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APOLLO EXPERIENCE REPORT - GUIDANCE AND CONTROL SYSTEMS: LUNAR MODULE STABILIZATION AND CONTROL SYSTEM

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16. Abstract

A brief functional description of the Apollo lunar module stabilization and control subsystem is presented. Subsystem requirements definition, design, development, test results, and flight experiences are discussed. Detailed discussions are presented of problems encountered and the resulting corrective actions taken during the course of assembly-level testing, integrated vehicle checkout and test, and mission operations. Although the main experiences described are problem oriented, the subsystem has performed satisfactorily in flight.

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APOLLO EXPERIENCE REPORT

GUIDANCE AND CONTROL SYSTEMS:

LUNAR MODULE STABILIZATION AND CONTROL SYSTEM

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SUMMARY

The lunar module stabilization and control system, which is part of the lunar module guidance, navigation, and control system, is designed to control vehicle attitude and translation about or along all axes during a lunar module mission. Vehicle attitude and small translations are controlled by selectively firing the 16 reaction control system jets mounted on the ascent stage. The design concepts used were representative of the state of the art, and six different assemblies constituted the subsystem: (1) attitude and translation control assembly, (2) descent-engine control assembly, (3) rate gyro assembly, (4) attitude controller assembly, (5) thrust and translation controller assembly, and (6) gimbal drive actuator.

Subsystem procurement was at the assembly, or black-box, level, and the lunar module contractor was responsible for black-box integration. Design-feasibility and design-verification tests were performed at the assembly level during developmental phases, using early production equipment.

Subsystem-level testing was accomplished in vehicles and in the lunar module contractor Full Mission Engineering Simulator/Flight Control Integration Laboratory. All detailed test objectives were satisfactorily accomplished before the Apollo 11 (lunar module 5) mission.

INTRODUCTION

The initial concept and configuration of the lunar module (LM) guidance and control (G&C) system evolved during the period from 1963 to 1964. The initial concept included the Government-furnished primary guidance subsystem for the necessary guidance and navigation functions and the contractor-furnished stabilization and control subsystem (SCS) for the vehicle stabilization and control functions.

Additionally, the SCS was to act as a backup guidance system that would permit attainment of a safe lunar orbit if primary guidance were lost.

By mid-1964, the SCS stabilization and control functions had become fairly well defined, the design-control specifications had been completed, and subcontracts had been awarded for most assemblies. During the fall and winter of 1964, NASA and two participating contractors reviewed the LM and G&C requirements and the hardware capabilities of the primary guidance subsystem and the SCS. This review resulted in implementation of the integrated G&C concept. Additionally, the backup or abort guidance requirements were further established as being more complex than originally envisioned (ref. 1).

The LM G&C subsystem provides two paths or subsystems for vehicle G&C. The primary guidance, navigation, and control subsystem (PGNCS) provides the necessary G&C capability for mission completion. The abort guidance subsystem (AGS) provides the necessary G&C capability for mission abort if the PGNCS fails but does not provide the capability to complete a lunar landing mission. The SCS forms an integral part of both the primary and abort subsystems. As part of the primary system, the SCS includes the drivers for reaction control subsystem (RCS) jet operation; the electronic interface for descent-engine thrust and gimbal control; and the hand controllers for manual attitude, descent-thrust, and translational input commands. In the AGS, the SCS includes jet-select logic, signal summing, gain control, and the same hand controllers used for manual commands in the PGNCS. The attitude reference or steering errors are provided to the SCS by the AGS.

As an aid to the reader, where necessary the original units of measure have been converted to the equivalent value in the Systeme International d'Unités (SI). The SI units are written first, and the original units are written parenthetically thereafter.

FUNCTIONAL DESCRIPTION

The major features of both the PGNCS and AGS modes are summarized in the following brief functional description of the SCS. The LM SCS is part of the LM guidance, navigation, and control system and is designed to control vehicle attitude and translation about or along all axes during an LM mission. Vehicle attitude and small translations are controlled by firing one or more of the 16 RCS jets mounted on the ascent stage. Major translations are accomplished by means of the ascent or descent propulsion engine. Rotation around the LM X-axis, Y-axis, and Z-axis is termed yaw, pitch, and roll, respectively. Movement along an axis is termed translation. The LM axes form a right-handed orthogonal triad.

The LM guidance, navigation, and control system contains two independent guidance sections. The PGNCS is designed to control the LM during all mission phases. The AGS is designed to guide the LM to the command and service module if the PGNCS fails. A simplified block diagram of the SCS is shown in figure 1. The SCS flight hardware is shown in figure 2.

