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CR 86228

LUNAR ORBITER PROGRAM
GUIDANCE AND CONTROL SYSTEM
DESIGN SURVEY

August 1969

NASA/ERC DESIGN CRITERIA PROGRAM

GUIDANCE AND CONTROL

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Electronics Research Center

Cambridge, Massachusetts

Contract NAS 12-2027

AEROSPACE SYSTEMS DIVISION
THE BOEING COMPANY
Seattle, Washington

D2-114277-2

FOREWORD

This report is a design survey of the Lunar Orbiter Guidance and Control system and is one in a series of design surveys prepared for the NASA Design Criteria Program. The objective of this NASA program is to establish a unification of design approaches in the development of space vehicles and their major components. The surveys are intended to document design experience gained from specific NASA projects and will be used as a data source for design criteria monograph documents.

This design survey was performed in accordance with the Statement of Work in NASA contract NAS 12-2027, "Lunar Orbiter Design Criteria Survey," for the NASA Electronics Research Center Design Criteria Office, directed by Mr. Frank J. Carroll, Jr.

The material for this report was gathered, prepared, and documented under the direction of the Program Manager, J. E. Montgomery. Major sections of the report were contributed by Messrs. D. C. Fosth, E. W. Kangas, R. L. Maxwell, W. I. Mitchell, G. E. Morrison, G. A. Price, R. E. Risdal and W. F. Yee. Mr. V. L. Minter assisted in compiling and editing the final manuscript as well as writing portions of the report. Numerous other Lunar Orbiter team members who participated in the design, development, and operation contributed much needed assistance, consultation, and review.

The report is organized in the following manner:

Summary - briefly states the most important items from this design survey.

Introduction - describes the overall Lunar Orbiter program; the successes and failures; the spacecraft and the guidance and control system.

Subsystems - are described, requirements listed, development and operation are discussed in detail, conclusions and recommendations are given.

Components - are treated in a manner similar to the subsystems but to a finer level of detail.

Bibliography - lists a selection of significant references for those who "want to dig deeper."

The contractor's designation for this report is D2-114277-2.

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Prepared by

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TABLE OF CONTENTS

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TABLE OF CONTENTS (Cont'd)

TABLE OF CONTENTS (Cont'd)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Cont'd)

LIST OF ILLUSTRATIONS (Cont'd)

LIST OF ILLUSTRATIONS (Cont'd)

1.0 SUMMARY

This design survey presents the results of an eight month study of the Lunar Orbiter Guidance and Control System and program documentation. Highlights of the experiences gained from program initiation through the final terminal crash maneuver of L.O. V are reported. The things of value to the designer of future spacecraft are emphasized. The evolution of the design was studied and the reasons for changes have been indicated. Considerable effort was made to identify and report the cause for difficulties that occurred. Reasons for success as well as failure are summarized in the following paragraphs beginning with the total spacecraft program; proceeding to an in-depth review of the guidance and control system, its subsystems, and finally the G&C components.

PROGRAM FEATURES

The items which had a major impact on the overall program were the following:

1) Basic program objectives remained unchanged throughout the design and operational flight phases.

2) The test concept was that the initial flight would be a complete operational mission. Every flight spacecraft was subjected to a "Flight Acceptance Test" which simulated the nominal operation environment for temperature, vacuum and vibration. Every component was subjected to prior tests which eliminated serious problems at the spacecraft level.

3) Technical reviews among the contractors and government agencies were held throughout the program beginning with engineering information exchanges, preliminary design reviews, critical design reviews, spacecraft compliance reviews, and culminating in a Flight Readiness review which involved all the companies and agencies concerned with the launch. It was most significant that the standard of conduct for these reviews was "What was right rather than who was right."

4) A strong team concept with close working relationships was maintained among customer and contractor personnel. Test teams were assigned to particular spacecraft at final assembly, and continued through test, planned launch, and flight operations. The continuity of personnel from design through flight operation was important.

5) Space proven hardware and techniques were used wherever possible.

6) The incentive features of the contract covering delivery, cost, and performance motivated the contractor and the government to fulfill their side of the contractual interface in a timely manner.

GUIDANCE AND CONTROL SYSTEM

Guidance and Control System concepts which contributed to the success were:

1) "Put the intelligence on the ground," was a guiding criterion when choosing between competitive designs. This led to a simple system implementation.

2) Complementary paths of information via the engineering telemetry gave a comprehensive picture of spacecraft status and performance. Alternate modes of operation allowed the flight operations crew to exploit the remaining spacecraft capability. Examples were: the use of the high gain antenna signal strength map to corroborate the questionable roll information from the star tracker; the use of the solar panel array current and voltage as a measure of the angle off the sun for angles exceeding the sun sensor saturation level.

3) Operational flexibility was provided. For example, the mission could be controlled from either stored programs or real time commands.

4) A three axis stabilized configuration was selected which best suited the mission requirements. The technology was developed by the Jet Propulsion Laboratory for interplanetary spacecraft.

ATTITUDE CONTROL SUBSYSTEM

The most significant features of the design are summarized in the following:

1) The attitude control performance actually experienced was close to that predicted for each of the following items: deadzone width, limit cycle rate, maneuver rate, settling time, accuracy, effect of disturbance torque, and impulse requirements.

2) The options available in attitude control provided the work arounds which salvaged the missions. The options were:

   a) To hold attitude using error signals from the Sun and Canopus sensors or using the gyros in the rate integrate mode. This gave an inertially referenced attitude when the Canopus tracker gave trouble.

   b) To select wide or narrow reaction control threshold deadzones. This gave 2^{o} or $.2^{o}$ limit cycle, also $.05^{o}$/sec or $.5^{o}$/sec maneuver slew rates; the former conserved reaction control gas.

   c) To select coarse or fine sun sensors. This gave a wide angle reference and an additional deadzone option. It was used to conserve gas when re-acquiring the sun.

   d) To switch the Canopus tracker off and on without necessarily using the error signal for attitude control. This technique was used often to track Canopus instead of glint, also to allow the image dissector tube to "heal" after being exposed to excessive light flux.

REACTION CONTROL

The cold gas Nitrogen reaction control subsystem operated reliably on all flights, with a delivered specific impulse of at least 68 seconds, and with leakage less than .5 lb of gas per year. The most important design features were:

1) The brazed and welded construction prevented leakage.

2) The cleanliness requirements, i.e., no metallic particles larger than 5 microns, nor non-metallic greater than 25 microns were allowed, prevented contamination from becoming a problem causing leakage or jamming. Filters provided a redundancy to the cleanliness provisions.

3) The common N_2 supply for the Reaction and Velocity control subsystems provided an adequate gas supply.

THRUST VECTOR CONTROL SUBSYSTEM

The most significant design features of the thrust vector control were:

1) Electromechanical position servo actuators were developed to gimbal the engine. This control concept was adopted 4 months after contract go-ahead. The one pound thrust reaction control jets on the tips of the solar panels were deleted.

2) Dynamic stability of design provided a 6 db gain margin based on a worst case combination of conditions. The large center of mass changes postulated for propellant migration between paired tanks did not occur: thus, flight performance did not approach design limits.

3) Dual lead-lag compensation networks on gyro position error signal made the system susceptible to mechanical and electrical noise. Problems were solved by filtering high frequency noise and setting limits on allowable gyro noise.

VELOCITY CONTROL SUBSYSTEM

The velocity control subsystem functioned as intended on all flights. An error analysis provided the basis for permitting the use of existing hardware.

GUIDANCE CONCEPT

The most important feature of the guidance concept was the use of ground based intelligence, skill and facilities in determining the spacecraft state vector, computing required maneuvers, commanding their execution, determining the new state and adjusting the subsequent planned events to best accommodate the mission objectives. The major error found in locating photo sites was the uncertainty in the lunar gravitational model.

INERTIAL REFERENCE UNIT

The Lunar Orbiter IRU provided a 3 axis attitude reference and integrating accelerometer in a unit weighing 13.4 lb. and requiring less than 30 watts. Gyro drift was in the range of $1/2^{\circ}$ per hour. Once committed to launch no life, wear-out, or reliability problems occurred on missions up to a year in length.

The principal problems were the result of:

1) Unrealistic severe vibration requirements were originally specified and relaxed late in the qualification program.

2) The Sperry SYG 1000 gyro was not being produced in quantity. The low yield of acceptable gyros made it necessary to obtain a second source of gyros.

The back-up program using the Kearfott alpha gyros in the Sperry frame and electronics was successful.

SUN SENSORS

The Sun Sensors used an arrangement of cells tailored to the Lunar Orbiter configuration to give complete spherical coverage. The sensor elements were the same as those used on the O.S.O. spacecraft built by Ball Brothers. To avoid introducing pointing errors from the nearby illuminated moon required the wide field of view cells to be switched out. The problem most affecting delivery and test schedules was caused by weather and cloud conditions at Boulder, Colorado since the actual sun was used for calibration and test.

CANOPUS STAR TRACKER

The Canopus star tracker was consistently hampered by "glint" from illuminated spacecraft appendages, from earthlight, moonlight, and direct sunlight. Causes of the trouble were the sensitivity of the tracker to stray light coupled with the arrangement of the tracker field of view and the spacecraft appendages.

FLIGHT ELECTRONICS CONTROL ASSEMBLY

This unit was specifically designed to the requirements of the Lunar Orbiter spacecraft and mission. It provided timing, sequencing and control of all spacecraft events from either stored program or real time commands. It also provided the electronics for closing the control loops around the reaction and thrust vector control subsystems.

The criteria most emphasized in the design were:

a) High reliability - affected the choice of parts, processes, and circuit techniques. The design was essentially single thread, but with redundant clock oscillators.

b) Low power - driver circuits were designed to be in the off state during quiescent periods.

c) Low weight - integrated circuits were selected for the programmer even though the reliability of mass produced articles in 1964 was not yet firmly established.

d) Resistance to electromagnetic interference - voltage margin for digital logic state switching was required for worst case design conditions of temperature and voltage.

