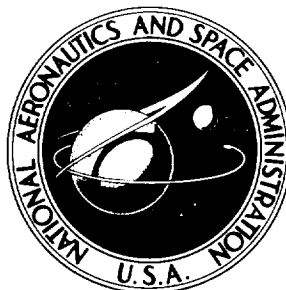


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**APOLLO EXPERIENCE REPORT -
GUIDANCE AND CONTROL SYSTEMS:
COMMAND AND SERVICE MODULE
STABILIZATION AND CONTROL SYSTEM**

by Orval P. Littleton

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • SEPTEMBER 1974

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16. Abstract The concepts, design, development, testing, and flight results of the command and service module stabilization and control system are discussed. The period of time covered is from November 1961 to December 1972. Also included are a functional description of the system, a discussion of the major problems, and recommendations for future programs.					
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APOLLO EXPERIENCE REPORT
GUIDANCE AND CONTROL SYSTEMS:
COMMAND AND SERVICE MODULE STABILIZATION
AND CONTROL SYSTEM

By Orval P. Littleton
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SUMMARY

A description of the stabilization and control system for the Apollo command and service module is presented in this report. The main areas discussed are concepts, design, development, testing, and flight results.

The stabilization and control system requirements initially consisted of three main areas: attitude reference, displays and controls (including automatic thrust-vector control), and reaction control. After extensive simulations, manual thrust-vector control was added. Also, the Block I system was designed to allow for performing onboard maintenance. However, because of humidity problems encountered during Project Mercury and the time required to isolate and correct representative problems, the onboard maintenance capability was deleted in favor of the Block II concept. In lieu of onboard maintenance, the Block II design featured hermetically sealed devices, potted cordwood construction, and redundancy in critical areas (for example, two gyroscope packages; two rotation hand controllers; two flight director attitude indicators; and redundant translation, abort, thrust on/off, and thrust-vector control paths).

The stabilization and control system performed satisfactorily on the 4 unmanned and 11 manned flights of the Apollo Program. This performance can be attributed to conservative design techniques; to the use of state-of-the-art components; to extensive developmental, qualification, and acceptance testing; to extensive vehicle checkout at prime-contractor facilities and at the NASA John F. Kennedy Space Center; and to various simulation programs at the NASA Lyndon B. Johnson Space Center (formerly the Manned Spacecraft Center) and at contractor facilities. Recommendations for future programs are given.

INTRODUCTION

Initially, the command and service module (CSM) stabilization and control system (SCS) was to provide the capability for controlling spacecraft rotation and translation, an attitude reference system (ARS), and the displays necessary to allow the astronauts to monitor automatic operation as well as to exercise optimum control of the spacecraft during manual maneuvers. A primary design requirement was that no single failure would result in the loss of the spacecraft or the flight crew. Because of the predicted high accelerations from the service propulsion system (SPS) engine, manual thrust-vector control (MTVC) was not considered feasible and thus was not required. Extensive simulations subsequently showed that MTVC could be used effectively by the astronauts, however, and MTVC became the primary backup mode used in the event of a failure in the primary guidance, navigation, and control system (PGNCS).

From early in the program (November 1961) until the major redesign in June 1964, the SCS was the primary method of flight control; that is, the SCS was in series with the guidance and navigation (G&N) system (fig. 1). As the primary method of flight control, the system had to be highly reliable to meet the mission requirements. Despite the efforts of reliability and design engineers, the equipment mean time between failure was of the same order of magnitude as the mission time. Hence, the most feasible solution to the problem was to provide standby redundancy in the form of onboard spares.

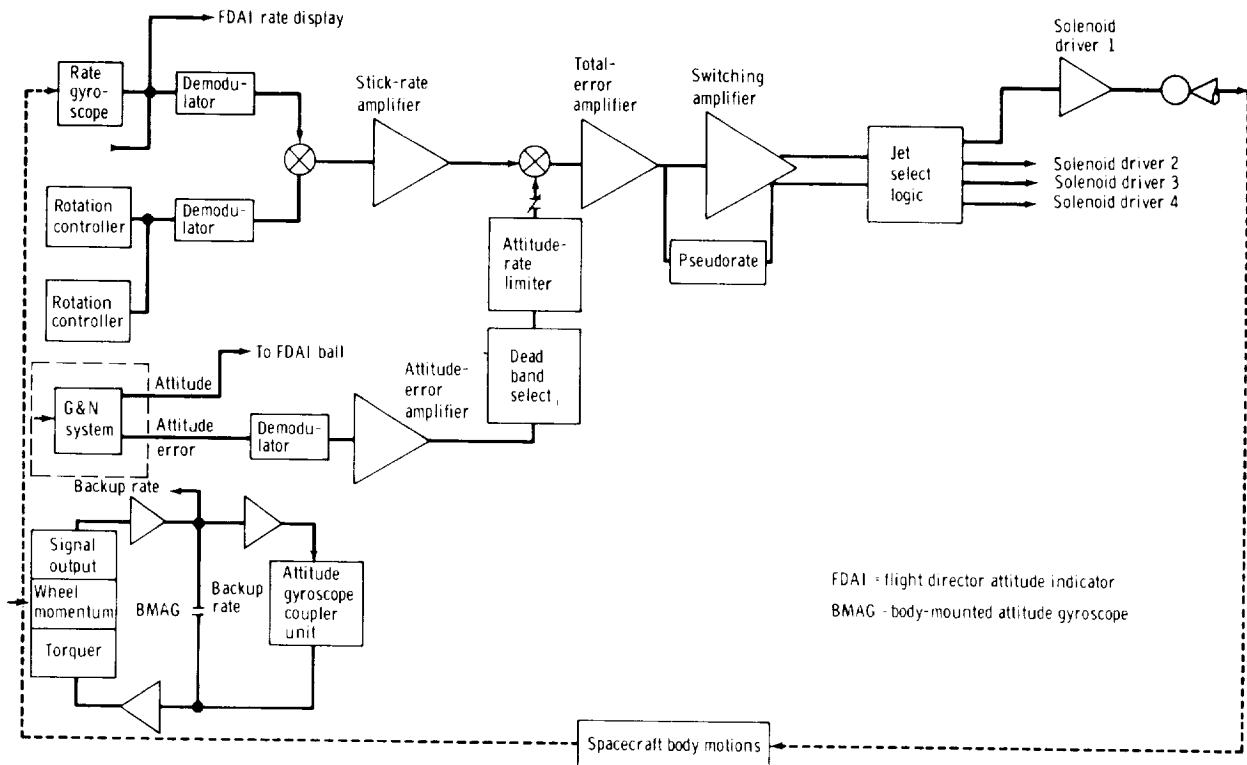


Figure 1. - Diagram of the Block I guidance and navigation attitude control mode.

The SCS was initially planned with complete standby redundancy in the form of plug-in modules or subassembly spares, or both. With the aid of simple checkout equipment, the crewmen would be able to detect malfunctions at the module or sub-assembly level and to make the necessary replacements. The mean time to repair (estimated to be 30 minutes) was not believed to be significant during the midcourse trajectory because attitude-disturbing torques would be small and trajectory corrections could be made after the system was restored to normal operation.

In 1963, problems involving humidity, connectors, single-point failures, and relays led to a new design, called Block II. The details of these problems are discussed later. Definition of the Block II concept resulted in elimination of in-flight maintenance; adoption of hermetic sealing of devices for humidity purposes; use of state-of-the-art connectors; and addition of redundant control paths and, eventually, of special screening tests to improve the reliability of the relays. It was also specified that backup steering would be provided by the SCS in the differential-velocity (ΔV) modes during the transearth injection phase and during the translunar and trans-earth midcourse velocity corrections.

As a backup to the PGNCS, the Block II SCS provides stabilization and control of the spacecraft for rotation, translation, and SPS thrusting (using the CSM reaction control system (RCS) engines and the SPS engine). The Block II SCS also provides the required displays and controls to permit the necessary crew interface with the controlling elements. There are additional interfaces with the PGNCS (fig. 2).

The Block II definition phase covered the period from June 4 to August 18, 1964. This phase was a period of cooperative effort between the prime contractors and NASA to determine the feasibility of the Block II approach, to establish requirements in the interface areas, and to detail the mechanization. The evolution of the SCS design is shown in the time line in figure 3. A discussion of the final design follows.

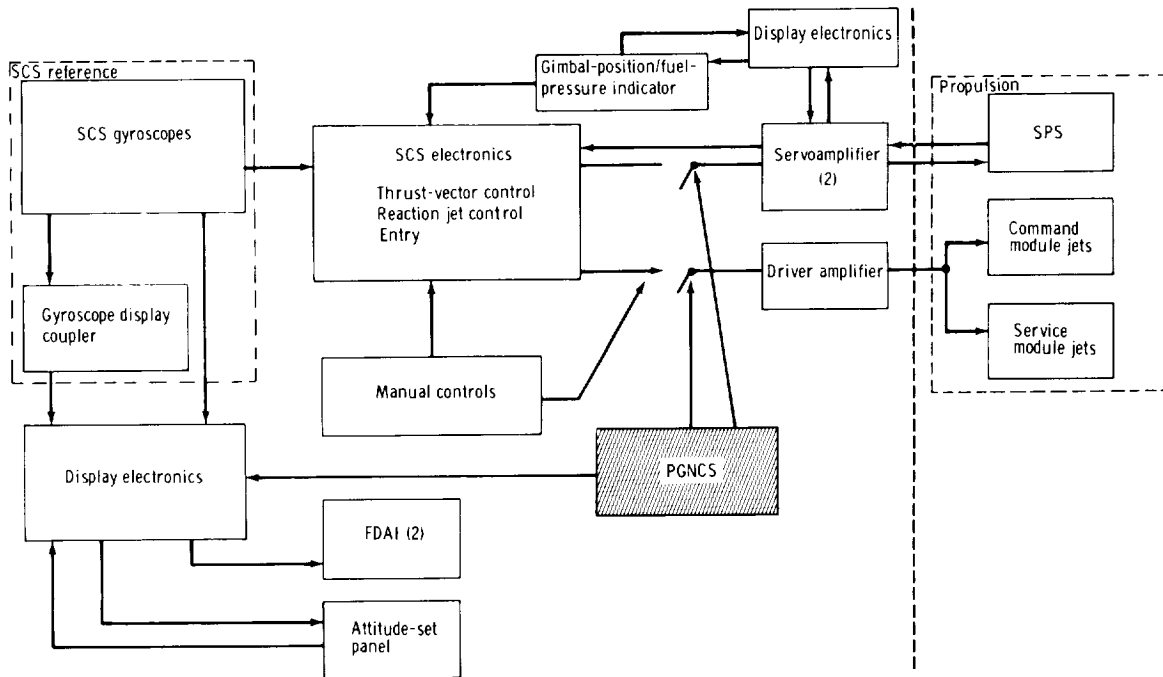


Figure 2. - Functional schematic of the stabilization and control system.