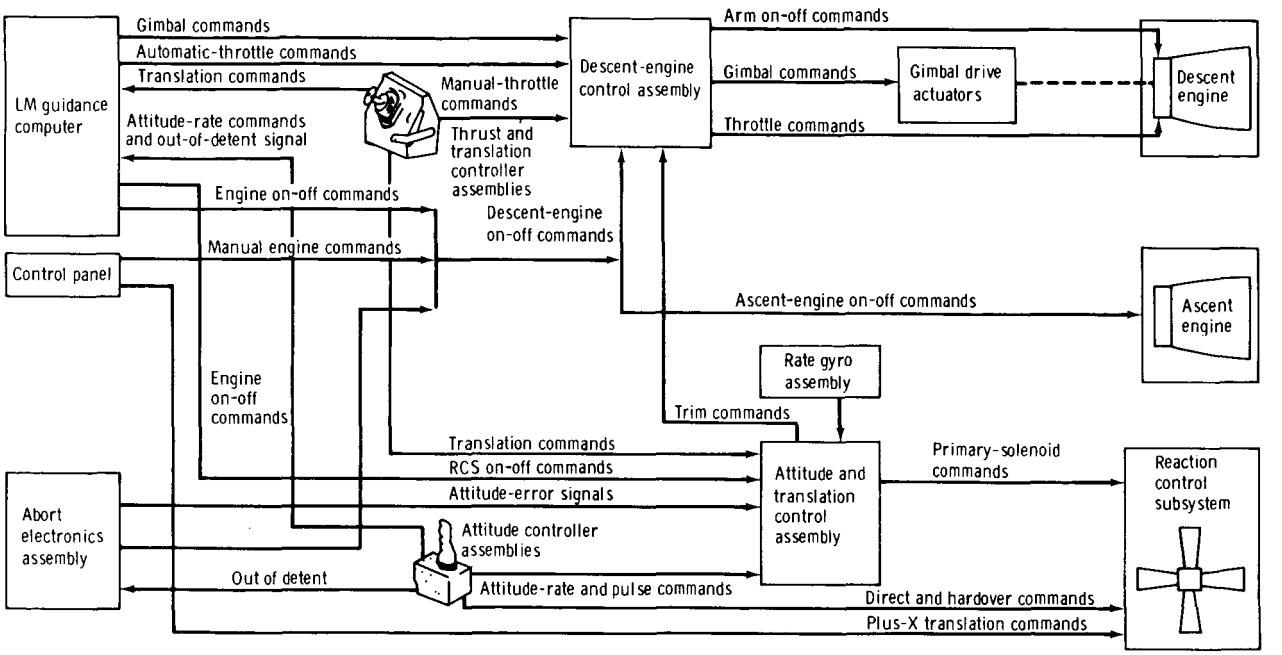

Figure 1.- Block diagram of LM SCS.

Primary Mode

When operating in the PGNCS mode, the SCS performs the following functions.

1. Converts RCS jet commands issued by the LM guidance computer (LGC) to the electrical power required to operate the RCS-jet solenoid valves for attitude and translation control

2. Accepts discrete (on-off) descent-engine gimbal commands from the LGC (Upon receipt of an "on" command, the descent engine is gimbaled about its axes at a constant angular rate until the command is removed.)

3. Accepts LGC automatic and manual engine on-off commands and routes them to the propulsion system to start or stop the descent or ascent engine

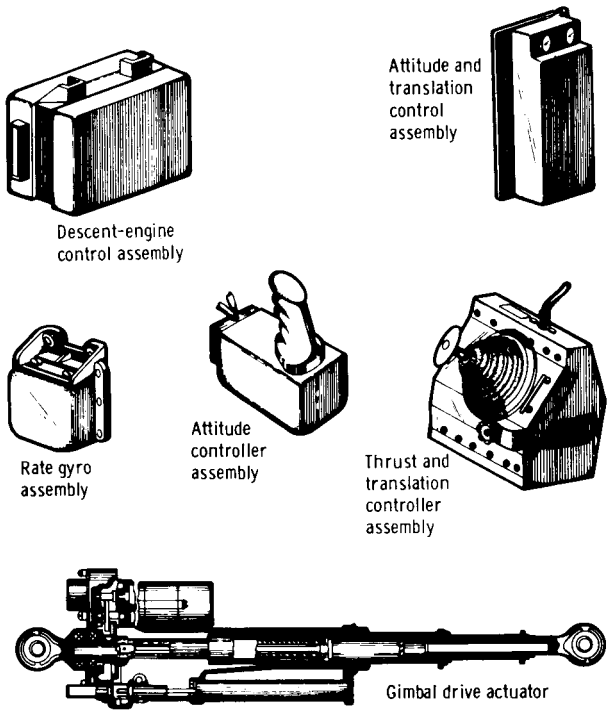

Figure 2.- Lunar module SCS hardware.

4. Accepts LGC automatic thrust commands and thrust and translation controller assembly (TTCA) manual thrust commands to control the thrust of the descent engine

5. Provides manual attitude and translational commands to the LGC

The LM digital autopilot (DAP) has interfaces with 16 RCS solenoid driver preamplifiers in the attitude and translation control assembly (ATCA) and with the pitch- and roll-trim gimbal servomotor drives in the descent-engine control assembly (DECA). During the descent-engine burn, the DAP tries to control the spacecraft attitude with the trim gimbal servomotor drive to save RCS propellants; however, if the rather slow trim gimbal servomotor drive does not keep the attitude error within specified bounds, the RCS jets are used until the error is brought within the attitude dead band, and control is then returned to the trim gimbal servomotor drive.