Unexpected design and manufacturing problems which were solved during the program were: difficulty in obtaining approved parts, static discharge causing micrologic module failures, contamination of one lot of micrologic modules, stress from the potting process causing electrical value change, cross talk between the redundant clocks causing extra bits in the time words.

THRUST VECTOR CONTROL ACTUATOR

This unit was a hermetically sealed servo using a d-c motor and solid state electronics. It was developed specifically for this application. The short design and development time allowed before first article delivery pushed the bulk of the servo de-bugging problems into the acceptance and qualification phases of the program. With hindsight one can say the specified requirements for 10°/sec rate, for $\pm$ 2.9° stroke, and for irreversibility during launch were somewhat conservative.

RECOMMENDATIONS FOR DOING DIFFERENTLY

The important items which should be done differently on future spacecraft are:

1. The mission system and component requirements analysis should be done before specifications are sent to vendors and designs are firm. It is important that both performance and environment be realistic.

2. The Star tracker optical, mechanical, and electronic analysis and design should be checked in detail before proceeding with construction. Test methods and facilities to evaluate component and spacecraft glint should be developed.

3. Inertial instruments for an IRU should be selected from a mature design to avoid extensive development problems. The marks of maturity for gyros are to be in quantity production on the same article, to have service experience, and to have service recorded for failure analysis. Any change, even of the float fluid, should be considered a major design departure.

4. The ability to command one spacecraft only when two or more are present should be assured: preferably by assigning different radio link frequencies to each.

5. Operational procedures developed to solve emergency situations should not be continued on subsequent missions; for example, flying off-sun for thermal relief, or continuing flights with the Canopus tracker glint problem unresolved.

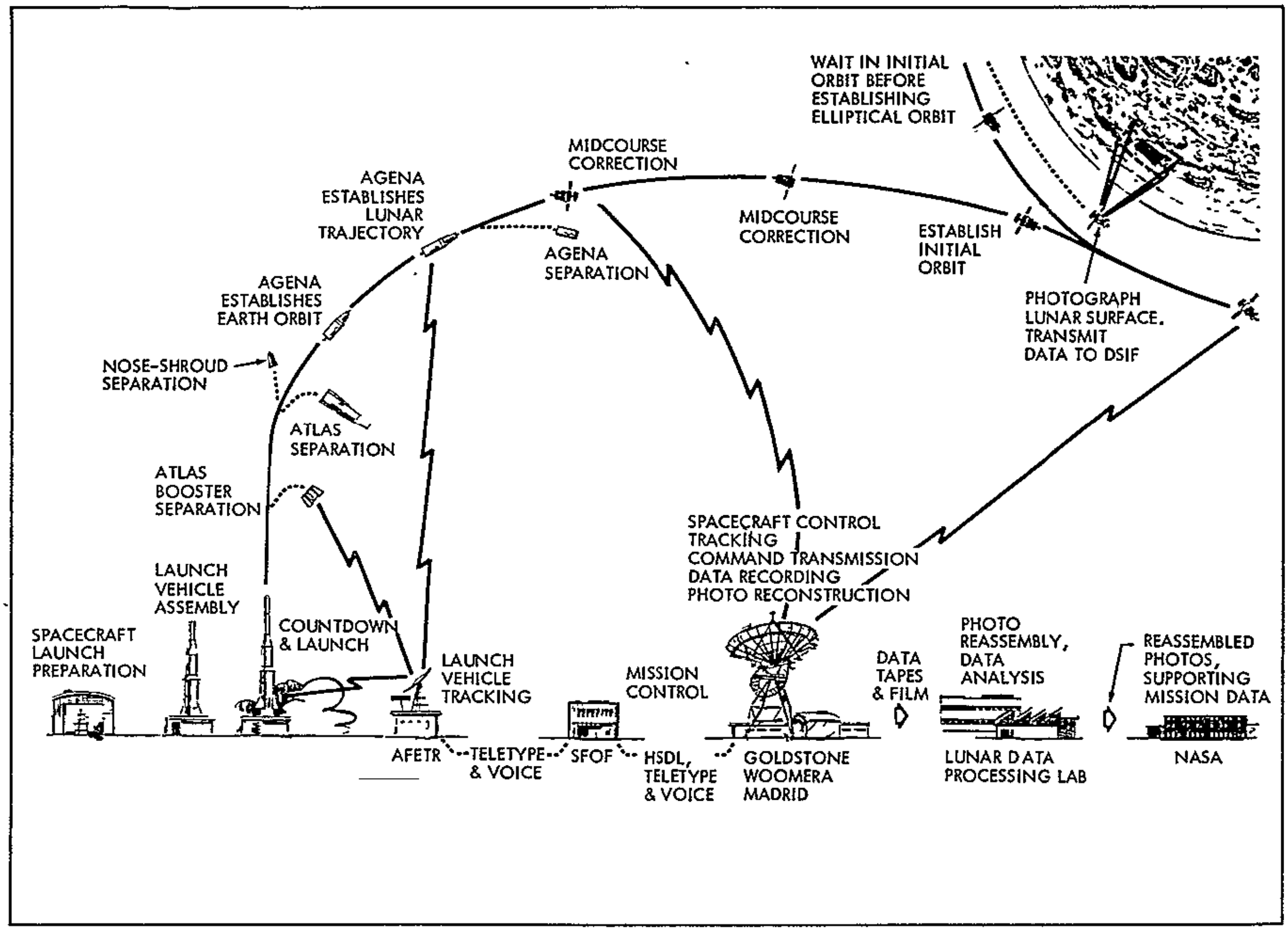


Figure 2-1: THE MISSION

2.0 INTRODUCTION

The Lunar Orbiter program was directed by NASA's Office of Space Science and Applications, and managed by the agency's Langley Research Center. Contract award was made to The Boeing Company in March 1964, and the first mission occurred on August 10, 1966. The primary objective of the Lunar Orbiter spacecraft was to obtain high resolution photographic data of specific lunar areas to aid in the selection of Apollo landing sites. Secondary objectives were to obtain data about the size, shape, and gravitational field of the Moon; the micrometeroid and radiation levels in the lunar environment; and moderate resolution photography of extensive areas. However, it was not required that all objectives be satisfied on any one mission. Specific mission plans were not identified until shortly before each spacecraft flew. This resulted in a broad range of hardware capability being designed into the spacecraft. It provided the flexibility to plan specific missions around this design capability and to take advantage of results from previous flights. This flexibility also provided the capability to utilize backup modes and work-around methods to accommodate in-flight anomalies.

A typical mission is illustrated in Figure 2-1. The flight profile events are illustrated in Figure 2-2 and described in the following. The spacecraft and Atlas/Agena launch vehicle underwent countdown as one functional unit, called the Lunar Orbiter space vehicle. At the completion of countdown the Atlas boosted the space vehicle to an altitude of 85 miles before Atlas separation. The spacecraft nose shroud, no longer required for atmospheric protection, was jettisoned prior to Agena firing. The Agena accelerated the Agena/Orbiter combination into a 100-mile-altitude parking orbit. After coasting to the translunar injection point, the Agena engine was re-ignited and accelerated the spacecraft to escape velocity. The Lunar Orbiter was then separated and began its coast to the Moon.

After spacecraft separation, signals from the flight electronics control assembly initiated squib firing to deploy solar panels, antenna booms, and to activate the reaction control subsystem. Sun sensors and rate gyros controlled the opening of small nitrogen thrusters, causing the spacecraft to pitch and yaw until the correct sun attitude was attained. The roll rate was reduced to near zero by another set of thrusters. At a later time, the spacecraft performed a series of roll maneuvers to enable the star tracker to acquire Canopus. This completed the initial three axis orientation and stabilization of the spacecraft.

Continued tracking by the deep space stations enabled the Space Flight Operations Facility to compute spacecraft range, velocity, and flight path. Deviation from the intended trajectory was corrected by transmitting the necessary guidance commands to the spacecraft flight electronics control assembly, where they were stored and subsequently executed. At the proper time the spacecraft broke the sun/star reference attitude and oriented the spacecraft to point the rocket engine in the desired direction. Attitude reference was then provided by the inertial reference unit. The 100-pound thrust engine was ignited, changing the spacecraft velocity vector until the integrated output of accelerometer measuring the velocity change, shut