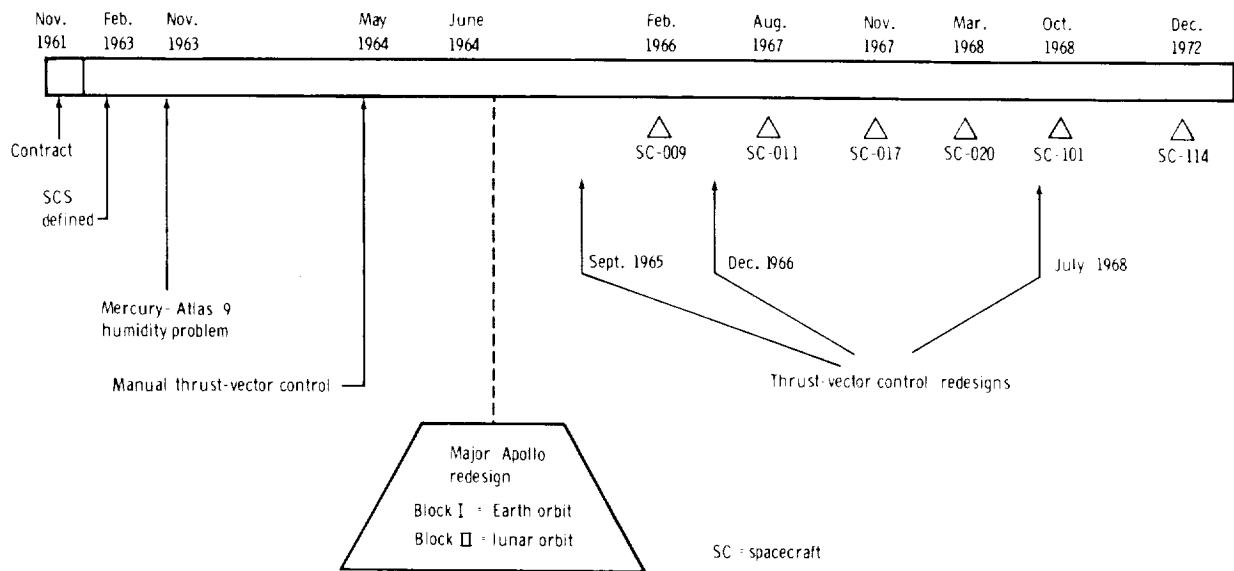


Figure 3. - Evolution of the stabilization and control system.

DISCUSSION

Attitude Control System

The attitude control system (ACS) provides backup attitude flight control capability for the spacecraft during all mission phases and also provides for thrusting of the SPS engine. The ACS can be operated in either the automatic or the manual mode. Propulsive force is provided by 16 hypergolic-propellant, nonthrottleable engines (nominally 445-newton (100 pound) thrust), and low limit-cycle rates are maintained by pseudorate feedback.

The ACS is contained within the SCS and is composed of the electronics necessary to accept command signals from the rotation and translation hand controls, rate and attitude-error signals from the ARS, and configuration switching signals from the main control panel. The ACS causes the CSM RCS jets to fire in such a manner as to control vehicle motion.

The ACS provides the following backup functions in case of a primary control system failure, or in those instances when a shutdown of the primary system is desired.

1. Holds the spacecraft attitude within a selectable dead band, using error signals from the ARS
2. Provides for manual control of spacecraft attitude in the following four configurations: proportional rate commands, on/off acceleration commands to the

RCS solenoid automatic coils, minimum-impulse commands from the rotation hand controller (RHC) to the RCS solenoid automatic coils, and on/off acceleration commands from the RHC direct switches to the RCS solenoid direct coils

3. Provides for manual translation commands in six directions with the translation hand controller (THC)

4. Provides for automatic rate stabilization of the vehicle

5. Provides for manual ullage commands to either the automatic or the direct RCS solenoid coils

6. Provides override logic for compatibility between automatic and manual operation (Astronaut direct commands take precedence and cause the appropriate automatic functions to be inhibited.)

A simplified functional block diagram of the single-axis ACS is given in figure 4. The ACS thruster arrangement is shown in figure 5.

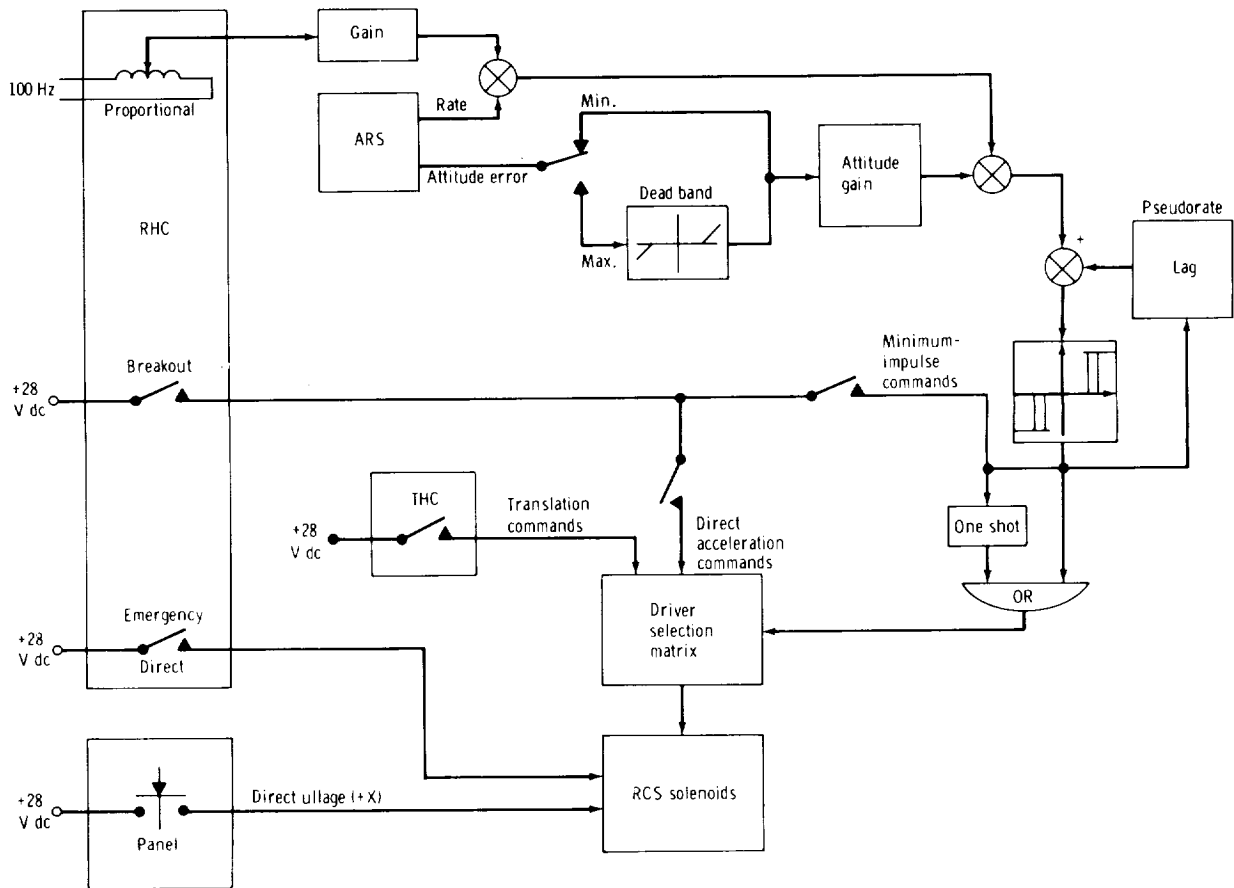


Figure 4. - Functional schematic of the attitude control system.

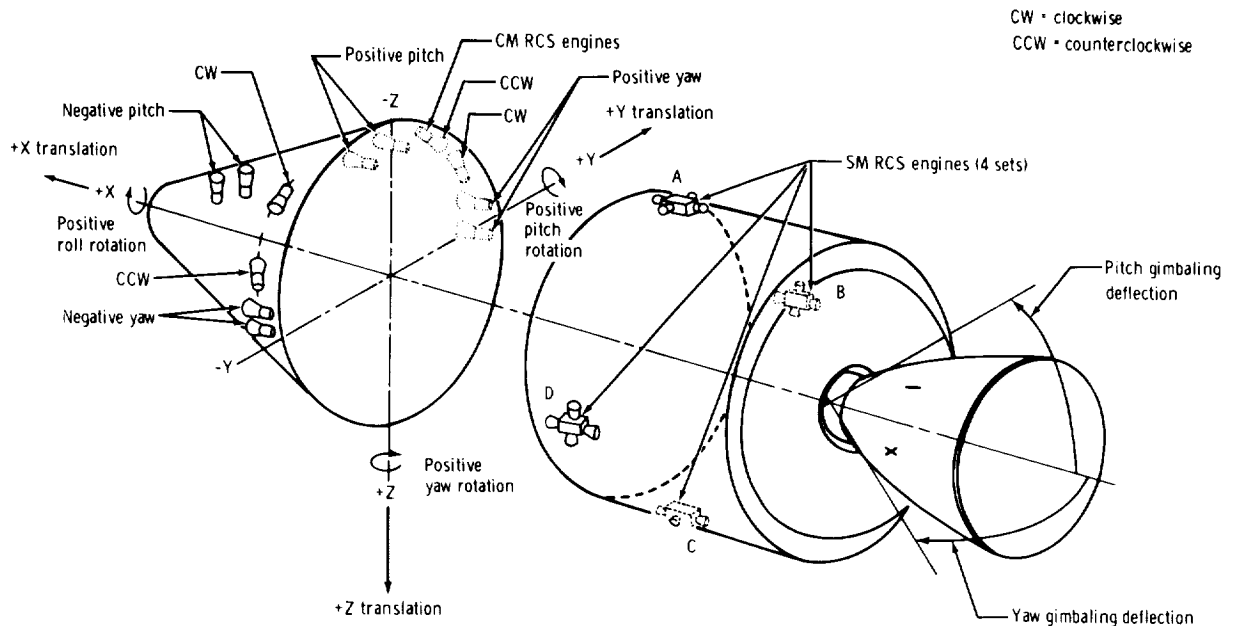


Figure 5. - Thruster arrangement of the attitude control system.