The DAP works in conjunction either with a PGNCS guidance loop to provide an automatic G&C system or with the TTCA and the attitude controller assembly (ACA) for manual G&C. In the latter mode of operation, the DAP provides the crewman with an integrated stabilization and control system for performing translational maneuvers and for maintaining rate command/attitude hold.

The DAP monitors the eight RCS thruster-off signals and chooses the best set of jets to use under the combined conditions of rotational commands, translational commands, and disabled jets. Propellant economy, minimization of the number of RCS jet firings, and operation with detected and undetected jet failures are of primary concern. The LGC will issue automatic engine on, engine off, descent-engine thrust magnitude, and descent-engine gimbal position commands, all automatically under program control.

The rate-command/attitude-hold mode is the normal means of astronaut control of the spacecraft. The maximum maneuver rate about any axis of the spacecraft is 20 deg/sec. The ACA acts as an analog device during rate command by producing a voltage proportional to stick deflection. The voltage, which represents

commanded rate, is converted to a binary number (quantized at approximately 0.625 deg/sec/bit) and presented to the DAP. When the stick is out of detent, the DAP tries to match the vehicle rate to the rate commanded by the hand controller. When the rate error is less than the rate dead band (0.4 deg/sec during descent, 1.0 deg/sec during ascent), the jets are no longer commanded on. However, if the rate error exceeds a specified bound (2.0 deg/sec during ascent, 1.4 deg/sec during descent), four jets are used to torque the spacecraft. When the ACA is returned to the detent position, the DAP computes the time of jet firing required to zero the rates of the vehicle. When the rate of the vehicle is brought inside a dead band about zero rate, the contents of the coupling data unit (CDU) registers are transferred to the desired CDU registers, and attitude steering about the newly attained position of the vehicle is begun. If the spacecraft has large pitch- and roll-rate errors, diagonal jets are used; the jets are thus selected efficiently.

The crewman uses the minimum-impulse mode to control the spacecraft with very small rate maneuvers. Each discrete deflection of the ACA 2.5° or more out of detent will cause the DAP to issue commands to the appropriate jets for a minimum impulse. In this mode, an astronaut must anticipate on his own and perform rate damping and attitude steering.

Selection of the minimum-impulse mode enables the crewman to perform an economical low-rate maneuver to a new orientation of the spacecraft. After completing the maneuver, the crewman codes the display and keyboard, and the DAP returns to attitude hold and causes the vehicle to limit cycle, with the normal attitude steering about the new orientation.

Abort Mode

When operating in the abort guidance mode, the SCS performs the following functions.

1. Accepts attitude-error signals from the AGS or manual attitude-rate commands from the ACA and fires the proper RCS jets to achieve attitude control (Rate damping is achieved by summing the error signal or the rate command with the rate gyro feedback signals.)

2. Accepts manual translational commands from the TTCA and fires RCS jets to accelerate the LM in the desired direction

3. Automatically gimbals the descent engine for trim control

4. Accepts AGS automatic and manual engine on-off commands and routes them to the descent or ascent engine

5. Accepts TTCA manual-throttle commands to control the thrust of the descent engine

The abort system provides the same basic autopilot modes provided by the DAP; that is, automatic, rate command/attitude hold, and pulse. In these modes, the ATCA (rather than the LGC) receives signals from the TTCA, the ACA, and the AGS (steering errors or attitude reference) and provides the necessary jet-select logic for jet firing. In the pulse mode, a series of minimum-impulse jet firings is provided when the ACA is deflected 2.5°. This series of pulses contrasts with the single pulse used for this mode in the primary system.

SUBSYSTEM DESIGN AND DEVELOPMENT

The discussion of subsystem design and development includes subsystem requirements definition, assembly design and development, and subsystem verification tests.

Subsystem Requirements Definition

The major elements of the subsystem functional and performance requirements were developed from 1963 to 1965. Periodic meetings were held at the NASA Lyndon B. Johnson Space Center (JSC) (formerly the Manned Spacecraft Center (MSC)) during the requirements definition phase. Interested personnel from MSC, NASA Headquarters, and three contractors participated in the meetings, which were helpful in bringing different viewpoints into focus regarding the establishment of detailed requirements concurrently with hardware design. Before the integrated G&C concept was adopted, the more significant results of these meetings were recorded in the proceedings of the LM SCS meetings for the period from 1963 to early 1964. The implementation of the integrated concept began in September 1964 and is documented in the LM G&C implementation meeting minutes.

Assembly Design and Development

Hardware procurement by the LM contractor was at the black-box or assembly level. Individual design-control specifications were prepared for each assembly by the LM contractor, and competitive bids were used for vendor selection. The task of assembly integration was retained by the contractor. Overall hardware-design and development-test requirements are defined in a Type I contract specification. In this specification, broad subsystem functional and interface requirements are stated; the scope of the developmental and qualification test programs is established; and applicable publications, specifications, and standards are defined.