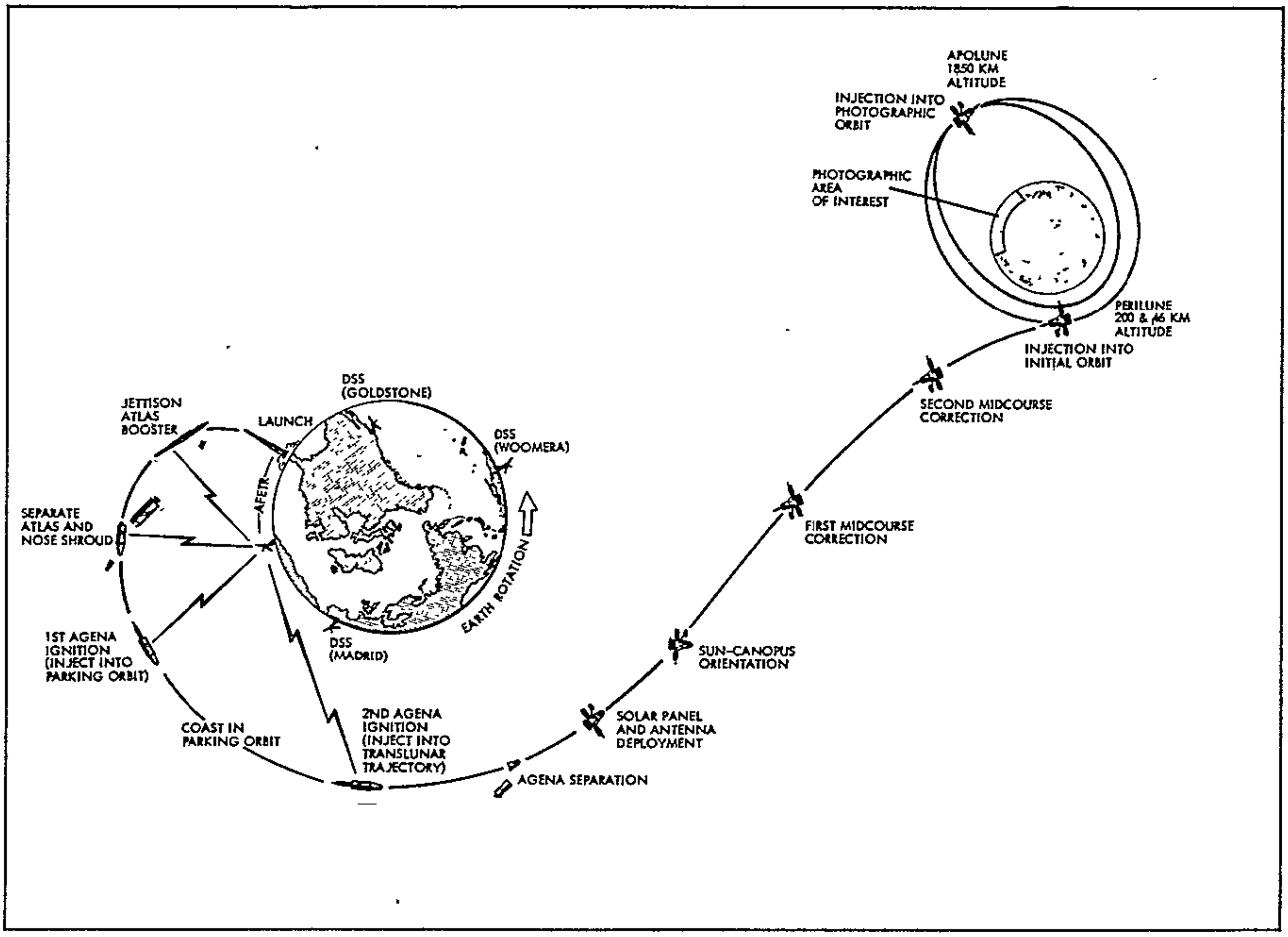


Figure 2-2: TYPICAL FLIGHT PROFILE

off the engine. The spacecraft systems were designed to perform a second midcourse correction if required. However, none of the five missions required a second correction.

Trajectory data obtained throughout the translunar flight was used by the Space Flight Operations Facility to compute the velocity and direction changes required to achieve initial lunar orbit. The necessary commands were again transmitted to the flight electronics control assembly. At the proper time the spacecraft was injected into Lunar Orbit. Additional tracking of the spacecraft by the deep space stations accurately established the actual orbit and subsequent orbit changes were accomplished as required by the mission.

The photographic phase of the mission required a precise knowledge of the orientation and position of the spacecraft to point the camera at the desired target area. The desired spacecraft maneuvers from the sun/star reference attitude were determined at the Space Flight Operation Facility from tracking data. Commands were transmitted to the flight electronics control assembly for execution at the proper time. The flexibility of the photographic subsystem, coupled with the ability to design the mission for a range of orbit parameters such as inclination and perilune altitude provided great flexibility in selecting the size, shape, and location of the areas to be photographed.

MISSION PERFORMANCE

The mission results are summarized in the following so as to provide a frame of reference and basis for understanding: (1) difficulties or problems in mission operation as a result of the design selected, (2) how it should or should not have been done (hindsight), and (3) the work-around techniques that were possible because of the design flexibility.

Major program accomplishments were:

a) Initial program objectives essentially completed in three flights.

b) Photographed over 99 percent of visible side of Moon with resolution capability at least 10 times better than Earth based observations.

c) Photographed virtually the entire far side of Moon.

d) Provided data for the selection of eight candidate Apollo landing sites.

e) Provided photographs of numerous areas of scientific interest with resolution capability of 1 to 2 meters.

f) Developed techniques to employ vertical, oblique, forward stereo, side stereo, and converging telephoto stereo photography to conduct site search, site confirmation, and mapping missions of celestial bodies.

g) Devised operational techniques to command and control three spacecraft in lunar orbit at the same time.

h) Crashed all spacecraft on the Lunar surface (four of five upon command from the SFOF) and eliminated potential future problems of unwanted communication from the spacecraft or of collision with Apollo.

i) Provided the Apollo Manned Space Tracking Network with a live model for training purposes during the L.O. extended missions.

j) Provided radiation data in the lunar environment.

MISSION I

L.O. I (spacecraft number 4) was launched August 10, 1966.

a) Photographed nine potential landing sites in a southern latitude band of Apollo zone of interest with eight meter resolutions.

b) Eliminated some potential Mission II sites from further consideration.

c) Accurately established the desired lunar orbit characteristics.

d) Provided data for lunar gravitational model for 12 degree orbit inclination.

e) Performed 574 attitude maneuvers, five velocity maneuvers, and processed 4,510 transmitted commands.

f) Obtained the first oblique photos of Moon and the Earth as seen from the vicinity of the Moon (Figure 2-3).

g) Obtained the first high quality photos of the far side of the Moon.

h) Transmitted more than 400 photographs to Earth, providing photographic coverage of about 150,000 square miles of the lunar surface visible from Earth and about two million square miles of the Moon's far side were photographed from about 1000 miles above the surface.

MISSION II

L.O. II (spacecraft number 5) was launched November 6, 1966.

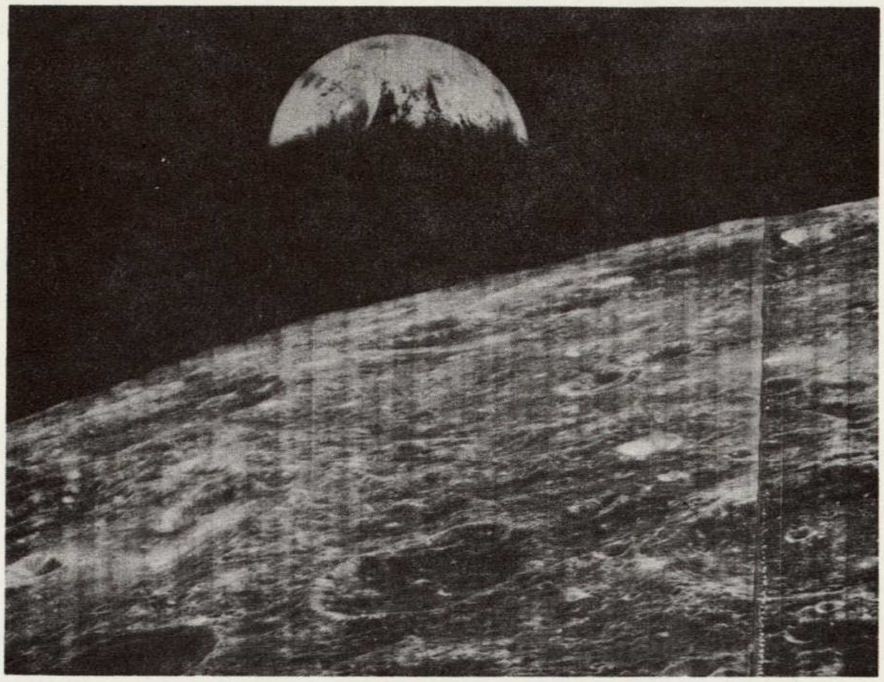
a) Photographed 13 potential landing sites in a northern latitude band of Apollo zone of interest with one meter resolution.

b) Improved oblique photography control and pointing procedures (e.g., Copernicus photo, Figure 2-4).

c) Photographed 17 secondary photo sites within the orbit inclination limits on front and far sides of the Moon.

d) Identified Ranger VIII impact area.

e) Developed procedure to synchronize DSN station clocks to within 50 microseconds.



Earth was photographed by Lunar Orbiter I from 232,000 miles away. With the Moon in the foreground, this photograph closely duplicates the scene astronauts will view as they prepare to land on lunar surface. Center of Earth in the photo is Bechuanaland, South Africa. Lunar horizon spans about 450 miles and is 1500 miles away from Lunar Orbiter's camera.

Figure 2.-3: EARTH AS SEEN FROM MOON

Copernicus Crater, with mountains rising to approximately 3000 feet with slopes up to 30 degrees, was photographed by Lunar Orbiter II from an altitude of 28.4 miles and 150 miles away.

Figure 2.-4: COPERNICUS CRATER

f) Performed 284 attitude maneuvers, seven velocity maneuvers, and processed 3,571 transmitted commands.

g) Survived four known micrometeoroid impacts.

h) Changed its orbit inclination from 11.8 to 17.5 degrees to provide scientists with additional information about the Moon's gravitational field.

i) Transmitted 417 photographs to Earth.