Attitude Reference System

The purpose of the Block II ARS is to provide a backup attitude reference capability for the Apollo spacecraft control functions. The primary attitude reference is provided by the PGNCS. The ARS consists of two sets of three body-mounted attitude gyroscopes, either set of which can be used in the rate mode, and a coupler unit that generates Euler angles. The principal problem with the ARS was the drift rate; the problem was solved by improved design and increased knowledge of this equipment.

In providing a backup attitude reference capability for spacecraft flight control operations, the following functions are performed by the ARS.

1. Provides vehicle attitude errors (in body coordinates) to the vehicle backup control system during the coast and thrusting phases
2. Provides vehicle attitude errors (in body coordinates) for display on attitude-error needles
3. Provides a set of Euler angles, which describes the orientation of the vehicle reference frame with respect to some inertial reference frame, for display on a three-axis attitude ball
4. Provides total roll angle (about the command module (CM) stability axis) for display during entry

5. Provides resolution of small Euler angular errors (treated as vector quantities) into body-axis errors for display on attitude-error needles (These angular errors are the difference between a set of Euler angles that describes the desired vehicle orientation and a set that describes the actual vehicle orientation.)

Controls and Displays System

The controls and displays system (figs. 6 and 7) provides the hand controllers, the flight instruments, and the dial settings necessary for the astronaut to monitor and control the spacecraft during the various flight modes. The principal developmental problem associated with the controls and displays was fraying and fracturing of the cabling for the hand controllers. The design trade-offs among durability, flexibility, and flammability were so difficult that advancement in the design of flexible cabling was the final solution.

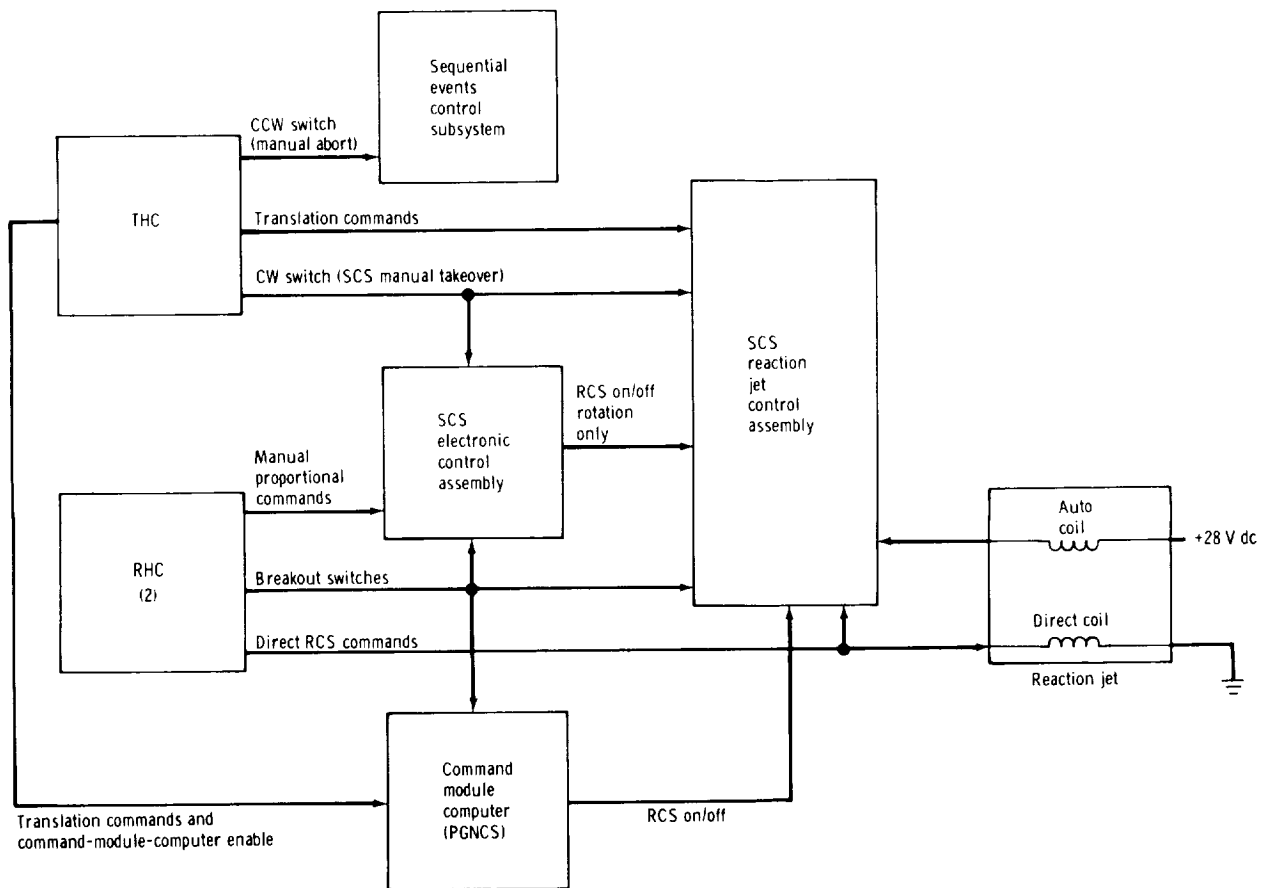


Figure 6. - Block diagram of the controls system.

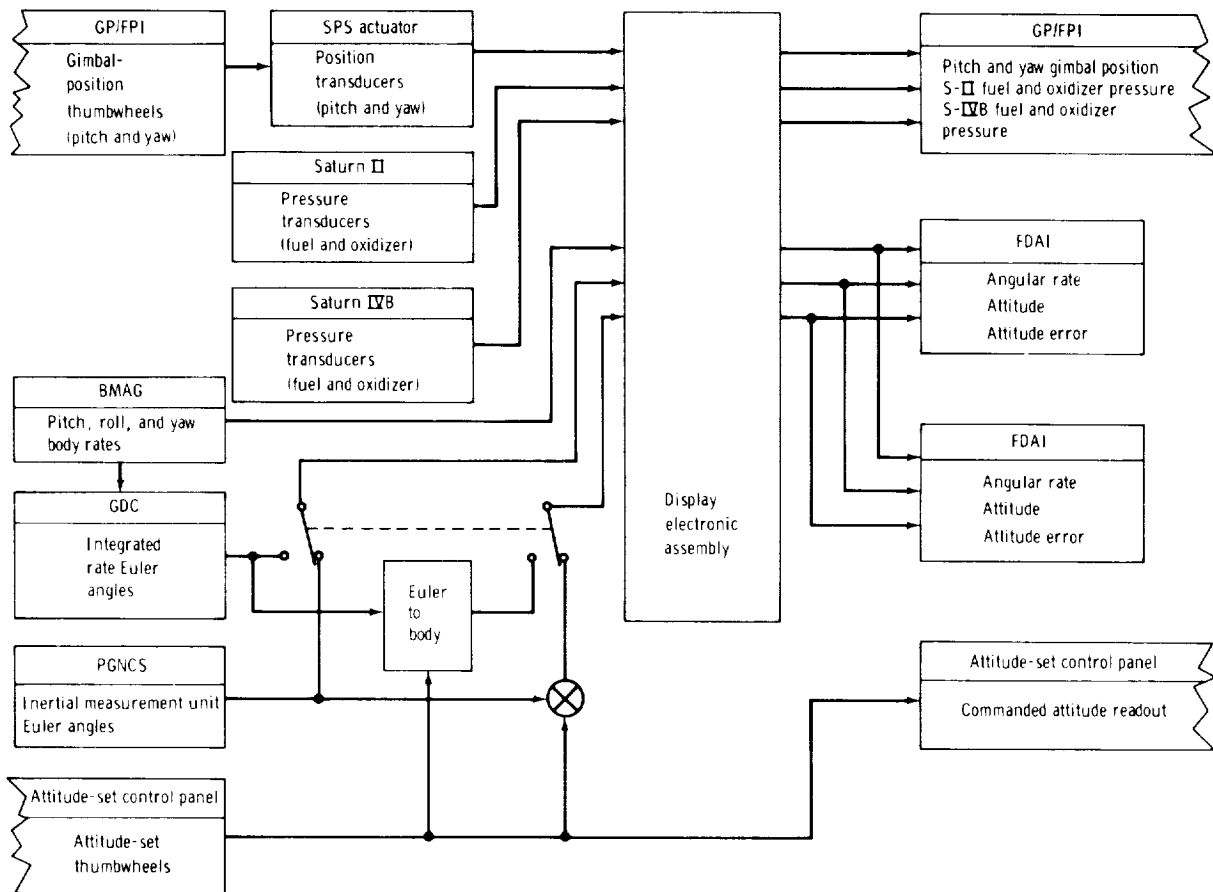


Figure 7. - Block diagram of the displays system.

System description. - The control functions and the display functions of the controls and displays system are described in the following paragraphs.

Control functions: The control functions of the Apollo SCS are provided by the translation controller (fig. 8), the rotation controller (fig. 9), and the attitude-set control panel (ASCP) (fig. 10).

The translation controller (1) provides manual acceleration control of the spacecraft (CSM combination only) rectilinear motion in both directions along the three principal axes, (2) initiates a CSM/Saturn IVB (S-IVB) abort command to the spacecraft mission sequencer by counterclockwise rotation of the controller handle, (3) transfers spacecraft thrust-vector control (TVC) from the PGNCS to the SCS by clockwise rotation of the controller handle, and (4) is capable of simultaneous multi-axis actuation.