The requirements in this specification are reflected and interpreted as appropriate in the individual-assembly design-control specifications, which are Type II documents.

The same general pattern was followed in the development of each assembly. Developmental tests performed to provide data for use in the design (or in the support of the design) are categorized as design-feasibility tests and design-verification tests. The feasibility tests were performed to select components and parts, to investigate the performance of breadboard or preproduction components or subassemblies under various environmental conditions, to select materials, and to substantiate safety margins or other analytical assumptions. The verification tests were performed to verify the capability of the design to meet the end-use performance and environmental requirements. These tasks were performed on early production equipment (flight packaged but not of flight quality) under selected critical environments and provided the necessary confidence that the equipment design would pass the formal qualification tests.

The formal qualification tests were performed at the assembly level on two production units. One of these units was subjected to sequential, singly applied environments at design-limit conditions (design-limit test); the other was subjected to one operational cycle and one subsequent mission cycle at nominal mission conditions (endurance test). The environments and levels were defined in each of the design-control specifications as well as in the appropriate certification test requirement (CTR). It should be noted that the qualification tests were performed at the assembly level and not at the subsystem level.

The LM test article 3 (LTA-3) structural vehicle was used to gather vibration data from which vibration requirements were generated for each assembly represented by a mass model during vibration tests. The LTA-5 vehicle was used to provide further data on descent-engine and RCS vibration environments. Data obtained from the LM-1 flight test were used to refine the vibration requirements further. The vibration environments used in the assembly qualification tests were defined in Apollo Spacecraft Program Office memorandums.

An acceptance test was performed by the vendor on each production assembly to demonstrate that the assembly satisfied all applicable requirements in the design-control specification. Initially, the plans for acceptance testing included tests under both vibration and thermal-vacuum environments, but these environmental tests were eliminated for most assemblies as an economy measure. However, testing under these environments was reinstituted in late 1967 and in 1968 (contract-change authorizations 726 and 1108), and many assemblies already delivered to the LM contractor had to be returned to the vendor for reacceptance. Each assembly was also subjected to a preinstallation test at the contractor facility upon receipt from the vendor and before installation in a vehicle. This test was performed at ambient environmental conditions. Some of the more pertinent features of individual-assembly development are described in the following paragraphs.

Attitude and translation control assembly.- The ATCA was provided by a vendor under LM contractor purchase order (P.O.) 2-24470-c. The ATCA is composed of four output subassemblies, three analog subassemblies, one power supply, one wiring subassembly, the assembly chassis and cover, electrical connectors, and one elapsed-time indicator. The complete assembly was specified to have a maximum weight of 12 kilograms (27 pounds). The ATCA was mounted on cooling rails in the aft equipment bay of the LM ascent stage.

The ATCA assembly-level developmental tests were performed on two units. Design-feasibility tests were performed on breadboard serial number (S/N) 001S, and design-verification tests were performed on preproduction unit S/N 004, part number (P/N) LSC-300-140-5. During design-verification testing, one failure occurred that required redesign. This failure consisted of a radiofrequency interference at 120 megahertz and its harmonics. The problem was resolved by adding decoupling capacitors at the pulse-ratio modulator. One additional failure was resolved by changing the specification with regard to the distortion of the 800-hertz power-supply voltages that occurred at 150 hertz during audio-conducted susceptibility testing.

Two production assemblies (P/N LSC-300-140-7, S/N 011 and 013) were used in the qualification test program. Endurance qualification tests were performed on unit S/N 011, and design-limit tests were performed on unit S/N 013. The test requirements are contained in CTR LCQ-300-007, dated November 30, 1966, and revision A, dated August 24, 1967.

No significant problems were encountered during qualification testing. However, a postqualification inspection of unit S/N 011 revealed the presence of cracks in some of the solder joints in the interconnection boards used to electrically and physically interconnect a number of cordwood modules. The modules contain copper bus wires that are soldered to the top of the interconnection board; the resulting space between the cordwood and the board is filled with urethane. The cracked joints were caused by mechanical stress resulting from temperature expansion of the urethane filler. The solution to this problem was to strengthen the joint by making it a reflowed convex joint. Because of schedule and cost considerations, a design change to provide stress relief was not made. Instead, all flight attitude and translation control assemblies were modified.

An ATCA design change became necessary after the qualification had been completed. The change was made to eliminate a high-rate-limit-cycle (nonminimum impulse) condition for lightweight ascent conditions. The high-rate limit cycle was caused by marginal control-loop dynamics resulting from a slight change in the RCS thruster characteristics. The ATCA change consisted of altering the pulse-ratio modulator nonlinearity parameter from a value of 0.1 to 0.3. This change was qualified on ATCA unit S/N 013, after modification, to the requirements of CTR LCQ-300-029, dated July 10, 1968. The ATCA change was effective for LM-4 and subsequent lunar modules.