MISSION III

L.O. III (spacecraft number 6) was launched February 5, 1967.

a) Photographed 12 potential Apollo landing sites selected from Missions I and II.

b) Photographed 31 secondary sites within the 21 degree orbit inclination on front and far sides of the Moon.

c) Performed the first comprehensive photographic site confirmation mission.

d) Positively identified the landing site of Surveyor III and photographed the Surveyor I landing site and spacecraft (Figure 2-5).

e) Provided data for lunar gravitational model for a 21 degree orbit.

f) Performed 383 attitude maneuvers, 7 velocity maneuvers, and processed 3,615 transmitted commands.

g) Completed the primary Apollo requirements for photographic information from an Orbiting Spacecraft. Coverage provided by L.O. III was:

- Prime sites - 2,200 square miles high resolution

  11,500 square miles wide angle

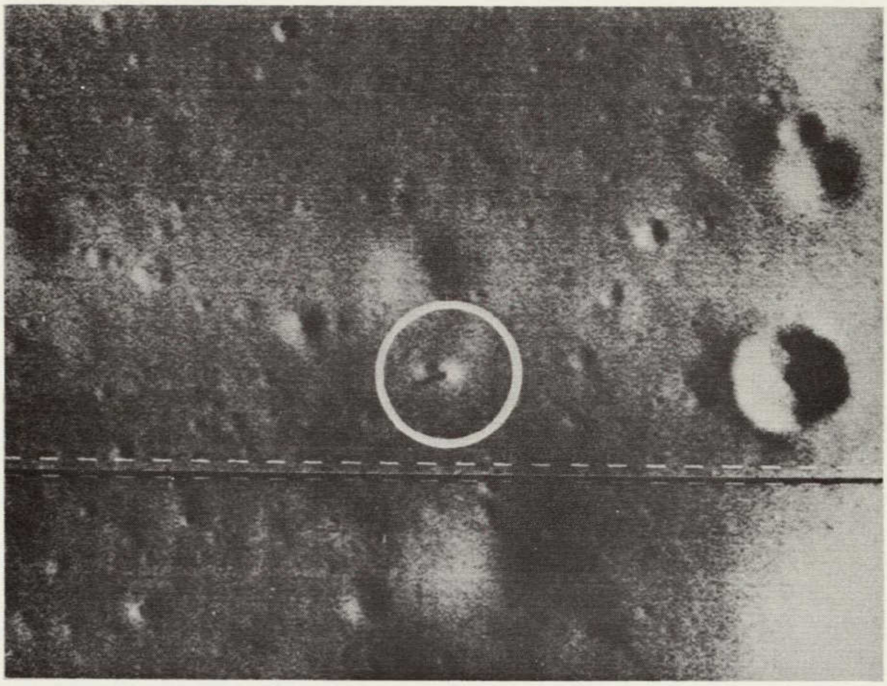
- Secondary Photography

  Near side - 350,000 square miles area of scientific interest

  e.g., Kepler (see Figure 2-6).

  Far side - 900,000 square miles.

h) Transmitted 327 photographs to Earth.



This 350- by 500-foot area was photographed by the high-resolution camera aboard Lunar Orbiter III. Surveyor I can be detected as a white object casting a shadow approximately 30 feet long. The spacecraft was located by triangulation of distant objects seen by Surveyor I cameras.

Figure 2.-5: SURVEYOR I LANDING SITE

Lunar Orbiter III's wide-angle lens captured this view of the lunar horizon. The most prominent crater La Kepler, approximately 20 miles in diameter and over 1 mile deep. The smaller, almost perfectly formed crater to the right is about 1/2 mile deep and 3 miles in diameter.

Figure 2.-6: CRATER KEPLER

MISSION IV

L.O. IV (spacecraft number 7) was launched May 4, 1967.

a) Completed first photographic mapping mission of Moon. Over 99 percent of near side (7,000,000 square miles) was photographed at resolutions of 60 to 100 meters.

b) Provided first photographs of surface detail for all limb and polar regions.

c) Provided first details of Orientale basin (Figure 2-7).

d) Provided trajectory data for 85 degrees orbit inclination to improve the determination of the lunar gravitational model.

e) Performed 586 attitude maneuvers, four velocity maneuvers, and processed 7,111 transmitted commands.

f) Provided data to relocate Mission V sites to increase the scientific data obtainable.

g) Survived two known micrometeoroid impacts.

h) Operated in orbits with full sunlight for long periods of time.

i) Transmitted 359 photographs to Earth.

MISSION V

L.O. V (spacecraft number 3) was launched August 1, 1967.

a) Completed photography of the far side and 45 near side photo sites to support numerous scientific requirements with resolution down to 2 meters. Figure 2-8 is from a wide angle view of the crater Tycho.

b) Provided a near full-Earth photo centered in Indian Ocean showing land mass outlines.

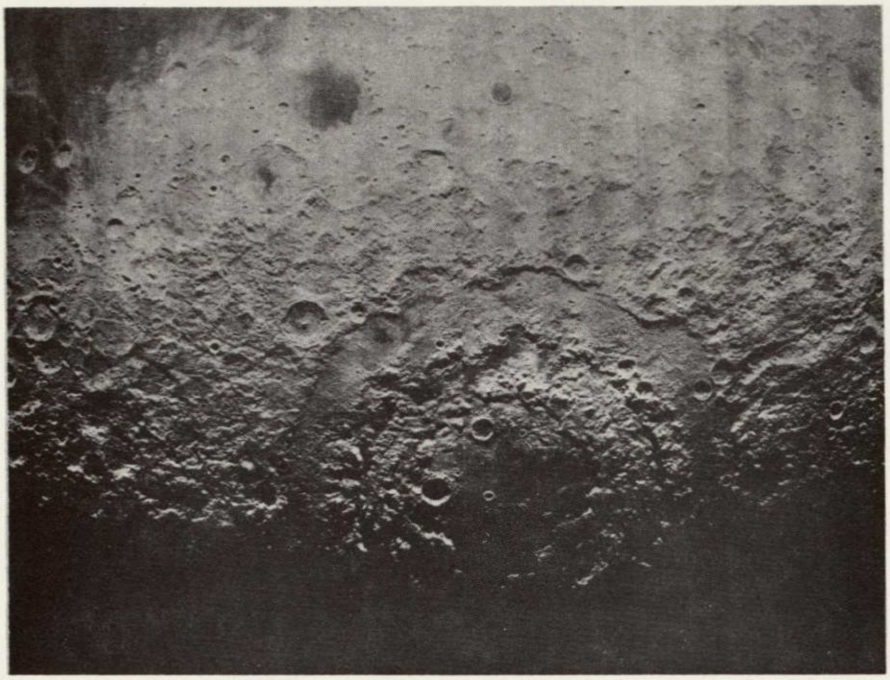
c) Performed 532 attitude maneuvers, six velocity maneuvers, and processed 4,525 transmitted commands.

d) Survived the six hour eclipse of October 1967 without difficulty.

e) Spacecraft was tracked optically in lunar orbit during a special experiment utilizing reflected sunlight directed toward Kit Peak Observatory.

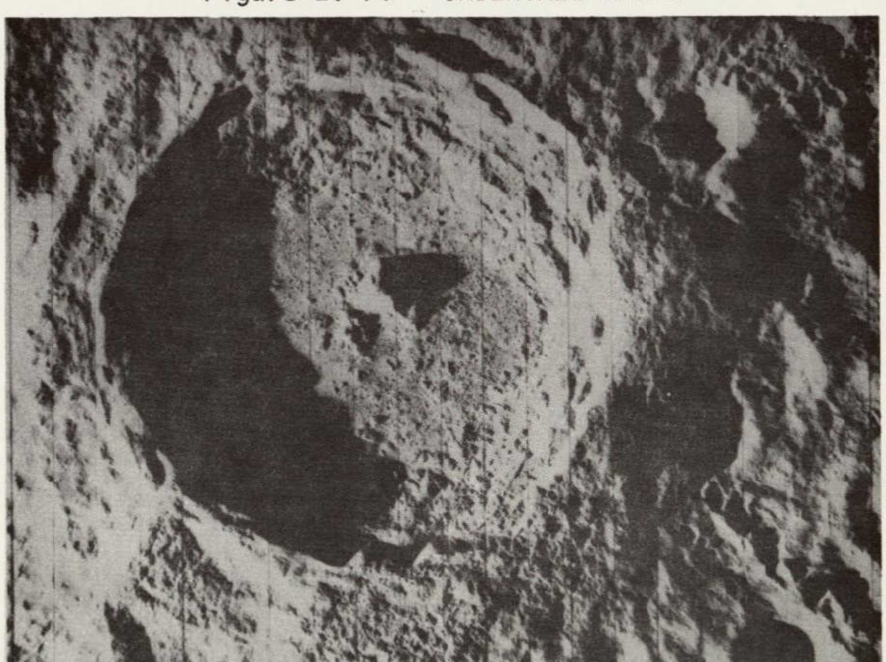
f) Provided low altitude high inclination data to improve the determination of the lunar gravitational model.

g) Transmitted 425 photographs to Earth.



This photograph was taken by the Lunar Orbiter IV at an altitude of 1690 miles and shows the enormous complex called the Orientale Basin. The Cordillera Mountains which form the outer scarp are more than 600 miles in diameter and rise to an altitude of 20,000 feet above the surrounding surface. Orientale is believed to be the most recent of all the Moon's large circular basins.

Figure 2.-7: ORIENTALE BASIN



Tycho, a very young, large crater, was photographed in this crisp detail by Lunar Orbiter V from an altitude of 135 miles. The crater, which is more than 50 miles from rim to rim, is considered by astronomers and scientists to be quite young because the debris associated with its formation is superimposed on older topographic features over a large area of the Moon's visible side.

Figure 2.-8: Tycho

PROBLEM AREAS

Problems experienced during the program are summarized in the following paragraphs.

a) Launch of L.O. I was delayed one month due to the unacceptable photo subsystem. During the second launch period launch was delayed one day by questionable launch vehicle instrumentation.

b) Major failures affecting photo data recovery were:

L.O. I - Telephoto images degraded because focal plane shutter control circuitry was susceptible to electromagnetic interference. Corrected for missions II through V.

L.O. II - TWTA failed to turn on 12 hours to planned end of mission. 98.5 percent of all photos had been read out.

