendezvous necessarily had to succeed for the LM crew to return safely to earth. Three major independent navigation processes were underway simultaneously: (1) in the LM, McDivitt and Schweickart were using the LM rendezvous radar, IMU, and the LM computer; (2) in the CM, Scott was using the CM optics, IMU, and CM computer; and (3) on the ground, mission control was using the tracking network to feed data to the MSC real-time computer complex. Initial separation of the two vehicles was followed by several maneuvers which brought the LM to the desired intended position and orbit from which the rendezvous would start: the LM about 10 nautical miles above and about 70 nautical miles behind the CSM orbiting at about 133 nautical miles altitude. The descent stage was jettisoned and the reaction control jets used to make the initial rendezvous maneuver based upon the LM onboard solution of (-39.2, 0, -0.7).* The ground solution checked well: (-39.3, 0, 0). Although the CM was navigating, it did not have the processor in its computer program to generate these maneuver solutions. This capability will be in the next flight, Apollo 10. Its determination of the LM state vector was comparing very well with the other sources.

The initial rendezvous maneuver was intended to cause the LM to pass through a point behind and below the CSM where another burn called CDH, constant delta height, would put the LM in a trajectory below, behind, and concentric with the CSM orbit. The onboard LM solution for the CDH maneuver (-39.2, +0.1, -13.7) was executed with the LM ascent engine. The ground solution was (-38.2, -0.9, -15.1).

With the LM behind, below, and concentric with the CSM it was now catching up with the CSM. The computation capability for the final critical TPI (transfer phase initiate) maneuver to put the LM on an intercept trajectory with the CSM was included in the CM computer program so that this solution was also available. The LM navigation and computation indicated (+19.4, +0.4, -9.7). The CM navigation and computation indicated (+19.5, +0.5, -9.0). The ground tracking solution was (+19.6, +0.1, -10.5). Another parameter of the TPI maneuver of special interest was the time, measured from liftoff from earth, that the maneuver was to be initiated. The three sources agreed closely: LM (97 hours: 57 minutes: 59 seconds), CM (97:58:09), and the ground (97:57:45).

The TPI maneuver was made by the LM with its own solution using the reaction control jets. During the 20 minute coast towards the CSM,

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*X, Y, and Z velocity change components in ft/s; X horizontal in orbital plane in direction of orbital velocity, Z vertical down, and Y completes the right-handed set.

the intercept trajectory was improved by two small midcourse corrections determined onboard. The line-of-sight rates as the LM approached the CSM indicated zero. The braking maneuver was done manually to the prescribed velocity step versus range schedule.

This rendezvous exercise in earth orbit was designed to duplicate as close as possible the return of the LM to the CSM orbiting the moon after the lunar landing. The excellent comparisons of the independent navigation being performed on each vehicle and with ground tracking results provide a measure of the spacecraft development maturity in preparation for the lunar landing.

MIDCOURSE NAVIGATION[12][13]

For onboard midcourse navigation between the earth and moon, the apparent positions of these bodies against the background of stars as seen from the spacecraft is measured by superimposing images of the two-line-of-sight sextant in the command module. The output of the sextant is the measured angle between a known reference star and a visual navigation feature of the earth or moon. This angle is transmitted to the computer as a serially incremented count of 10 arc seconds per count. Therefore, before its use and periodically during its use, the navigator must zero the instrument by superimposing any celestial image from the two lines of sight on top of itself. This calibration process zeros the navigation angle counter in the computer and corrects for any fixed angle error in the instrument which can arise from thermal changes of the space environment.

To make a navigation sighting, the midcourse navigation program in the computer, P23, can point the sextant's two lines of sight at the reference star and landmark identified by the navigator. It is then the navigator's task to center the superimposed star image onto the landmark, if landmarks are being used, or onto the near or far substellar point of the horizon, if the horizon target is being used. When superposition is achieved, the navigator pushes the "mark" button which signals the computer to record the navigation angle and the time of mark. Using a Kalman optimum recursive filter formulation, the computer then determines the state vector change this measurement would cause if incorporated and displays the resultant position and velocity change magnitudes. If the navigator is satisfied with the display and is satisfied subjectively with his mark, he allows the computer to incorporate the state vector change.