Descent-engine control assembly.- The DECA was provided by a vendor under LM contractor P.O. 2-24486-c. The DECA contains approximately 1000 parts; 95 percent of these parts are packaged as functional entities in 27 cordwood assemblies, and the remaining parts are mounted directly to the chassis. Functionally, the DECA is made up of two trim-error subassemblies, two trim-error malfunction-detection subassemblies, one automatic-throttle subassembly, one manual-throttle subassembly, one power-switching subassembly, one elapsed-time indicator, two electrical connectors, and the chassis. However, these subassembly functions are not packaged as separate, replaceable, or interchangeable subassemblies. The complete assembly weight was specified to be no greater than 3.33 kilograms (7.35 pounds), and the assembly was mounted on the LM descent stage. No cold-plate cooling was required.

The DECA assembly-level developmental tests were performed on two units. Design-feasibility tests were performed on breadboard model S/N 001, and design-verification tests were performed on preproduction model S/N 004. No significant problems discovered in design-verification tests required redesign.

Two production assemblies were used in the DECA qualification test program. Unit S/N 009 (P/N LSC-300-130-5) was used for endurance testing, and unit S/N 012 (P/N LSC-300-130-9) was used for design-limit testing. The part-number differences resulted from a design change made to the trim-fail circuits during the LM critical-design review before the beginning of qualification testing. In the S/N 009 configuration, an automatic trim shutdown was provided for use if the fail circuits detected a failure. In the S/N 012 configuration, a manual disable capability was provided instead of the automatic shutdown. All flight vehicles following LM-1 were equipped with a DECA having the manual disable capability. The qualification test requirements are stated in CTR LCQ-300-006, dated April 23, 1966. No problems encountered during qualification testing necessitated a design change.

A delta-qualification test was performed on retrofitted DECA unit S/N 012 after the previously mentioned DECA testing had been completed. The delta-qualification test was performed to qualify two DECA design changes that had been made because of a requirement for vibration of higher level than that specified by CTR LCQ-300-006. The two design changes were made as a result of problems encountered during vehicle-level tests on LM-1. One change was the addition of a coincident-pulse-detector circuit in the automatic-throttle counter circuit. This change was made to prevent complementing of the counter output caused by simultaneous receipt of thrust-increase and thrust-decrease pulses from the LGC. The other change was made to lower the 4.3-volt direct-current power-fail-monitor threshold. This change was made so that voltage drops in the vehicle cabling that supplies 4.3 volts to the DECA would not result in erroneous operation of the power-fail monitor. The vibration levels were increased as a result of LTA-3 tests, which

showed the DECA vibration levels to be higher than those used in the original qualification testing. The added qualification test requirements are defined in CTR LCQ-300-009, dated June 5, 1967.

Rate gyro assembly.- The rate gyro assembly (RGA) was provided by a supplier under LM contractor P.O. 2-24465. The RGA consists of three subminiature rate gyros (located so that their sensitive axes form an orthogonal triad), a support and insulator assembly, a component board and connect assembly, and an elapsed-time indicator. The gyro is a conventional aircraft-type rate gyro modified to accept 800-hertz power for compatibility with the primary guidance power frequency. The maximum specified weight of the assembly was 0.9 kilogram (2 pounds).

The RGA developmental tests were performed on six assembly-level hardware items. Feasibility tests were performed on two of three units in addition to vibration testing of a mass model and materials and components testing. Design-verification tests and critical-environment tests were performed on four assemblies. No major design deficiencies were encountered during these tests.

Two production assemblies subjected to qualification testing successfully fulfilled all requirements; no major problems were encountered. The test requirements are defined in CTR LCQ-300-008, dated April 25, 1966. After test completion, vibration data from LTA-3 tests revealed higher vibration levels than those used in qualification. Revision A of CTR LCQ-300-008, dated May 25, 1967, reflects these increased levels. Revised vibration tests at these higher levels were performed successfully on one of the RGA qualification units.

An RGA problem that occurred on the Apollo 10 mission (LM-4) apparently was caused by static friction (stiction). The yaw rate gyro "hung up" for approximately 40 seconds. Subsequent analysis indicated that contamination was the most likely cause of the problem and that subjection of the gyro to a questionable rework process during manufacture probably contributed to its contamination. Because the manufacture of all gyros had been completed, a stiction test was instituted to screen gyros with potential sticking problems. Another RGA problem, discovered during the LM-6 checkout at the NASA John F. Kennedy Space Center (KSC), was evidenced by a lack of gyromotor synchronization. Although the LM-6 unit had operated properly during previous RGA acceptance and vehicle tests, an analysis of the unit revealed a deficient gyromotor hysteresis ring. A low-voltage-margin test was instituted to screen flight units for this deficiency.

The low-voltage-margin test seems to be an effective screen for marginal motor characteristics. However, the stiction test is not believed to be effective in detecting all contaminated units. A stripdown and particle count performed

on three gyros revealed counts varying from approximately 400 to 1300 and sizes ranging from 0.025 to 0.889 millimeter (0.001 to 0.035 inch). Previous stiction tests on these gyros indicated a slight stiction on two and no stiction on the third.