L.O. III - Film advance motor failed after reading out 73.8 percent of the 211 dual exposures taken.

L.O. IV - Film advance problems necessitated early Bimat cut and the loss of approximately 20 planned photos of areas beyond the western limb.

L.O. V - None.

c) Operational problems encountered were:

1) Star Tracker Glint - L.O. I through V

- Stray light resulted in erroneous signals to star tracker.
- Satisfactory operation wás possible during solar occultation.
- Periodic attitude update maneuvers were required.
- Error signals from star tracker were used by subsystem analyst to determine and predict roll position.

2) Spacecraft Thermal Control - L.O. I through V

- Equipment mounting deck thermal paint degraded from solar energy.
- Spacecraft was pitched off sunline for extended periods to provide spacecraft temperature control.
- Mirrors were added for Missions IV and V to reduce thermal energy exposure.

3) Intermittent Film Hangup - L.O. III

- o Readout of a single framelet was repeated.

- o Real time commands were employed to control and reinitiate readout when hangups occurred. Hangup was attributed to interference between teflon separators and readout looper rollers.

4) Film Advance Control - L.O. IV

- o Intermittent film advance logic signals caused readout and processing problems.

- o Manual commands were employed to set up special logic control configuration to allow specific functions and inhibit other functions.

- o Real time commands were used to set up automatic reinitiation of readout mode at end of each scan cycle.

5) Camera Thermal Door - L.O. IV

- o Door failed to open on command during first and second photo orbits.

- o Real time commands were used to control door operation from partially closed to full open.

6) Photo Subsystem Window Temperature - L.O. IV

- o Condensation developed on window when camera thermal door was left open following above problem.

- o Spacecraft was oriented so that solar energy would maintain window temperatures. (Not successful.)

- o Direct light on window penetrated light baffles and light streaked film.

- o Partially closing door gradually eliminated all problems.

7) Loss of Photo Coverage - L.O. IV

- o Light streaking of film degraded photos of eastern areas.

- o Mission plans were altered to rephotograph areas by apolune photography near end of mission with some reduction in resolution capability.

8) Communications Subsystem - L.O. II, III, and V

Unintentional commands were received, transferred to the programmer, and executed by L.O. II and III during the primary mission of Spacecraft L.O. III, IV, and V. These commands occurred with the other spacecraft transponder in uplink lock under weak uplink signal conditions. Bit errors developed in the address code such that commands addressed to another spacecraft would be accepted and executed.

9) Extended Mission Attitude Control - L.O. V

Loss of 2.5 pounds of nitrogen gas occurred during a weekend period when the spacecraft was not being monitored by the SFOF and required that the mission be terminated several months earlier than planned.

SPACECRAFT CONFIGURATION

The spacecraft configuration is illustrated in Figure 2-9. The Lunar Orbiter weighed 850 pounds and when prepared for launch measured five feet in diameter by five and one-half feet high. During launch the solar panels were folded against the base of the spacecraft and the antennas were folded against the sides of the structure. When the solar panels and antennas were deployed in space the maximum span became seventeen and one-half feet across the antenna booms, and twelve feet across the solar panels.

The primary structure consisted of the main equipment mounting deck and an upper section supported by trusses and an arch. Located in the upper section were the velocity control engine with its tanks for oxidizer, fuel and pressurant, and the attitude control thrusters, remote sun sensors, and the switching assembly. The nozzle of the engine extended through an insulated heat shield. The equipment mounting deck supported the camera, communications equipment, electrical system equipment, and the following G&C components: inertial reference unit, sun sensor, star tracker, and the flight electronics control assembly.

The electrical power system was a solar panel rechargeable battery system with voltage regulation and charge control. In full sunlight the solar panels would produce about 375 watts; the battery was rated at 12 ampere hours. The electrical system voltage would vary from 22 volts when the battery was supplying the load to 31 volts when the solar panels were operating.

The communications system was compatible with the Deep Space Network S-band system. It received, decoded, and verified commands sent to the spacecraft from earth, and transmitted to earth all data gathered by the Spacecraft.

A low power operating mode delivered spacecraft performance telemetry and data from the lunar environment experiments (radiation and meteoroids) to Earth at 50 bits per second. Telemetry was in digital form, and was passed through a signal conditioner, a multiplexer encoder and a modulation selector before transmission.

A high power communication mode was used to transmit photographic data in analog form and brought into use the spacecraft's high gain antenna and a travelling wave tube amplifier. Performance and environmental telemetry were mixed with the photographic information in the transmission.

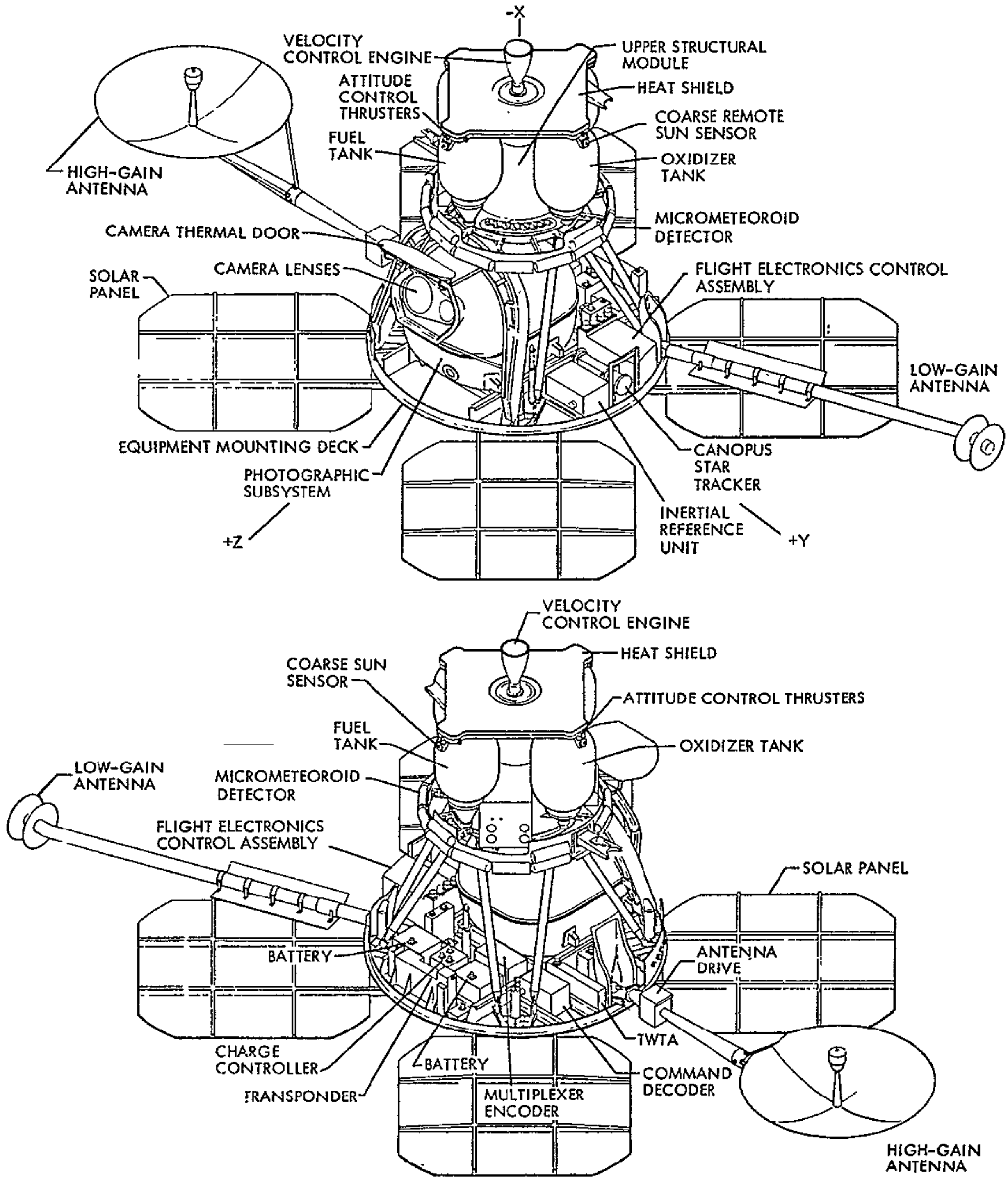


Figure 2-9: LUNAR ORBITER SPACECRAFT

In the tracking and ranging mode, the transmitting frequency of the transponder was locked to the frequency of the signal being received from Earth in a precise ratio. The signals were then used to determine the radial velocity of the spacecraft to an accuracy of about one foot per second. When interrogated by the Deep Space Network ranging system, the transponder signal measured the distance between the Earth and the spacecraft with an accuracy of about 100 feet. This was an important part of the total G&C function and the data obtained in this mode was used to determine the necessary commands required to perform midcourse corrections and orbit insertion.

The transponder received a transmitted command from Earth and passed it to a decoder where it was stored temporarily. The command was then re-transmitted to Earth through the transponder to verify that it had been correctly received. When verification was confirmed, an execute signal was sent from Earth causing the decoder to pass the command along for immediate or later use as required. The command transmission rate was 20 bits per second.

Spacecraft temperature control was provided to maintain desired component temperatures. An aluminized mylar thermal barrier enclosed the area between the insulated upper heat shield and the equipment mounting deck. Indium foil .005 inch thick was used between high heat dissipating components and the equipment mounting deck to provide a known heat transfer coefficient. The interior of the spacecraft was tied together thermally and the primary means of heat transfer was the equipment mounting deck, which faced toward the sun when the spacecraft was in a sun reference attitude. The equipment mounting deck was coated to obtain a high heat emission-absorption ratio to provide an equipment deck temperature between 85 and 35 degrees F when in orbit about the Moon.