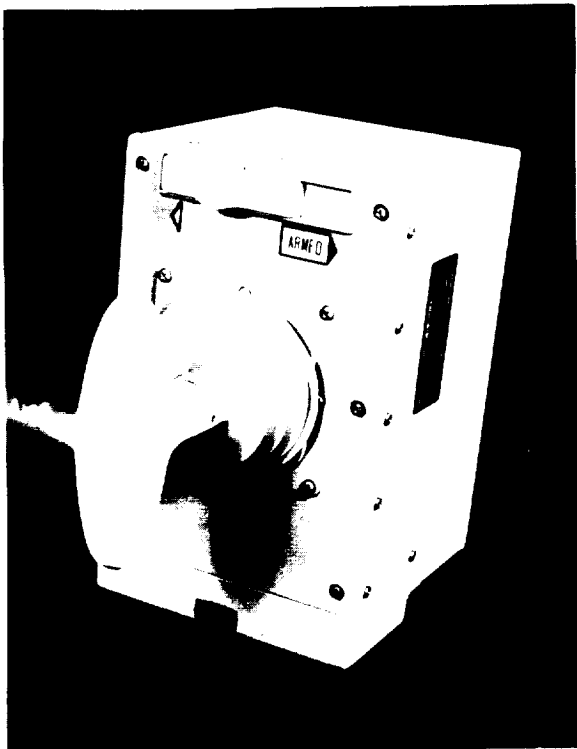


Figure 8. - Translation controller.

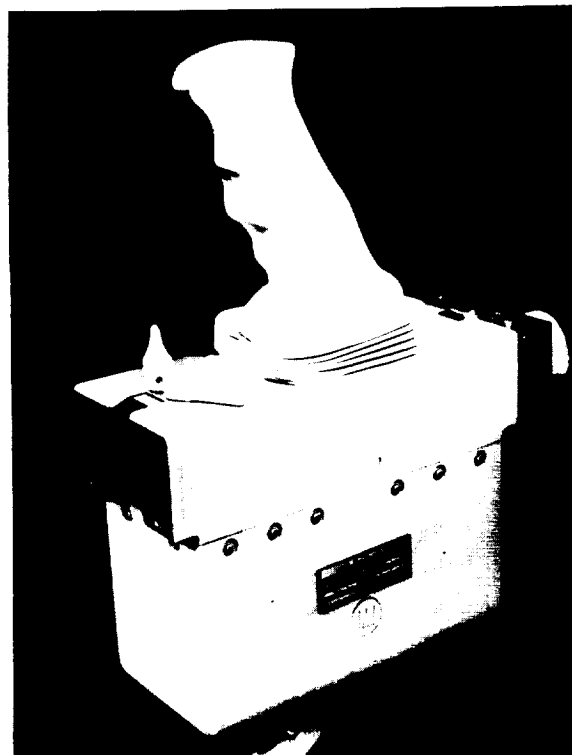


Figure 9. - Rotation controller.

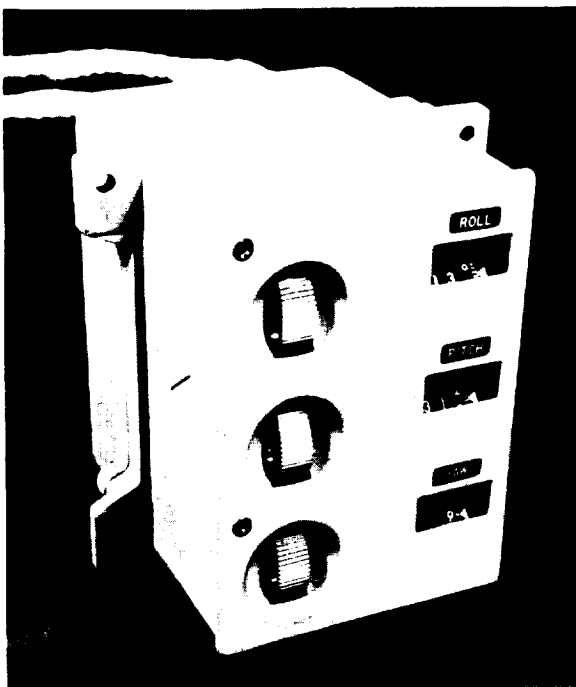


Figure 10. - Attitude-set control panel.

The rotation controller (1) provides three-axis manual proportional body-rate commands or direct angular acceleration (by switch selection) for spacecraft rotational motion in both directions about the three principal axes from breakout switch actuation (1.5° travel from neutral) to the soft stops (10°), (2) provides emergency angular acceleration commands directly to the RCS jet solenoids by controller emergency direct switches (11°), (3) provides manual proportional SPS-engine gimbal-position control in pitch and yaw during MTVC, and (4) is capable of simultaneous multi-axis actuation.

The ASCP provides selectable, three-axis, spacecraft inertial-pointing signals by means of thumbwheel controls and dial readouts. These signals are summed with either the PGNCS inertial measurement unit (IMU) or the SCS gyroscope display coupler (GDC) inertial-attitude signals to produce total-attitude-error signals that

are displayed on the flight director attitude indicator (FDAI) attitude-error needles to allow precise manual rotation maneuvers to the selected inertial angles.

Display functions: The display functions of the Apollo SCS are provided by the FDAI (fig. 11) and the gimbal-position and fuel-pressure indicator (GP/FPI) (fig. 12). The FDAI performs the following functions.

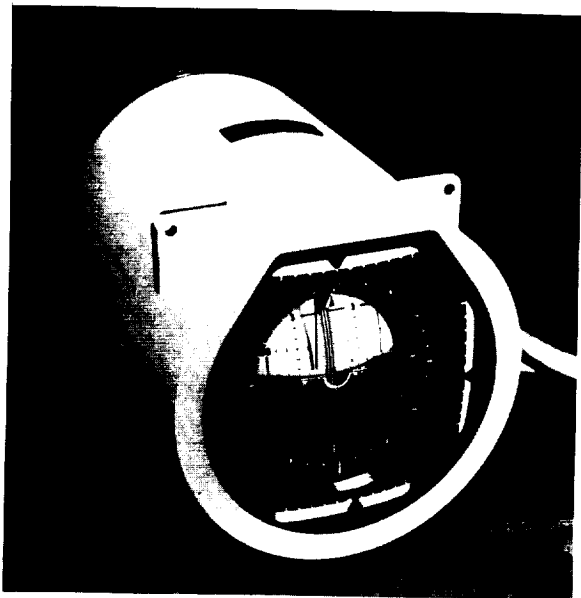


Figure 11. - Flight director attitude indicator.

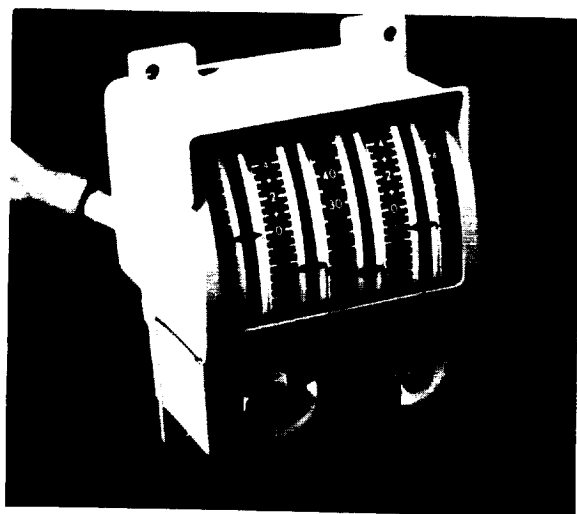


Figure 12. - Gimbal-position and fuel-pressure indicator.

1. Provides an inside-out display of the spacecraft attitude with respect to a selected inertial frame of reference by means of a graduated three-axis ball and appropriate reference indexes

2. Provides a "fly-to" display of the spacecraft angular position relative to an inertial reference in all three axes by means of attitude-error needles

3. Provides a fly-to display of the spacecraft angular rate about each of the three mutually perpendicular spacecraft body axes before the 0.05g level during entry (After 0.05g is reached, the displays provide the angular rate about the spacecraft pitch axis and about the roll and yaw entry axes.)

4. Provides a centralized display of attitudes, vehicle rates, and attitude errors

5. Provides coarse attitude orientation (obtainable with the ball) and fine orientation using attitude-error needles

6. Indicates the IMU maneuver limits and the approach of these limits during maneuvers

7. Provides a monitor and a cross-check of reference-equipment conditions by comparing body-axis rates adjacent to body-axis attitude errors

8. Provides a monitor of the SCS execution of reference-system commands in three axes by three command needles

9. Provides variable scale factors in rate and error indications for varying precision of the maneuvers required during the different mission phases

The GP/FPI (1) displays the Saturn II fuel and oxidizer pressures on one of the sets of redundant meter movements during boost, (2) displays the S-IVB fuel and oxidizer pressures on the other set of redundant meter movements during boost, (3) displays the angular position of the service module (SM) main propulsion engine gimbal about the pitch and yaw axes, one on each of the dual sets of meter movements, and (4) provides for manually inserting initial-condition command signals into the actuator servosystems for both pitch and yaw by means of thumbwheel controls.

Problems. - Listed in the following paragraphs are the major problems that occurred during design and development of the Block II SCS controls and displays.

Flight director attitude indicator: Problems were encountered in the design of the FDAI in the following areas.

1. Spacecraft installation: During the development of the new Block II FDAI having an instrument panel backmounting feature, the need to support a thermal coldplate from the rear face presented a major vibration susceptibility problem. Because the FDAI was cantilevered from the front face, the need arose to provide structural support at the rear face without interfering with the coldplate. The design solution required redesign of the coldplate to incorporate through holes, modification of the FDAI backplate to receive supporting pins, and addition of supporting pins to the secondary structure. The obvious effect of this design was the loss of ready access to both the FDAI and the coldplate for maintenance.

2. Needle vibration: Because of the thinness, the length, and the cantilevered support of the attitude-error needles, redesign was required as a result of vibration testing. These needles were made more rugged to withstand the vibration environment.

3. Electroluminescent (EL) lighting: When the Block II control and display redesign was instituted, the contractor implemented the use of integral EL lighting. The SCS subcontractor experienced difficulties in orienting commercial vendors to aerospace requirements. In addition, problems were encountered with the consistency of EL lamp characteristics, with the reflections and aberrations from nearby surfaces, and with the uniformity of light intensity and color. These problems ultimately were resolved, but the contractor was forced to change the EL power and power factor requirements. It was found that better lamp operation was obtained when the supply voltage was higher than the nominal spacecraft voltage.