Because the stiction screening test was not effective in detecting all contaminated units, a plan was developed to reduce gyro contamination levels. This plan required that the gyros be returned to the vendor for rework. In rework, the gyros were disassembled and internal cavities and dead spaces sealed off to enhance gyro cleaning. The gyros then were flushed to specified contamination levels and refilled under white-room conditions with filtered fluid. The stiction screening test was continued following the cleaning rework. No further stiction problems were encountered either in screening or in flight.

Attitude controller assembly.- The ACA was provided by a vendor under LM contractor P.O. 3-50010. The principal components of the ACA are three linear-output transducers, one three-axis rotational control mechanism, a housing and grip, two electrical connectors, and 43 switches. The specified maximum permissible weight, including connectors and cables, was 2.15 kilograms (4.75 pounds).

Four attitude controller assemblies were used in the development-test program, which consisted of two phases. Phase I tests were performed on two preprototype assemblies fabricated in the model shop. The primary objective of Phase I testing was to obtain enough test data to establish an acceptable prototype design. The Phase II development tests were performed on a prototype controller; a second unit was available as a backup. The primary objective of these tests was to provide hardware-design verification in preparation for a complete mission simulation. The developmental testing did not reveal any major design problems. However, some test discrepancies required design or procedural changes (or both). For example, one procedural change required that the switch packages be subjected to high temperature (339 K (150° F)) during ACA buildup to ensure correct switch overtravel adjustment. This particular change was not effective in ensuring correct overtravel adjustment because a high percentage of attitude controller assemblies exhibited improper switch adjustment when thermal-vacuum acceptance testing was later imposed as an acceptance requirement. This problem is discussed in the following paragraphs.

Two production assemblies were subjected to the qualification tests, as required by CTR LCQ-300-001, revision B, dated November 22, 1966. Both units successfully completed testing without any significant discrepancies. A problem was encountered at low temperature (255 K (0° F)) with the pitch-axis torque hysteresis, which was below the required 60-percent minimum value. It was determined that at low temperature, the minimum hysteresis level in the pitch axis could be changed to 50 percent without serious effect on ACA use. Therefore, the requirement was changed, and a hardware-design change was not necessary.

A switch-actuation problem discovered during LTA-8 thermal-vacuum tests at MSC was caused by improper switch adjustment, or turn-in, during ACA final assembly and calibration; improper adjustment prevented switch actuation under thermal or vacuum conditions (or both). This problem was discovered just as the thermal-vacuum acceptance test requirements of contract-change authorization (CCA) 1108 were being imposed on ACA acceptance. Because the manufacture of all assemblies had been completed at the time of CCA 1108 implementation, it was necessary to return the controllers to the vendor for reacceptance. Many of the controllers examined contained improperly adjusted switches that did not actuate during thermal and thermal-vacuum conditions. This problem was resolved by reworking all flight attitude controller assemblies to newly developed switch-calibration procedures and by implementing more effective quality inspections. The recalibrated controllers were used on LM-3 and subsequent lunar modules.

Much concern was evident in the program regarding the endurance of the ACA centering spring. This concern was stimulated by the switch-adjustment problem and by a spring failure at the NASA George C. Marshall Space Flight Center (MSFC) on one of the development-model assemblies (with uncontrolled usage) that had been provided to MSFC for laboratory use. The spring failed after an estimated 100 000 actuations. It was estimated that a flight ACA would undergo fewer than 1000 ground and flight actuations. However, because of the difficulty in predicting the actual number of actuations and the actual spring life, a redesigned spring was incorporated when the supply of the existing spares was exhausted in spring-life tests. The redesigned spring was installed only in an ACA that required rework at the vendor; other assemblies used springs of the original design.

Thrust and translation controller assembly.- The TTCA was fabricated in-house by the LM contractor. The TTCA contains a position transducer for commanding proportional throttle signals and 12 switches for providing three-axis translational signals to either the primary or the abort control system. A mode-selection lever is provided for selecting either the throttle or the jet mode. The maximum specified TTCA weight was 2.38 kilograms (5.25 pounds).

The TTCA developmental testing was performed on three controllers. Feasibility tests were performed on one controller, and verification tests were performed on two controllers. No major problems were encountered.

Two production assemblies were subjected to the qualification tests required by CTR LCQ-300-003, revision A, dated November 9, 1966. A total of 13 failures occurred during qualification testing; one design change and two procedural changes resulted. The design change resulted from an excessive force spike (more than 66.7 newtons (15 pounds), compared with 31 newtons (7 pounds) as the maximum permissible) during X-axis cycling in the throttle-mode portion of the integration and checkout test. The high force occurred after 258 cycles; at that time, the soft-stop parts were replaced. After approximately 250 cycles, the high-force spike

occurred again. A failure analysis revealed that the high-force spikes had been caused by a wearing action on the chrome surface of a throttle linkage cam. The spike bungee and the cam were redesigned to reduce cam surface stresses. The redesigned model was subjected to 1500 cycles without degradation or visible wear. This problem has not reappeared on the redesigned units.