GUIDANCE AND CONTROL SYSTEM

The Lunar Orbiter G&C system is illustrated in the functional block diagram of Figure 2-10. The primary functions performed by the G&C system were attitude control, velocity control, and sequencing and timing of spacecraft events.

The system provided three axis stabilization and control of the spacecraft, utilizing a design concept that simplified the spacecraft functions and equipment by placing greater emphasis upon Earth-based facilities. The total G&C function was accomplished by the spacecraft G&C elements working with the Deep Space Network, Space Flight Operations Facilities, and the flight operations team through the communications subsystem. A key factor in this implementation was the ability to utilize the decision making ability of the flight operations team. The twofold result of this system approach was first the realization of a spacecraft system that was simple, reliable and straightforward to operate, and secondly a flexible G&C system that could accommodate both a wide variety of mission tasks and flight anomalies.

The guidance and control system was composed of the hardware items below:

1) Flight Electronics Control Assembly - all parts of the G&C system were interlinked by this unit. It consisted of two distinct functional sections, a programmer and closed-loop electronics. The programmer

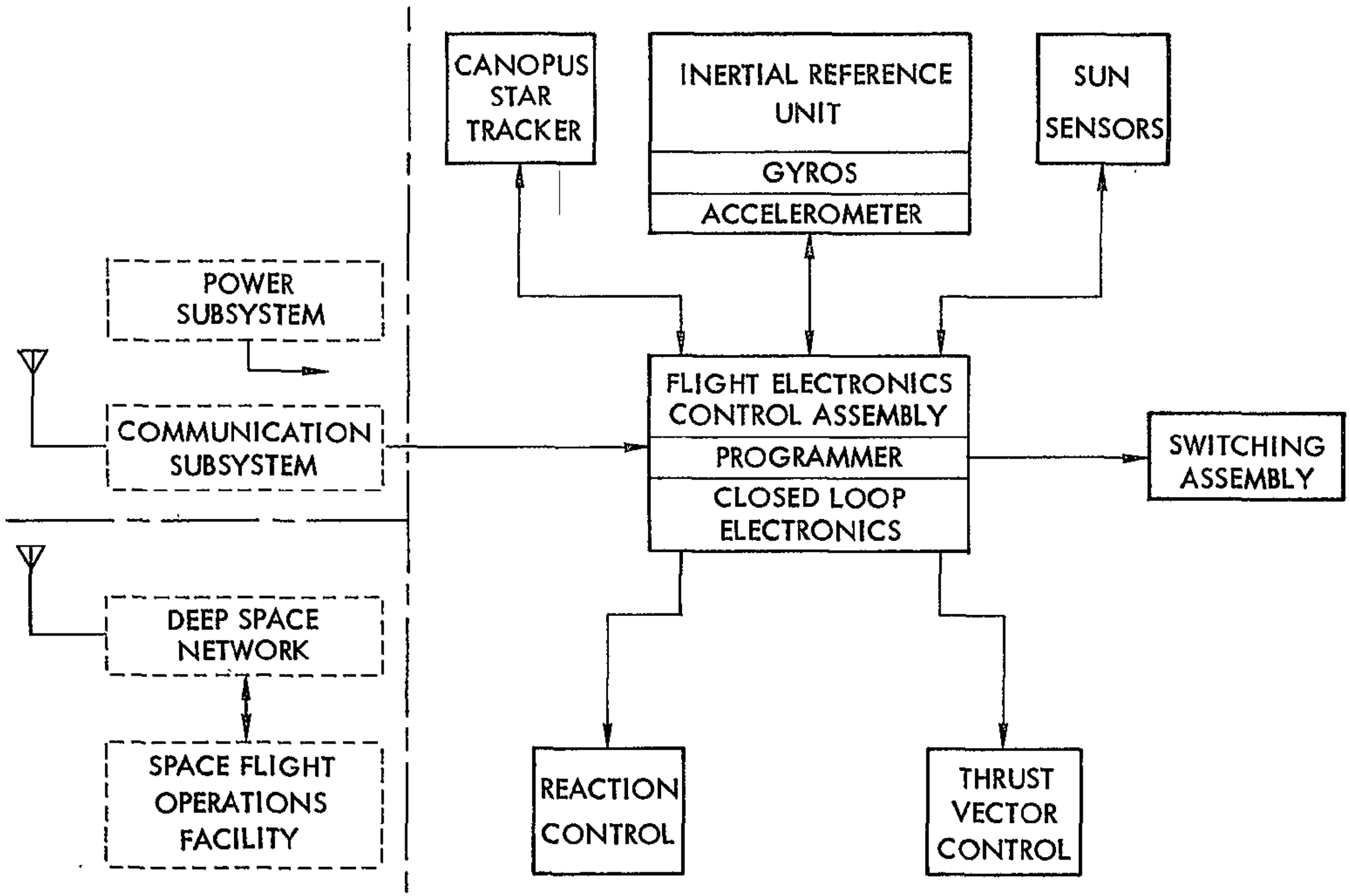


FIGURE 2-10 GUIDANCE AND CONTROL SYSTEM

was a low-speed digital data processor, with memory capacity large enough to provide sixteen hours of control from stored commands. It controlled all timing and sequencing of spacecraft events. It integrated the maneuver magnitude signals from the gyro or accelerometer. The closed loop electronics performed the tasks of attitude control. It was basically all analog in operation, accepting error signals from the sensors and gyros, and controlling the reaction thrusters and thrust vector control actuator upon command from the programmer.

2) Inertial Reference Unit - this unit consisted of three single-degree-of-freedom, floated, rate-integrating gyros, and one pulsed integrating pendulum-type accelerometer. The gyros operated in either of two modes selected by the flight electronics control assembly; rate mode or rate-integrate mode. The accelerometer was on but readout only during velocity maneuver.

3) Sun Sensors - there were five sensors on the spacecraft that provided 4π steradian coverage to assure sun acquisition and maintain pitch and yaw attitude. Four remote coarse sensors were located at the corners of the heat shield; the fifth, a combination coarse and fine sensor viewed through the equipment mounting deck.

4) Star Tracker - the celestial reference for the spacecraft roll axis was provided by this unit. It produced a roll error signal and a star map voltage that was transmitted via telemetry. The flight operations team interpreted the star map to provide assurance that the tracker was looking at Canopus rather than some other star.

5) Reaction Control - eight reaction control thrusters using nitrogen gas generated the torque necessary to control the spacecraft in roll, pitch or yaw. Approximately 14 1/2 pounds of nitrogen were stored in a titanium sphere at a pressure of 3500 psi. Ten pounds were budgeted for attitude control, and four were used for pressurizing the fuel and oxidizer tanks.

6) Thrust Vector Control - spacecraft attitude in pitch and yaw was maintained during velocity maneuvers by a gimbaled engine. The engine was positioned by two actuators responding to error signals from the gyros.

These six items of the G&C system were interconnected by the flight electronics control assembly to form four functional subsystems. Many of the requirements and much of the functional development was accomplished on this subsystem basis, even though some elements were not exclusively a part of only one subsystem. The four subsystems and the items that composed each one, were as follows:

1) Attitude Control Subsystem - including the programmer, sun sensors, star tracker, closed loop electronics, and inertial reference unit.

2) Reaction Control Subsystem - included the programmer, reaction thrusters, regulators, N_2 supply, and plumbing.

3) Thrust Vector Control Subsystem - included the programmer, thrust vector actuators, closed loop electronics, and inertial reference unit.

4) Velocity Control Subsystem - included the programmer and accelerometer.

Section 3.0 through 6.0 discuss each of these subsystems individually.

GUIDANCE & CONTROL SYSTEM DEVELOPMENTAL GUIDELINES

The development guidelines which had a major impact on the spacecraft and the Guidance and Control system are listed in the following paragraphs. The overall program schedule with selected emphasis on Guidance and Control events is shown in Figure 2-11. The major events recur in discussions of particular development problems.

1. Compatibility between the spacecraft and the Atlas Agena booster was required. Most important effects were the launch envelope allowed and the booster environment -- acceleration, vibration and temperature.

2. Use of off-the-shelf, space proven, and high reliability hardware wherever possible was a program requirement coupled with a thorough quality assurance program from the program start. All modifications to existing hardware were analyzed and tested as carefully as though it were a new design.