4. Lighting control: A Variac transformer is part of the spacecraft wiring so that display lighting intensity can be varied. The current surge resulting from on/off operation of the lighting control loads the alternating-current inverter and, thereby, causes line voltage variations. These voltage variations cause rate-sensor oscillations, which result in inadvertent reaction jet firings. To minimize this effect, a resistor was placed across the Variac transformer, and the spacecraft operational procedures were changed so that the display lights were dimmed but not turned off.

Attitude-set control panel: During vibration testing, the ASCP attitude-set thumbwheels tended to drift slowly. To eliminate this problem, friction-type disks were incorporated into the thumbwheel support mechanism.

Hand controllers: The following problems were encountered with the hand controllers.

1. Cabling: Major problems occurred with the cables from both the RHC and the THC. Because of their exposed positions in the CM cabin, the cables were subjected to excessive wear and tear during checkout. Constant movement and stowage flexed the cables until cracks appeared. Removal of the outer cable covering exposed a Teflon braid that had no abrasion or puncture resistance. When applied to the cables, other nonflammable materials either cracked when flexed or were not sufficiently flexible. Strain relief of the cables at the point of exit from the enclosures was a major problem because the excessive strains actually encountered were not originally envisioned and, consequently, were not specified in the initial design requirements. The subsequent trade-off among material flammability, durability, and flexibility characteristics proved to be extremely difficult. The use of a special covering and the restriction of hand-controller use during the ground test phase solved the problem. An alternate hand controller also was used during this time.

2. Handle shape and force characteristics: The handle shape and the force/deflection characteristics are subjective requirements and were defined only after a long and tedious evolutionary process. The design was changed continually during the middle period of the program, even after hardware commitments had been made. A belated solution was the generation by the NASA Lyndon B. Johnson Space Center (JSC) (formerly the Manned Spacecraft Center (MSC)) of the specifications in this particular area.

Thrust-Vector Control System

The TVC system provides flight control of the Apollo spacecraft during thrusting of the SM SPS. The system provides for both automatic and manual control about all three body axes when the CSM and the lunar module (LM) are docked (CSM/LM configuration) and when the CSM and LM are undocked (CSM-only configuration). A detailed discussion of the TVC system, the design requirements, the problems, and the changes made throughout the Apollo SCS TVC development is presented in the following paragraphs.

System description. - The description of the TVC system includes control modes and functional requirements.

Control modes: The Apollo spacecraft has several modes for controlling powered flight. The modes can be divided into the two major categories of primary and backup. The primary system uses the command module computer. The backup system is composed of analog components and includes automatic and manual control. Control modes for the TVC system are discussed in the following paragraphs. A functional diagram of the TVC system is shown in figure 13.

1. Primary automatic mode: The Apollo Block II primary TVC system does not use the SCS electronics but uses an all-digital controller that sends commands directly to the SPS engine-gimbal servomechanism through a digital to analog converter.

2. Backup automatic mode: In the backup automatic mode (referred to as the SCS ΔV mode), the TVC system uses information from the SPS engine-gimbal servomechanism (position) and from the vehicle attitude and rate gyroscopes to generate and send control signals through the SCS TVC electronics to the SPS engine-gimbal servomechanism. The system does not operate on an attitude command signal but automatically holds the thrust vector fixed in inertial space. This procedure limits the use of the mode to real-time SPS-engine burns.

3. Manual mode: The manual mode of operation (referred to as direct or rate MTVC) provides for commands from the pilot to the SPS engine-gimbal servomechanism by means of a proportional hand controller.

4. Roll axis control: For each of the operational modes just described, the pitch and yaw axes are controlled by the angular motion of the SPS engine. The roll axis, however, is controlled by firing the SM RCS jets. The roll control may be completely automatic, completely manual, or a combination of both.

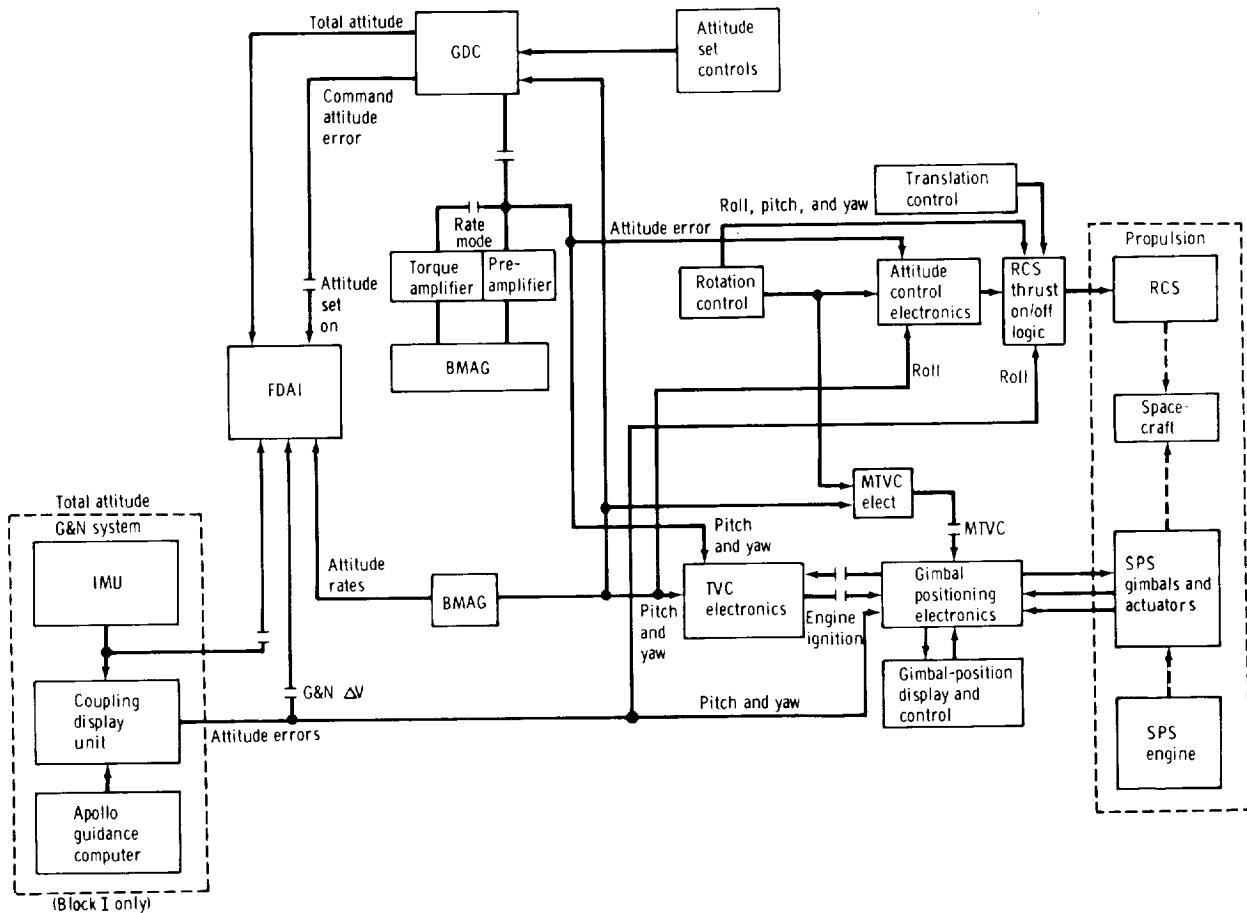


Figure 13. - Functional schematic of the thrust-vector control system.

Functional requirements of the SCS TVC: To fulfill functional requirements, the SCS TVC must provide pointing and stabilization during the thrusting maneuvers, which include midcourse corrections, lunar-orbit insertion, transearth injection, Earth-orbital retrofire, and high-altitude aborts. The SCS TVC also must provide for stable operation of the SPS engine-gimbal servomechanism.

Problems. - In developing a TVC system of the magnitude required for the Apollo Program, many problems were encountered. Some problems were not significant enough to record; only the more significant problems and the manner in which they were resolved are detailed here.

The SPS actuator development: Problems in design of the SPS electromechanical actuator were primarily caused by difficulties in the development of the electromagnetic clutch. The principal problems with the clutch were overheating and nonlinearity. The nonlinear effects, although affecting the vehicle limit cycle and making one actuator appear slightly different from another, did not interfere substantially with the system operation. The primary cause of clutch overheating was thrust misalignment during the SPS burns. The overheating caused a decrease in clutch gain, not only for that burn but also for subsequent burns. This gain decrease could have been as great as 50 percent. However, a new actuator could have had a clutch gain of greater than 50 percent of nominal. As a result, the control system had to be designed to allow for clutch-gain variations of ± 50 percent.

Three design changes were made to the actuator in an attempt to prevent clutch overheating. Overheating was eventually prevented by changing the gearing of the motor to the clutches. The slip speed between the rotating input and the stationary output members of the clutches was less, and the subsequent heat generation was less. The penalty for the design change was that, for a given motor speed, the SPS-engine bell moved more slowly.

Large-signal stability: A direct result of lowering the SPS actuator rate capability was a loss of some large initial-condition or nonlinear stability capabilities. When the Block II system was designed, the large-signal stability problem was resolved by a combination of imposing smaller limits on the SPS gimbal excursion and of lowering the autopilot gains.

Pointing error: As a direct result of lowering the autopilot gains to eliminate the large-signal stability problems, the pointing error requirements were degraded; the degradation, however, was shown to be acceptable for Apollo missions.

Body-bending data variation: The variation in body-bending data was a major problem. The original design was based on the first set of data (1963). In April 1965, NASA deleted modal testing for the CSM and for the CSM/LM. As a result, the autopilot was changed to be less sensitive to body bending in September 1965. At that time, tolerances of ± 15 percent for the bending frequency and ± 10 percent for the mode shape were assumed.

In the spring of 1965, a subcontractor performed a modal analysis by which the structure was separated into the component parts rather than assumed to be a beam.

These data indicated a first-mode frequency 50 percent less than that of the 1963 data. This variation was considerably outside the 15-percent tolerance; however, the data were considered preliminary.