The first manned trip to the vicinity of the moon of Apollo 8 during Christmas 1968 gave an excellent test of the Apollo system's onboard navigation capability. Although ground tracking navigation was the primary system, the onboard navigation system had the task of confirming a safe trajectory and providing a backup for return to earth in the remote chance that ground assistance became unavailable for onboard use.

Earth Horizon Navigation Reference

Apollo 8 was to use sun-illuminated visual horizons rather than landmakrs for operational simplicity. It was expected and confirmed from earth by Astronaut Don Eisele in Apollo 7 that the earth's horizon does not provide a distinct target for visual use. Moreover, the filter in the sextant beamsplitter, designed originally to enhance the contrast between water and land when looking down at the earth, filters out the blue in such a way as to make the horizon even more indistinct at the higher altitudes where weather phenomena are less likely to cause optical uncertainty in the altitude. Originally a blue sensitive photometer had been designed for horizon detection in the phototype sextant models but was removed from the production systems since a decision had been made that ground tracking would be the primary source of midcourse navigation. Without the photometer, interest in the earth's horizon as a visual target resulted in demonstrations on simulators that, in some subjective way, the human with a little experience can choose an altitude sufficiently repeatable, at least as good as only several kilometers. Accordingly, a few weeks before the Apollo 8 launch the navigator, command module pilot Jim Lovell, spent a few hours on the sextant earth horizon simulator at MIT/IL in Cambridge for training and to calibrate the horizon altitude he seemed to prefer. He was remarkably consistent in choosing a location 32.8 kilometers above the sea level horizon. This value was recorded into the onboard computer.

The plan for the mission was to examine the sextant angle measurements made early while still near the earth and, based on the spacecraft state vector measured by ground tracking, infer in real time the horizon altitude Lovell was using. This was done because there was sufficient justification to suspect the fidelity of the simulation at MIT/IL. After the first eleven sightings on the earth at a distances from 30 to 35 thousand nautical miles from earth, it was estimated that he was using an 18.2 kilometer altitude and the onboard computer was reloaded with this new value. (Later during the mission it was agreed that a truer estimate was nearer 23 kilometers but the value was not changed since the difference was too small to be of concern.)

Translunar Midcourse Navigation

Following the horizon calibration, the first translunar midcourse correction of almost 25 ft/s was performed. The large size of this correction was due to trajectory perturbations resulting from the maneuvers performed in getting the spacecraft safely away from the third stage of the launch vehicle. After this midcourse maneuver, the spacecraft state vector in the onboard computer was made to agree with the value obtained from ground tracking. The important parameter of the translunar navigation, predicted perilune altitude, was 69.7 nautical miles — very close to a true value estimate later of 68.8.

The next 31 navigation measurements were made using the earth's horizon, modeled at 18.2 kilometers altitude. Being sufficiently far from both earth and moon during this time it is not surprising that the initially good state vector was degraded. At the end of this period, the indicated perilune was 32 nautical miles below the moon's surface.

The navigator then made five calibration measurements of the sextant which resulted in a change in calibration by the one bit or 10 seconds of arc. With the next nine sightings, still using the earth's horizon, the predicted perilune was increased to 92.9 nautical miles — about 22 nautical miles too high. The exact altitude of the earth's horizon being used was unimportant for these sightings since the distance from earth was now approximately 140,000 nautical miles so that the 10 arc second accuracy of the sextant was the predominant source of error.

The next group of 16 sightings was made using the lunar horizon at a distance of about 45,000 nautical miles. As would be expected, the first few of these resulted in fairly large changes in the estimated state vector while the remaining had a very small effect. At the end of this group of measurements the indicated perilune was 67.1 nautical miles.