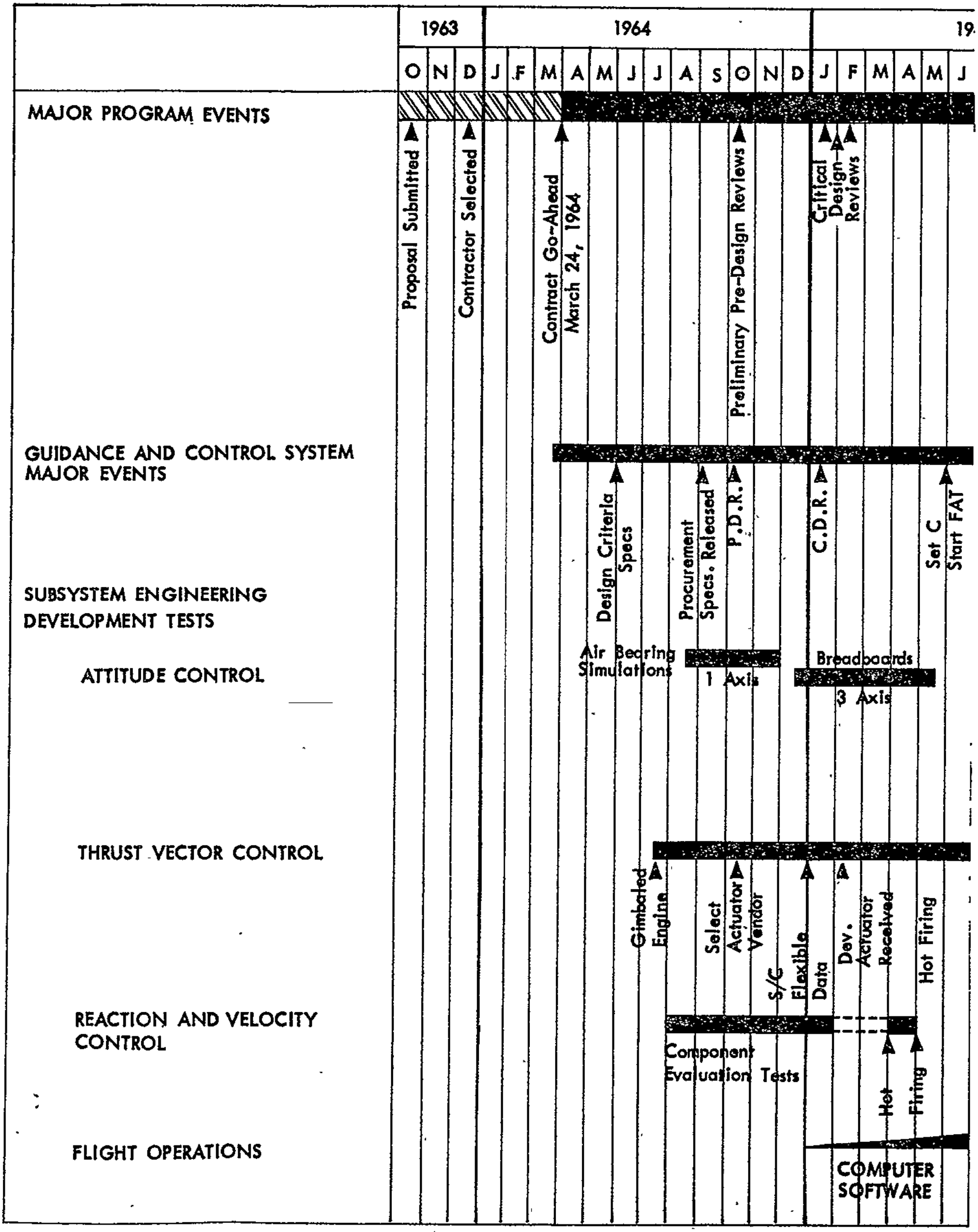
3. A comprehensive test and qualification program was required that encompassed all levels of test beginning with fabrication testing and concluding with spacecraft launch and checkout. This included not only functional and environmental test of all flight hardware, but development, qualification, and reliability demonstration tests of components, subsystems, systems and the entire spacecraft.

4. In addition to providing flight performance information in real time, it was an objective that spacecraft performance telemetry be sufficient for monitoring all subsystem and component performance to a level adequate to isolate problems to the black box replacement level for all spacecraft level tests.

5. Modular design was provided as illustrated by the propulsion and reaction control assembly. The advantage was in being able to assemble, test, and service the module without later changes to the plumbing required by a higher assembly level.

6. An unwritten but often quoted philosophy when choosing between alternatives was: "Put the intelligence on the ground." The objective was to keep the spacecraft and its functions simple, by assigning the flight operations team the responsibility of decision making.

7. The spacecraft reliability goal for the successful completion of a 30-day mission was 0.70 based on a realistic single thread assessment of state-of-the-art hardware. The reliability allocation to the guidance and control system was .83.


1963
1964
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O N D J F M A M J J A S O N D J F M A M J
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Proposal Submitted
Contractor Selected
Contract Go-Ahead March 24, 1964
Preliminary Pre-Design Reviews
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GUIDANCE AND CONTROL SYSTEM MAJOR EVENTS
Design Criteria Specs
Procurement Specs. Released
P.D.R.
C.D.R.
Set C Start FAT
SUBSYSTEM ENGINEERING DEVELOPMENT TESTS
ATTITUDE CONTROL
Air Bearing Simulations
1 Axis
Breadboards
3 Axis
THRUST VECTOR CONTROL
Gimbaled Engine
Select Actuator Vendor
S/C Flexible Data
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REACTION AND VELOCITY CONTROL
Component Evaluation Tests
Hot Firing
FLIGHT OPERATIONS
COMPUTER SOFTWARE

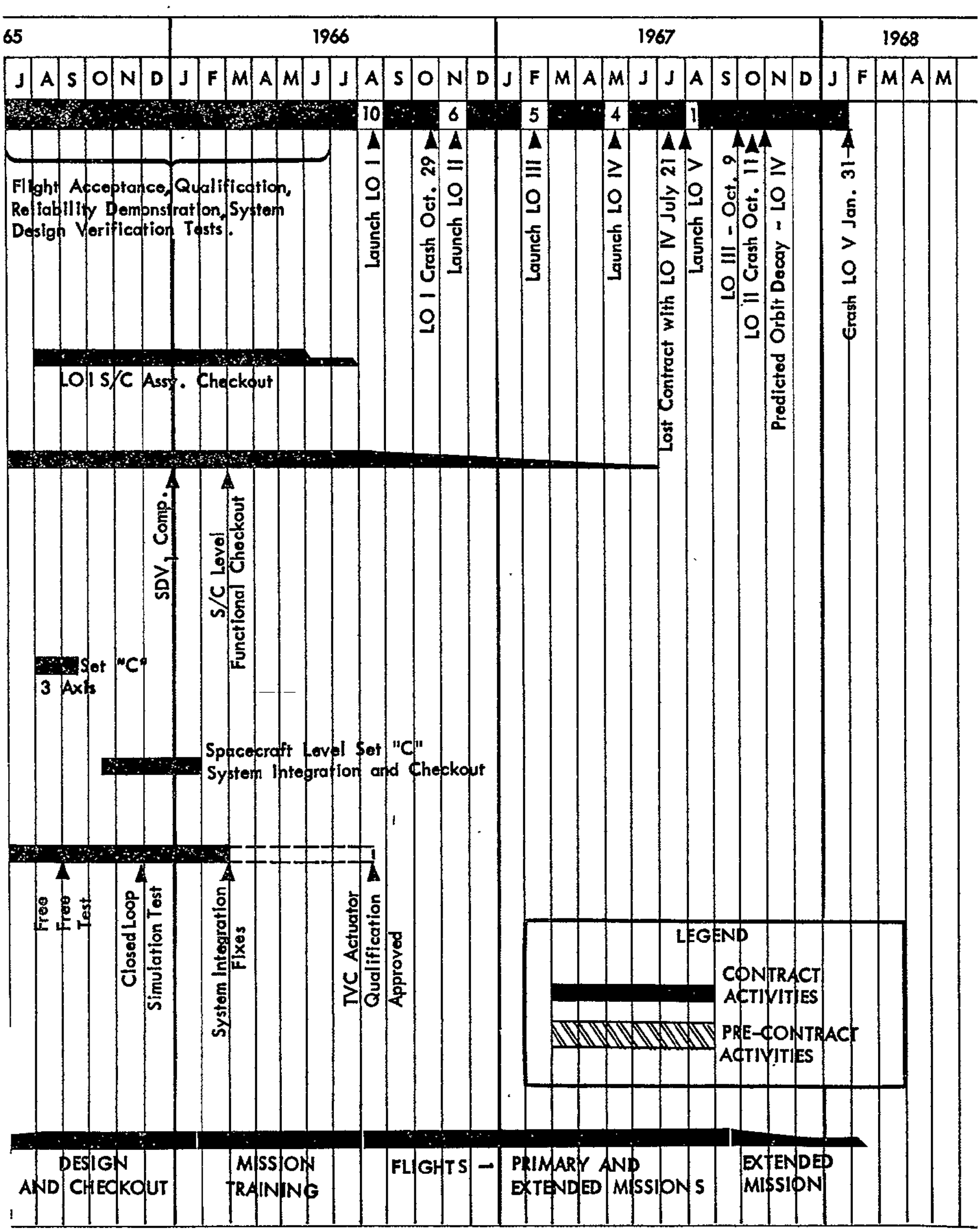


FIGURE 2-11 LUNAR ORBITER PROGRAM SCHEDULE

8. The spacecraft was to remain in orbit after completion of the photo-mission with suitable equipment working for an extended lifetime to obtain additional selenodetic and environmental data. The lifetime goal was one year.

GUIDANCE AND CONTROL SYSTEM DESIGN CONSTRAINTS

The system requirements were:

1. To acquire the sun within 20 minutes from staging from the launch vehicle and to maintain a sun orientation for power and thermal control.

2. To establish and maintain a Sun-Canopus reference attitude of the spacecraft anytime after six hours from injection on translunar trajectory.

3. To perform up to two midcourse velocity maneuvers as required.

4. To perform velocity changes to inject into initial and final orbits about the moon. The three sigma inaccuracy allowed for velocity maneuvers was not to exceed .5 feet/second plus 0.0008 times ΔV feet/second.

5. Perform maneuvers in space from the reference attitude for photography, velocity change, or communication with reorientation error not to exceed .75 percent of the commanded maneuver. Maneuvers about any axis of at least 180 degrees were required. Return to the reference attitude was required.

6. To maintain the proper attitude during engine firing.

7. Point the high gain antenna at Earth within the beam width of the antenna for the 30-day mission.

8. To provide sequencing and timing of all spacecraft events, with a timing resolution of 0.1 second.

9. To provide at least 5 percent side overlap between pictures taken on successive orbital passes.

3.0 ATTITUDE CONTROL SUBSYSTEM

The Attitude Control Subsystem (ACS) was one of four subsystems that constituted the spacecraft G&C system. The relationship of the ACS to the total G&C system is illustrated in Figure 3.1.