In January 1966, on the first Block I flight spacecraft (spacecraft 009), a TVC system instability occurred during checkout on the launch pad at the NASA John F. Kennedy Space Center. Evaluation of the data indicated that the cause was probably a body-bending resonance at approximately 17 hertz. The bending analysis at that time indicated that in the free-free (flight) mode, the resonance would be in excess of 30 hertz. Although the test was performed on the Saturn IB stack, it was believed that 17 hertz could be indicative of a free-free mode at the CM/SM interface. The Block I TVC system was modified and the instability corrected. However, considerable doubt was cast on the analytically generated bending data. The first two Block I vehicles did not exhibit any in-flight TVC instability, although components of signals at approximately 17 hertz were evident in the rate gyroscope traces of every Block I flight. Shortly thereafter, the modal analysis task was reassigned to another contractor.

In the late summer of 1966, the prime contractor produced another set of bending data. The CSM/LM analysis indicated a first-mode frequency of 1.12 hertz, which was in agreement with previously established data. At that time, notice was given that the Block II SCS was unstable and that modifications, if required, would have to be made promptly if existing hardware schedules were to be met.

Shortly thereafter, a decision was made to redesign the SCS to provide adequate stability margins for all existing sets of bending data. The performance requirements were changed to the final value at that time. This relaxation was required because system performance was degraded when the system gains were reduced to the extent necessary to provide adequate stability for the range of bending data used. The SCS electronic components fabricated were designed to accommodate dominant modal frequencies in the range of 1.0 to 1.6 hertz.

Revised (December 1966) subcontractor stiffness data for the LM were forwarded to the prime contractor from MSC. Using the revised data, frequencies were determined for the CSM/LM half-full propellant-loading condition in April 1967. The first three frequencies were 2.15, 2.6, and 2.9 hertz. Also at that time, subcontractor data were available for the CSM/LM quarter-full propellant-loading condition. The first two bending modes were 2.1 and 2.4 hertz. In the summer of 1968, it was decided to modify several spare electronics boxes to a design based on bending in the 1.75- to 2.86-hertz range. Modal testing in late summer of 1968 indicated bending frequencies of 2.76, 3.02, and 3.87 hertz. A decision to make the new design common to all spacecraft subsequent to spacecraft 103 (Apollo 8) resulted in fabrication of three separate autopilot configurations as flight articles before the first CSM/LM flight. The first CSM/LM flight spacecraft (Apollo 9) was used as the test article for an in-flight margin-stroking test. The results showed the first and second bending modes to be approximately 2.76 and 3.1 hertz, respectively.

Propellant slosh: The SM and LM propellant-slosh dynamic stability was a concern because of the lack of baffles in any of the tanks to provide damping, the lack of definition of the slosh dynamics data, and the existence of dynamic changes in the slosh data because of the vehicle center-of-gravity motion. Although these concerns

were not resolved completely, they were resolved to the extent that the total vehicle stability was not a problem during a given SPS-engine burn.

Bending-compensation technique for the CSM/LM configuration: As previously mentioned, the unavailability of good bending data resulted in the implementation of three different CSM/LM bending-compensation designs. Although lack of data was the primary cause, the technique used for the compensation design was partly responsible for the three complete redesigns. Frequency band filtering was used in conjunction with the notch filter technique. The notch filter was sensitive to both increases and decreases in the bending frequencies, whereas the rolloff-type filter was sensitive primarily to decreases. The rolloff filter can be designed such that frequency decreases can be handled by phase stabilization (and increases, usually, by gain stabilization), whereas the notch filter is limited to an "either/or" compensation technique. In retrospect, it is evident that rolloff filtering of the bending dynamics could have prevented some of the cost of the final design.

DEVELOPMENT

Several problems occurred during the development of the Block I system. The following paragraphs include details of those problems.

In-Flight Maintenance

In-flight maintenance was deleted for two reasons. First, although it was technically feasible for the astronaut to detect and replace a defective module or subassembly, it was not an easy task. Modifications made to fulfill the increased humidity requirement made the job even more difficult. In addition, under control system laboratory conditions, a trained technician often required many hours to locate and replace a defective component. Second, the Block II concept made the SCS a backup to the PGNCS and called for (1) complete repackaging techniques for humidity purposes and (2) the addition of redundant control circuits in critical areas; both of these factors, it was believed, eliminated the need for in-flight maintenance.

Humidity

In 1963, electrical failures occurred on the Mercury-Atlas 9 flight. The failures were traced to short circuits caused by condensation resulting from an unexpectedly severe environment of 40- to 70-percent relative humidity at a temperature of 297 ± 3 K ($75^\circ \pm 5^\circ$ F). On Apollo spacecraft, humidity and temperature are maintained by the environmental control system (ECS).

The ECS control is effectively limited to the cabin interior; that is, in front of the equipment panels. The atmosphere behind the panels, in corners, or in protected pockets is beyond direct control by the ECS. Moisture is removed through the suit circuit or with a vacuum cleaner. If the temperature falls below the dewpoint (approximately 289 K (60° F)), condensation will occur. Analysis indicated that condensed moisture could be expected in and around the equipment located in the pressurized

section of the CM even with the ECS operating properly. Based on these factors, a requirement was established that spacecraft equipment withstand 100-percent relative humidity during flight operation. Subsequently, the Block I SCS was modified to fulfill this new requirement.

The nature of the Block I changes consisted of backpotting and adding rubber seals to the electrical connectors. The system later passed qualification testing and was flown successfully on spacecraft 009, 011, 017, and 020. The change in the humidity requirement was a major factor in the Block II design concept.

Block I Connectors

The Block I SCS equipment boxes were interconnected electrically through small multipin connectors. The female half was attached to the spacecraft, and the male half was on the equipment enclosure. Mating difficulty was encountered because of the mass and length of the equipment enclosure and because of the high pin density of the connector. To mate the connector without breaking or bending pins was extremely difficult. Other problems included connector body breakage during insertion and removal of the enclosure, connector damage during crimping and pin removal, retention-spring breakage, and pin burrs.

Because of the numerous problems experienced with the connectors, an alternate procurement source was considered. However, the part was of an advanced design; and, because of the considerable investment in the original vendor, alternate-source procurement was considered impractical.

The high breakage and failure rate of the connector led to unexpectedly high usage of the available supply. This situation eventually resulted in prime-contractor schedule slippage because the supply was depleted and acceleration of resupply was not possible. The resupply problem was caused by the inability of the spring retention clip manufacturer to meet the resupply delivery needs.

The connector problem for Block I was not solved by a design breakthrough or by the purchase of a different connector; instead, development was continuous until the end of the Block I program. The redesign and repackaging of the SCS brought about by the Block II decision included a change to standard connectors. No major problems were encountered in this area on the Block II system.

Single-Point Failures

The process of searching for and eliminating single-point failures has been long and tedious. Designers must continually search the wiring diagrams for trouble areas and document their findings in Failure Mode and Effect Analysis and Single Failure Point summaries. Examples of single-point failures found and corrected to date are as follows.

1. SCS logic bus: A single short circuit to ground would disconnect power to all guidance and control mode and function switches.

2. TVC servoamplifier power switch: A failure in the single switch used for both servoamplifiers would disable TVC completely.

3. Entry roll display: A single power bus failure would disable all three methods of roll display.

4. SCS drivers: A single SCS failure would cause the primary mode to disable the reaction jet drivers.

5. Rotation control for MTVC: A single power failure would leave the rotation control inoperative during MTVC operation.

The identification of single-point failures is difficult and time consuming. A given system generally consists of two modes, the nominal mode and the backup or redundant mode. The nominal and redundant modes are similar, but the system is at least twice as complex as it would be without the redundant mode. A failure, of course, can occur in either mode. The failure conditions and ensuing states usually are known and understood by the designers. Special failure tests are performed to ensure that a single failure (and sometimes a multiple failure) does not disable a redundant system. Although this level of failure analysis is tedious, it is usually successful.

The level of difficulty for single-point failure analysis increases substantially when several systems must be combined to form a total, integrated system. The permutations of the problem elements and the types of failures increase considerably. Consideration must be given to power failures, switch failures, and primary-mode failures. Although these total-system effects should be analyzed at the time of system specification so that each system can be designed in terms of the whole, this procedure is seldom followed because of time and schedule constraints and because of a lack of detailed understanding of the total concept. As a result, many hours are spent over drawings checking for the required redundancy after the hardware has been committed for manufacture.

At this point in the failure analysis, modifications of ground rules introduce problems. For example, an original ground rule was that switches would not short circuit to ground. Experience with the switches later in the program caused this ground rule to be deleted. As a result, a whole new class of failure possibilities was introduced.

The complexity of modern space-flight systems and the stress on high reliability place a premium on fail-safe design and on minimization of single-point failures. Unfortunately, the current state of the design process does not appear capable of handling these requirements. Thus, the development of a method by which a system may be analyzed for possible failure modes quickly and thoroughly is highly desirable. Furthermore, the designers themselves should be made more fully conscious of failure modes and of the importance of system reliability.

Relay Failures

The relay selected by the SCS subcontractor for use in the Apollo SCS early in the Block I program was used throughout Block I and, subsequently, throughout Block II. This relay was chosen on the basis of its performance capability, its performance history, and its size and weight and on the basis of the quality assurance and reliability procedures and practices of the vendor. Few relay problems were initially encountered during the Block I program. Each problem or relay failure (six in all) was analyzed or resolved on its individual merits, and no pattern of failure modes evolved. One of the six Block I relay failures was caused by an internal solder-ball. Extensive research and testing were performed to ensure that this was not a serious problem in the existing Block I SCS. At that time, a change from a soldered case-to-header assembly to an electron-beam-welded assembly was made, and no further solderball problems occurred.

The welded-header relay was used exclusively in the Block II SCS. There were 88 such double-pole, double-throw relays used in this system. The subcontractor procured 4091 relays for the total Apollo Program in two separate 2-week periods (four lots) during January and July of 1966. Another 451 relays were retained by the vendor for group B and C lot acceptance test purposes. From this overall total (4542), there were 27 failures. A limited number of Block II SCS relay malfunctions were encountered during module testing before November 1966. The number and type of malfunctions were not unusual or repetitive, and the analysis and establishment of appropriate corrective action were performed with normal concern. However, in mid-November 1966, two relay failures occurred during black-box-level testing, including one in a qualification test. These failures triggered an extensive investigation by the subcontractor. The effect of the failures at that time was a major concern in the Apollo Program because many of the electronic systems contained these relays. Hence, the SCS subcontractor investigation was monitored very closely by both the prime contractor and NASA.