A problem occurred during an operational check at ambient conditions when a switch in the endurance-test unit failed to actuate. Results of a subsequent failure analysis showed that the discrepant switch had a release force of zero. Other tests revealed that a maximum reduction of 0.56 newton (2 ounces) in release force could be expected after 2500 to 15 000 cycles. Before this occurrence, the minimum acceptable switch-release force had been specified as 0.56 newton (2 ounces). This problem was resolved by increasing the minimum switch-release force to 1.11 newtons (4 ounces). Another switch-actuation problem was that the switch turn-in was insufficient to accommodate thermal-vacuum conditions. The existing requirement for a turn-in of 30° was not adequate for all environmental conditions and was increased to 45°. The additional turn-in resolved this problem. A tolerance study for the 45° switch adjustment was conducted by the LM contractor.

Gimbal drive actuator.- The gimbal drive actuator (GDA) was provided by a vendor under LM contractor P.O. 2-24478.

As an integral part of the development-test program, the GDA was subjected to a reliability-assurance (stress to failure) test in which the GDA was tested to failure under systematically increasing dynamic and environmental stresses. A failure was described as a deviation of performance from the minimum acceptable operating mode. The failure-mode-prediction analysis provided the basis for selecting critical stresses that were used in the stress-to-failure tests.

A qualification test performed on the GDA prototype model demonstrated that the prototype design satisfied the requirements under environmental conditions imposed in the order of occurrence of the applicable environment during a mission. An acceptance test was performed on each assembly to demonstrate that the assembly satisfied all the requirements of the applicable specification and that it conformed to the applicable approved design.

After the qualification test, the GDA was subjected to a reacceptance vibration test (CCA 726) and a thermal-vacuum test (CCA 1108). Operational failure of one unit during thermal-vacuum testing was attributed to low temperature, not to vacuum. Just before that failure, a GDA failure under ambient pressure and temperature at the LM contractor facility was caused by a reduced airgap between the motor-brake polarizer and the brake armature. As a result of testing (CCA 1108), it was determined that many of the units deviated from the performance requirements. The malfunction of the actuator was attributed principally to the motor. The predominant malfunction was coasting of units caused by failure of the brake to engage

in the alternating-current motor. The failure rate was approximately 50 percent on coast; however, one unit drew excess power. Although coasting of the actuator did not constitute a problem within the actuator, prolonged coasting caused the DECA to indicate a descent-engine trim failure.

The analyses of the Apollo 9 and Apollo 10 missions for a nominal and a faulty GDA brake showed negligible effects. Therefore, it was concluded that a GDA brake failure would not preclude the completion of these missions. However, the acceptability of a false caution-and-warning GDA failure indication resulting from a GDA brake failure was still a matter for consideration.

The vendor was directed to modify the GDA. The redesign affected only the motor area and embodied a constant-drag principle. The redesigned actuator was subjected to a delta-qualification test and an extended-life test. When these tests had been successfully completed, it was determined that the new design would be flown on Apollo 11 and subsequent missions.

After the new actuators had been installed in LM-5 (Apollo 11), a failure occurred - the actuator did not respond to a start command. The failure resulted from a phase-angle difference between the power circuit of the DECA and the GDA motor. A capacitor assembly was added at the DECA/GDA interface to improve the phase-angle relationship.

A review of the design-verification test, the qualification test, the vehicle test, and the mission data supports the fact that the GDA fulfilled all design requirements. The gimbal drive actuators performed all required functions throughout the Apollo 9, 10, and 11 missions. The caution-and-warning light indicating gimbal-trim malfunction was observed on the Apollo 10 mission. The actuation of this caution-and-warning signal was attributed to brake coasting because the GDA performed normally during subsequent use on the Apollo 10 mission.

Subsystem Verification Tests

Because the subsystem has been procured at the black-box or assembly level, none of the subsystem-level verification work was done by the equipment vendors. All subsystem-level testing was accomplished in vehicles and in the LM contractor Full Mission Engineering Simulator/Flight Control Integration (FMES/FCI) Laboratory.

Full Mission Engineering Simulator/Flight Control Integration Laboratory.- The FMES/FCI Laboratory was used to perform preinstallation tests on all flight equipment before it was installed in a vehicle. These tests were performed at ambient environmental conditions at the assembly level only. The FMES/FCI Laboratory

was used earlier in the Apollo Program for developing preinstallation test procedures and for performing assembly-integration tests. The integration tests, which included the first electrical interfacing of individual assemblies, were performed on early breadboard or preproduction equipment to verify interface compatibility and to help establish subsystem-level test tolerances. No major problems were encountered during these tests.

Closed-loop simulation tests were also performed to verify system dynamic characteristics and to supplement analytical evaluations. The previously noted ATCA change (associated with the high-rate limit cycle) was evaluated in a closed-loop simulation and supported an associated analytical assessment.