The final set of 15 translunar sightings was made about 30,000 nautical miles from the moon with little additional effect on the perilune estimation since it was now quite accurate. The final indication was 67.5 nautical miles or about 1.3 nautical miles lower than a value later reconstructed from ground tracking data. At this time the onboard and ground tracking data were practically identical and consideration was being given to using the onboard state vector for lunar orbit insertion. Although the state vector update from the ground hardly changed the onboard value, it was performed since there was no overriding argument to deviate from the flight plan. There were no more translunar navigation sightings. A second midcourse velocity correction of 1.4 ft/s was made 8 hours before lunar orbit insertion which lowered the perilune by about half a mile.

The transearth flight of Apollo 8 after 10 lunar orbits also provided a good measure of the onboard navigation capability. The transearth injection firing of the service propulsion system was targeted by ground data and executed in back of the moon by the onboard digital autopilot and guidance systems. This 3522.5 ft/s maneuver was followed by a single midcourse correction of 4.8 ft/s 14.7 hours later resulting in entry conditions at 400,000 feet altitude above the earth 57 hours from the moon which were 0.8 ft/s faster and 0.1 degree shallower than planned.

Although the primary navigation during this period was again the ground tracking network, 138 onboard navigation measurements were performed by Lovell as a monitor and backup. In order to see what would have happened without ground assistance, the actual onboard measurements were incorporated in a simulation with the computer initialized to the actual onboard state vector as it existed when the spacecraft emerged from behind the moon. The single transearth midcourse correction was added appropriately to this simulation in accordance with that actually measured by the inertial guidance system. (In the actual flight a new ground determined state vector was loaded into the computer at the time of this maneuver.)

The last of the 138 measurements was completed 16 hours before entry. The incorporation of these measurements left a hypothetical onboard estimate of entry flight path angle at 400,000 feet of -6.38 degrees as compared with the ground tracking estimate of -6.48 degrees. Either value difference was well within the conservative safe tolerance of ±0.5 degree from the center value of the entry corridor of -6.5 degrees. Another parameter of concern at this entry interface is the error in altitude rate. The onboard estimate of this quantity differed from that estimated by ground tracking by 236 ft/s. The conservative allowable tolerance is ±200 ft/s.

It should be emphasized that in the event ground data were not available, the plan was to continue the onboard measurements to optimize the final midcourse correction and state vector for safe earth atmospheric entry. In the absence of actual measurements the above simulation was extended using the planned sighting program with standard deviation errors in the sextant of 10 arc seconds and in the horizon of 3 kilometers. In addition, bias errors of 5 arc seconds in the sextant and 4 kilometers in the horizon were included. The resulting estimation error in the entry angle at entry interface had a standard deviation error of 0.03 degree and a bias of 0.007 degree. The corresponding altitude rate error had a standard deviation of 41.1 ft/s with a bias of 26.5 ft/s.

The capability of the onboard navigation system to bring the spacecraft safely back from the moon seems clearly demonstrated.

EARTH ATMOSPHERIC ENTRY

The CM system has now guided and controlled six earth atmospheric entries from outer space over a wide range of entry velocities. All had satisfactory performance and met objectives even though in two cases the planned entry range was not achieved.

Entry Experience

The unmanned flight of Apollo 3 was designed to give the CM heat shield a high entry heat load test. Accordingly, the onboard system was targeted to reach for maximum range far down near the toe of the entry footprint. Instead of the expected lift to drag ratio of 0.3 the command module unfortunately had only a value of about 0.28 which was not enough to fly the whole distance planned. The spacecraft entered the atmosphere at 28,500 ft/s but fell into the ocean 200 nautical miles short of the recovery forces after flying full lift up in a vain attempt to make good the range. Heat shield test objectives were met, however, and the inertial navigation indicated position was within a few miles of actual recovery coordinates. In all flights following this one, the lift to drag ratio has been extremely close to the intended value.

The entry of unmanned flight Apollo 4 was perhaps the most spectacular from an engineering point of view of any entries yet. Intended to be a high heat rate test on the heat shield, the entry velocity was unexpectedly higher than planned. By error, the ability to send the cutoff

External:

NASA/RASPO	(1)
AC Electronics	~~(3)~~
Kollsman	~~(2)~~
Raytheon	~~(2)~~
Capt. M. Jensen (AFSC/MIT)	(1)

MSC: (21&1R)

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