The two major distinctions made relative to the failures were the types of failures and the test level at which they occurred. These characteristics are summarized in table I. The relay investigation led to the following specific conclusions.

1. Although the relay-failure-rate predictions based on existing experience had not met the originally estimated failure-rate predictions, the SCS would meet the system reliability requirements.
2. Design of the SCS was such that no single relay failure would prevent mission success.
3. The relay represented the best proven state-of-the-art design and was fully acceptable for use in the Apollo Program.
4. Subcontractor in-house module and relay testing effectively screened out normally open, low-contact-force relays and low-insulation-resistance (wet) relays.
5. There appeared to be no change in the test criteria (at the device, module, or relay level) from the existing process that would reduce the probability of relay hangup.

TABLE I. - RELAY FAILURES

(a) Block I and Block II failures

Relay failures	Number
Block I	
In the field	0
At the module level	3
During device predelivery acceptance testing	1
During device qualification testing	<u>2</u>
Subtotal	6
Block II	
During parts testing	2
At the module level	14
During device predelivery production check	1
During device predelivery acceptance testing	1
During device qualification testing	2
During device postdelivery retrofit cycle	<u>1</u>
Subtotal	21
Total number of failures in 4542 relays	27

(b) Relay failure summary

Relay failure mode	Assembly level	Number of failures		
		Block I	Block II	Total
Contamination	Module	1	5	6
	Device	0	2	2
Normally open contacts failed to close	Module	2	8	10
Normally closed contacts failed to open	Device	2	2	4
	Part	0	2	2
Normally closed contacts failed to make contact	Device	0	1	1
Broken internal coil wire	Module	0	1	1
Poor solder connection	Device	1	0	1

Design Features

The general hardware design features of the Block II SCS for the Apollo spacecraft are characterized as follows.

1. The physical system comprises 14 contract end-items (fig. 14 and table II): one reaction jet engine on/off control (RJEC); one electronics control assembly (ECA); one electronics display assembly (EDA); one GDC; one thrust-vector servoamplifier (TVSA); two gyroscope assemblies (GA-1 and GA-2); two FDAI units; one GP/FPI;¹ one ASCP;¹ one THC;¹ and two RHC units.¹

2. These components are hermetically sealed, conduction-cooled, individually mounted structures; except for the gyroscope assemblies, no vibration isolation is provided.

3. The packaging of hardware components into individual assemblies is arranged for a logical functional grouping of system elements and for ease of system malfunction isolation and troubleshooting.

4. The electronic assemblies are electron-beam-welded aluminum structures that house stacked, cordwood-type modules containing high-reliability electronic piece parts. The parts are interconnected by welded sections of bus wires that also connect to module exit headers. The modules are interconnected by welded exit connections to bus wires sandwiched between thin Mylar-film matrix sheets. The matrices are connected to the external connectors by lead wires.

5. Approximately 15 percent of the total system electronics consists of integrated circuit flatpacks that significantly reduce system weight, volume, and power requirements.

6. The display panels and indicators are electron-beam-welded aluminum structures. The indicator enclosures house servometric-type meter movements and high-reliability electronic piece parts and incorporate EL dial face lighting. The control panel houses high-reliability electronic resolvers.

7. The hand controllers are aluminum structures incorporating a tapered-wedge-type lock/stow mounting feature. The controller-handgrip form factors and force and travel provisions are human-factored for optimum control performance in the space-flight environment.

8. System functions that are critical to crew safety are ensured by the addition of switchable redundant circuitry.

¹No coldplate cooling.

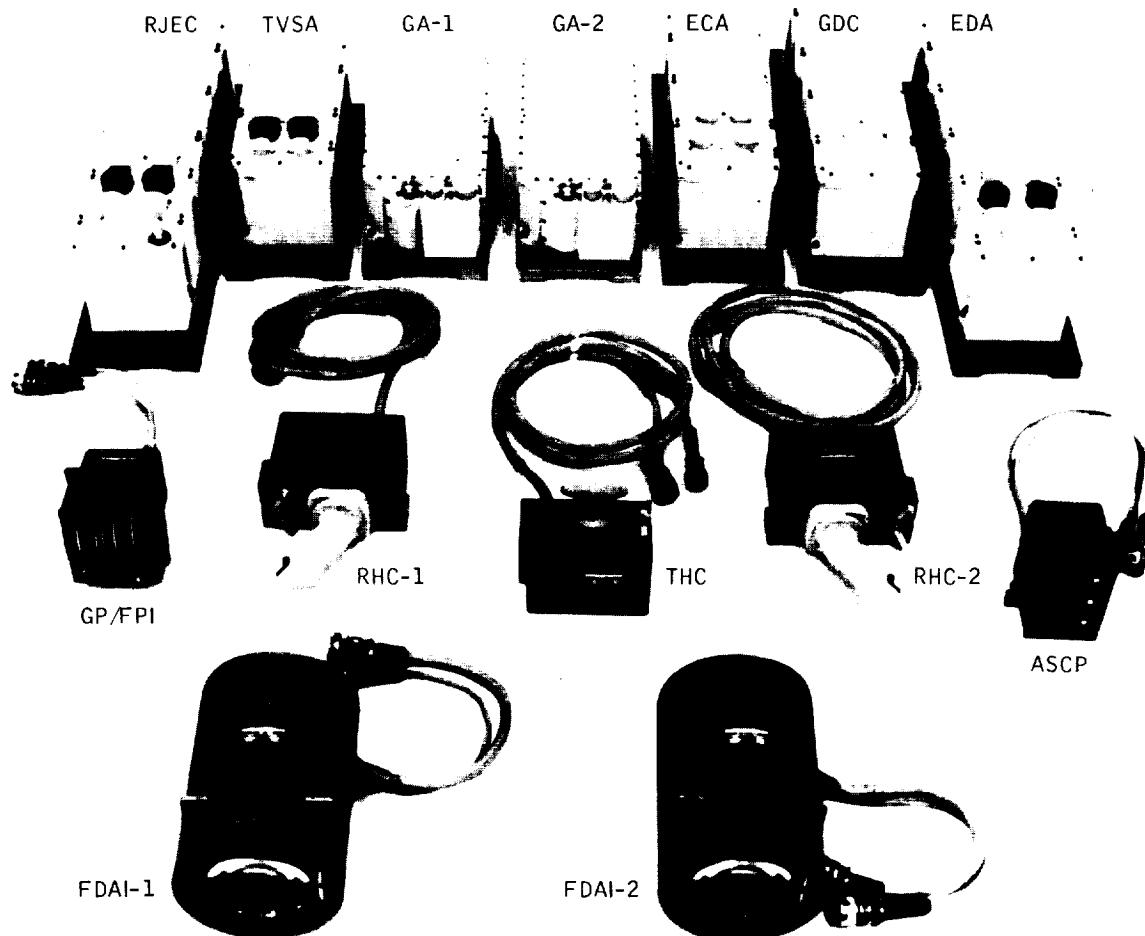


Figure 14.- Block II stabilization and control system components.

TABLE II. - HARDWARE PHYSICAL CHARACTERISTICS

Block II CEI ^a	Quantity per system	Weight		Volume		Block I CEI equivalent (b)	Reason for difference
		kg	lb	cm ³	in ³		
ECA	1	7.5	16.5	9 293	567	Pitch ECA (1)	Power saving and reliability considerations of Block II concept
TVSA	1	5.7	12.5	7 048	430	Yaw ECA (1)	
RJEC	1	9.4	20.7	13 145	802	Roll ECA (1)	
EDA	1	11.4	25.0	12 473	761	Display portion of DECA ^c (1)	Added Block II display redundancy
GDC	1	11.3	24.9	11 243	686	Auxiliary ECA (1)	Block II ARS mechanization differences
GA	2	20.8	45.8	17 209	1050	AGAP ^d (1) portion of DECA, and RGP ^e (1)	Reduced criticality of matched components
RHC	2	8.6	19.0	4 589	280	RHC (2)	Block II requirement of discrete control for each direction of rotation
THC	1	2.7	6.0	1 836	112	THC (2)	Block II requirement for redundant outputs
FDAI	2	8.2	18.2	11 473	700	FDAI (1)	No functional difference
ASCP	1	1.5	3.3	1 147	70	AS/GPD ^f (1)	Break up displays to provide greater flexibility of smaller packages
GP/FPI	1	1.3	2.9	1 180	72		Add fuel display and redundancy requirements
Totals	14	88.4	194.8	90 636	5530	(g)	Not applicable

^aContract end-item.

^bNumbers in parentheses indicate quantity.

^cDisplay electronics control assembly.

^dAttitude gyroscope accelerometer package.

^eRate gyroscope package.

^fAttitude-set gimbal-position display.

^gTotal quantity, 13; total weight, 108.4 kg (238.9 lb); total volume, 106 928 cm³ (6524 in³).

TESTING

The testing of the SCS was the culmination of the quality assurance program. The concepts, requirements, designs, and materials were merged during the testing into the final product that was demonstrated. The total testing effort was divided into four distinct categories: development, qualification, acceptance, and installation checkout tests.

Development Testing

The development tests were a series of functional, environmental, and engineering evaluation tests performed on materials, parts, and components to determine item suitability for incorporation into the SCS, to determine item performance characteristics, and to evaluate and improve designs. These tests included such activities as (1) determination of part, component, and subassembly characteristics; (2) selection of materials and parts by comparative tests; (3) evaluation in terms of critical environment, including anticipated or unknown amplification factors caused by the particular application; (4) determination of performance stability or repeatability; (5) evaluation of changed or improved designs; (6) establishment of design margins; and (7) evaluation of parameter variation effects.

The subcontractor was authorized ten complete engineering systems (five in both Block I and Block II configurations) to develop flyable configuration hardware. In Block I, the contractor attempted to enforce formal configuration control between the contractor and the subcontractor on these development systems. In retrospect, this practice was neither useful to the program nor helpful to the subcontractor; enforced configuration control was not even practical from a design standpoint. On Block II, configuration control was abandoned as a workable method for developing a system; instead, the more orderly approach of establishing a design baseline followed by formal design reviews was instituted.