The LTA-1 vehicle tests.- The LTA-1 vehicle was used for early systems-level testing. In general, the subsystem hardware used was of a preproduction configuration. The testing was discontinued at an early date; no significant hardware or interface problems were discovered during the limited test activities. The tests were useful in developing the vehicle operational checkout procedures.

The LTA-8 thermal-vacuum tests.- The LTA-8 thermal-vacuum tests were performed in the thermal-vacuum facility at MSC. The only problem encountered in the stabilization and control system was associated with the ACA switch-actuation deficiency. This problem was resolved as noted in the subsection entitled "Attitude controller assembly."

The LM-2 drop test.- The LM-2 drop-test program was conducted at MSC to evaluate subsystem integrity after vehicle drops representing worst-case lunar-touchdown conditions. Individual assemblies had been designed to survive landing shocks safely, and this capability had been demonstrated during the assembly-level qualification testing. However, cabling integrity had not been demonstrated, and these tests served to verify that no problems existed in this area of vehicle design.

MISSION EXPERIENCE

All detailed test objectives had been satisfactorily accomplished before the Apollo 11 (LM-5) mission. These objectives, pertaining in whole or in part to the SCS, are given in table I. The problems that occurred during each mission are described in the following paragraphs.

The LM-1 Mission

The LM-1 mission was unmanned and was flown without an operating AGS. Descent-engine burns, ascent-engine burns, and fire-in-the-hole staging were

TABLE I.- DETAILED TEST OBJECTIVES

Number	Objective	Description
P11.1	Descent propulsion system (DPS) gimbal actuators	Verify descent-engine gimbaling response to control
P12.3	AGS/control electronics section (CES) atttitude/ translational control	Demonstrate RCS translational and attitude control of unstaged LM using automatic and manual AGS/CES control
P12.4	AGS delta-velocity capability using DPS	Perform an AGS/CES-controlled DPS burn with a heavy descent stage
S12.8	AGS/CES attitude/translational control	Demonstrate RCS translational and attitude control of the staged LM using automatic and manual AGS/CES control
S12.9	Unmanned AGS-controlled ascent propulsion system (APS) burn	Perform an unmanned AGS-controlled APS burn

accomplished by using the SCS in conjunction with the LM mission programer. Attitude control was accomplished with only the rate-stabilization loop because the AGS normally provides the attitude reference. The main-propulsion burns and staging were performed with the SCS because of an LGC software/propulsion interface incompatibility.

The LM-3 Mission

The LM-3 mission (the first manned LM mission and an Earth-orbital mission) was accomplished with only one SCS discrepancy: a failure indication in the descent trim system. This indication was not an unexpected flight occurrence because the GDA coasting problem, which produces such an erroneous indication, had been experienced with LM-3 during checkout at KSC. (See subsection entitled "Gimbal drive actuator.") An evaluation of this problem in terms of mission effects had been made before the mission, and a decision was made not to replace the actuators with units having the coasting modification. Therefore, all concerned personnel were prepared for this occurrence, and no detrimental mission effects resulted.

The LM-4 Mission

The rate gyro "hangup" problem (previously noted in the subsection entitled "Rate gyro assembly") occurred in lunar orbit of the LM-4 mission during descent staging. The total time of abnormal operation was approximately 40 seconds. This problem was identified during data analysis after the mission. Except for this discrepancy, the subsystem performed normally.

The LM-5 Mission

The first lunar landing mission (LM-5) was accomplished without any known subsystem discrepancy or problem.

CONCLUDING REMARKS

Special care should be devoted to mechanical stresses that a given packaging design may place on solder joints. The solder-crack problems experienced on the attitude and translation control assembly, on other lunar module subsystems, and at the NASA George C. Marshall Space Flight Center indicate the need for such design care before a production commitment is made.

Because the mechanical calibration of some equipment may be affected by environmental factors, such equipment should be verified for acceptance by appropriate environmental tests. The problems experienced with attitude controller assembly switch calibration attest to the value of such tests.

The gyro contamination problem and its resolution indicate the need for special care in the design and fabrication of devices that are sensitive to contamination. In a zero-g environment, there is perhaps a greater tendency for contaminants to migrate from entrapped areas than would be the case in a one-g environment. Hence, special attention should be devoted to the elimination of features that may act as contaminant collection points and subsequent migration sources during zero-g operation.

It is believed that subsystem integration can best be achieved if a single vendor supplies the hardware to subsystem-level requirements. This approach contrasts with the lunar module stabilization and control subsystem procurement, which was at the assembly level with the lunar module contractor retaining integration responsibility. The impedance-mismatch problem between the gimbal drive actuator and the descent-engine control assembly resulted from a minor gimbal drive actuator change made without sufficient understanding of the interface effects. Although subsystem-level procurement is not a panacea for all problems, its application would make this sort of interface problem much easier to avoid than with assembly-level procurement.

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