Qualification Testing

The qualification tests consisted of a series of performance and environmental tests performed on production piece-part, end-item, and system hardware to demonstrate that the items met all applicable requirements of design and performance. Materials were qualified by the contractor and listed in an approved materials specification, which the subcontractor used. When new materials were selected, the subcontractor tested these materials and submitted test results and samples of each material to the contractor for approval. Electronic piece parts were procured from vendors by means of rigidly controlled high-reliability parts specifications. Each specification contained a section that delineated the testing requirements for the parts. This section included the following sequential series of tests.

1. Total sample (100 percent) processing tests: Visual inspection, high temperature, constant acceleration, temperature cycle, gross and fine leak tests, particle detection, burn-in, and X-ray

2. Group A acceptance tests: Group A 100-percent tests, visual inspection, mechanical inspection, and basic electrical parameters (Group A sample tests included high-temperature leakage, low-temperature gain, offset voltage, and dynamic resistance.)

3. Group B lot integrity tests: Visual inspection, mechanical inspection, solderability, temperature cycling, thermal shock, moisture resistance, shock, vibration, constant acceleration, terminal strength, salt atmosphere, high-temperature life, and 1000-hour operating life

The qualification testing of completed end-items and of the system was divided into two major areas: design proof tests and system life tests. The design proof tests were performed in climatic and dynamic environments and were imposed on one unit of each operating end-item to specification limits. The system or mission life tests consisted of exposure of one complete complement of operating end-items (or a system) first to vibration and then to a salt/fog atmosphere. All these tests were preceded by a checkout test to establish baseline performance data, and the environments then were sequenced to optimize use of time and facilities. Performance data were taken during and after the tests to demonstrate that the SCS met all applicable design and performance requirements and to verify that no serious design weaknesses existed that could cause inconsistent or marginal performance or a high probability of failure.

Acceptance Testing

Acceptance of completed SCS hardware by the contractor required that acceptance tests, at both end-item and system levels, be performed successfully before shipment. Successful performance was defined as a quality-control-witnessed test performed in accordance with a contractor-approved acceptance test procedure (ATP) wherein all test parameters measured fell within the specified tolerance limits. Parameters that were outside these tolerance limits could be submitted as a waiver request, which required contractor approval before shipment of the hardware. Each ATP specified the detailed steps to be followed by the test operator in performing the test, the configuration of the test articles, and any test equipment used to perform the test. Each ATP included the step-by-step settings to be made, the readings to be taken, the acceptable limits for each reading, and the instructions for recording data. The sheets on which these data were recorded became a part of the data package that accompanied each end-item and system throughout its service life.

Vehicle Checkout

As envisioned early in the Apollo Program, all hardware received from the subcontractor would undergo a receiving inspection test at the contractor facility. Before the receipt of the initial hardware, however, this concept was abandoned for one in which delivered hardware was to be installed directly into the spacecraft. The role of the contractor was to demonstrate, through tests, the continued functional and performance capability of the system when mated physically and electrically with actual spacecraft interfacing systems. The basic philosophy adopted for these tests was

two-pronged. First, proper system operation in the spacecraft structure and with spacecraft power applied through the spacecraft interconnecting wire harness was to be verified. All functions and modes were exercised to established performance requirements within allowable tolerance limits. Second, electrical and functional interface compatibility between the given system and the interfacing systems was to be demonstrated. These tests demonstrated compliance with the Apollo requirements for polarity, phasing, switching, scaling, and gain.

The NASA-supplied automatic checkout equipment (ACE) was used for testing all spacecraft systems. This ACE hardware is capable of injecting stimuli and accepting signal-conditioned readouts simultaneously for a large number of systems. Use of ACE superseded the normal troubleshooting techniques of past programs in which each system had its allotted time on the vehicle and its own test console with the capability for injecting stimuli, accepting readouts, and monitoring numerous test points for probing suspected trouble areas.

Service Life

The original service life requirements for the SCS were specified as 1000 hours of ground checkout and 400 hours of flight. These values were based on the best estimates available at the time. The 400 hours of flight represent a typical 14-day lunar mission and proved to be a good estimate. However, the estimated 1000 hours of ground checkout time were insufficient.

The subcontractor normally used between 300 and 400 hours before end-item delivery, considering that power was applied to lower level assemblies and to each subsequent higher level subassembly through end-item delivery. The contractor then performed system and combined-systems tests both at the manufacturing facility and at the launch facility. It became evident that the system had to be qualified to a higher than specified service life. Subsequently, one complete system was subjected to an additional 7700 hours of testing with no parameters falling outside the so-called "mission time extreme environment" values. Each device also was subjected to individual tests for comparison with the initial values.

FLIGHT RESULTS

Through 15 Apollo flights - 4 unmanned (spacecraft 009, 011, 017, and 020) and 11 manned (spacecraft 101, 103, 104, 106 to 110, and 112 to 114) - no major problems involving the SCS were encountered, and all SCS test objectives were achieved. In the following paragraphs, additional information on SCS performance and anomalies for each specific flight are presented.

1. Apollo-Saturn 201 (spacecraft 009), Apollo-Saturn 202 (spacecraft 011), Apollo 4 (spacecraft 017), and Apollo 6 (spacecraft 020): The SCS performed satisfactorily throughout these missions.

2. Apollo 7 (spacecraft 101): All detailed test objectives were accomplished satisfactorily. These objectives were SCS attitude control, SCS velocity control, manual takeover of TVC, and SCS drift checks. Two anomalies were reported during the Apollo 7 flight: one concerned the commander's RHC and the second concerned the EDA. The RHC failure (believed to be a sticky cam follower causing failure of a switch to open) occurred once during the flight. After the flight, the device was returned to the subcontractor facility, where it was subjected to room-temperature and high- and low-temperature vacuum tests with satisfactory results; that is, the flight failure could not be repeated. Design of the hand controller for spacecraft 103 and subsequent vehicles was different in this particular area (i.e., the sleeve bearings had been replaced with ball bearings), and no additional problems were experienced. In the EDA failure, FDAI-1 failed to transfer power to the SCS properly. System tests on the vehicle were performed by the prime contractor with satisfactory results. The device also passed a complete acceptance test performed by the subcontractor with no malfunctions. However, simulation of the failure (partial power transfer) on a subcontractor in-house system revealed that if a particular relay failed, the anomaly would occur. Although a relay malfunction could not be induced at either the device or module level, a solderball was found in the relay when it was opened for visual inspection. It was concluded that, under the zero-g condition in flight, the solderball could have caused the problem.

3. Apollo 8 (spacecraft 103): The SCS performance was satisfactory throughout the flight.

4. Apollo 9 (spacecraft 104): The SCS performance was satisfactory throughout the flight. The detailed test objectives of MTVC takeover from the PGNCs and the CSM autopilot stability margin-stroking test were accomplished satisfactorily.

5. Apollo 10 (spacecraft 106): In general, the SCS performance was satisfactory throughout the flight; however, the crewmen stated that the FDAI indicated excessive drift in the pitch and yaw axes, although no actual measurements were recorded. After the flight, thorough testing of both gyroscope assemblies and of the GDC at both device and system levels did not verify the anomaly. A comparison of attitudes displayed by the SCS and the PGNCs after Earth-orbital insertion showed differences of less than 0.1° in all axes; these fractional differences indicated proper performance early in the mission. Because no attempt was made to measure the drift accurately, it was possible that the actual drift of the FDAI was not as great as it appeared. The anomaly was closed.

6. Apollo 11 (spacecraft 107), Apollo 12 (spacecraft 108), Apollo 13 (spacecraft 109), Apollo 14 (spacecraft 110), Apollo 15 (spacecraft 112), Apollo 16 (spacecraft 113), and Apollo 17 (spacecraft 114): Performance of the SCS was satisfactory throughout these flights.

CONCLUDING REMARKS AND RECOMMENDATIONS

With the exception of three minor anomalies, performance of the stabilization and control system was satisfactory on all Apollo flights, which included 4 unmanned and 11 manned vehicles. This success can be attributed to conservative design techniques; use of state-of-the-art components; extensive development, qualification, and acceptance testing; extensive vehicle checkout at the prime-contractor facilities and at the John F. Kennedy Space Center; and various simulation programs conducted by the Lyndon B. Johnson Space Center (formerly the Manned Spacecraft Center), the contractor, and the subcontractor.

Following is a list of recommendations that should be considered for any future program similar to the Apollo Program.

1. A strong effort should be made to establish baseline requirements before the start of the hardware design process. For example, requirement changes affecting hand controllers, humidity, in-flight maintenance, and many other parameters caused major redesign during the Apollo Program.
2. Multiple-source procurement should be used for advanced design approaches. Generally, state-of-the-art design approaches should be adhered to; however, if this is not possible, alternate sources of procurement should be examined to prevent the development problems of a supplier from hindering the design of the total system.
3. Firm specifications should be provided for crew personal-preference items.
4. A technique should be used to assist in the rapid identification of single-point failures. The Apollo method required many engineers to search diagrams for problems, but this technique is not altogether successful for complex systems.
5. An accurate low-g propellant-slosh model should be developed. The Apollo model produced some physically unreasonable results in simulations, and predictions based on this model were not confirmed by any of the flights.
6. A better method of determining vehicle parameters such as body bending should be developed. The Apollo SCS underwent several redesigns because of updates in body-bending data.
7. Criteria and methods to extend the duration of hardware reliability should be established. The long checkout times for complex space vehicles combined with the extended duration of some missions place a premium on the storage and long-life operation of the equipment. As an example, to obtain a realistic service life estimate, system qualification testing might be extended to the point of hardware wearout.
8. Such devices as hand controllers should be designed as plug-in units so that cabling can be part of spacecraft wiring. Failure to follow this procedure on Apollo spacecraft resulted in the development of a completely new cable at a critical time in the program. Although the problem was resolved, extra care had to be taken for protecting the cables.

9. An in-house test capability setup that is maintained in a current configuration at all times should be provided for the investigation of interface, functional, performance usage, and operational problems. Use of such a capability setup would minimize surprises during the initial operational phases of a program and prevent use of a flight vehicle as a test bed in resolving complex problems.

10. Serious consideration should be given to the use of solid-state switching in lieu of relays. If relays are used, superscreening tests should be established to ensure high reliability.

11. Autopilots should be designed such that less sensitivity to spacecraft design variations (changes in vehicle bending mode frequencies and amplitudes caused by mass property changes) exists.

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