3.1 SUBSYSTEM DESCRIPTION

The ACS subsystem was basically a rate limited, conventional on-off, cold gas reaction control. A basic reference attitude was established by a Sun sensor and star tracker. A rate damping signal was provided by a rate gyro for each axis during normal celestial hold operation with the Sun sensor (Pitch and Yaw) or star tracker (Roll). During Sun and Canopus occultation periods attitude was maintained by the gyros in the rate integrate mode. Lead-lag networks provided stability.

The functions of the attitude control were to:

1) Acquire, stabilize, and maintain a Sun orientation of the spacecraft in pitch and yaw.

2) Acquire, stabilize, and maintain a Canopus referenced orientation in roll.

3) Maneuver sequentially one axis at a time away from celestial references as desired, e.g., for photographic or velocity change purposes.

4) Hold attitude to inertial references as required, e.g., for photo sequences, for velocity changes, and for periods of sun and Canopus occultation.

5) Return spacecraft to celestial reference orientation upon completion of above maneuvers.

6) Point the high gain antenna.

7) Provide delta velocity measurement and control for engine burns.

The attitude control block diagram for one axis is shown in Figure 3-2. The design features of the hardware components are discussed as individual topics in paragraph 8.0. The modes of attitude control operation were controlled by the flight programmer shown on the left side of Figure 3-2.

The center portion of Figure 3-2 was called the closed loop electronics portion of the Attitude Control subsystem and was physically located in the Flight Electronics Control Assembly. The remaining elements on the right of Figure 3-2 are the sensors, torquers and spacecraft dynamics.

Each axis was controlled independently (except that all deadbands were switched together and sun sensor select operated both pitch and yaw). The operating modes provided for limited rate about any axis, attitude hold to celestial and inertial references, maneuvers of precise magnitude, and logic to provide for occultations. These modes will be described in the following paragraphs, using the sequence of a normal mission.

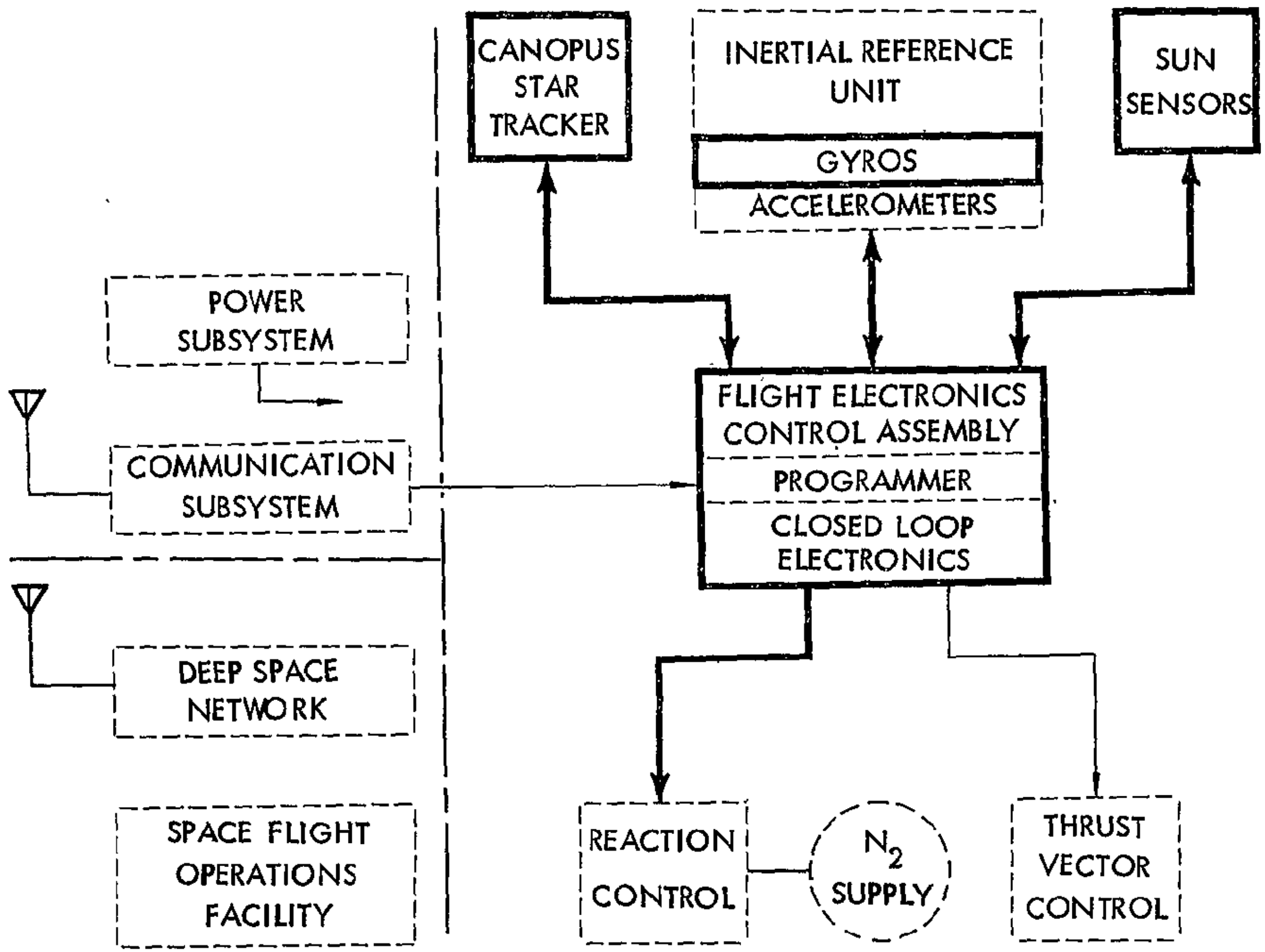


FIGURE 3.-1 G & C ATTITUDE CONTROL SUBSYSTEM

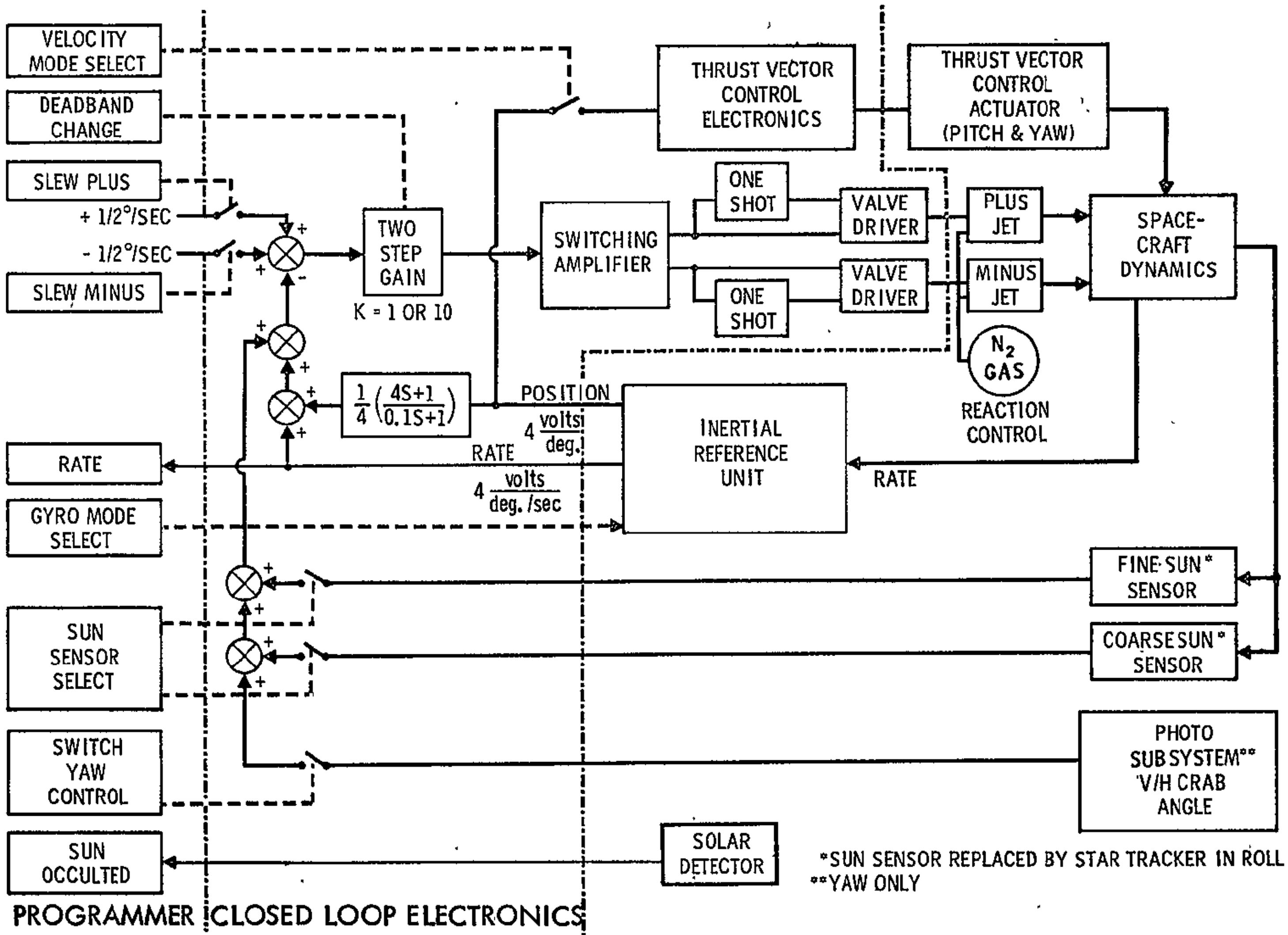


FIGURE 3-2 ATTITUDE CONTROL SUBSYSTEM BLOCK DIAGRAM

3.1.1 Sun Acquisition

The initial Sun Acquisition was accomplished according to Figure 3-3. For this initial acquisition, the sun sensor field of view was 4π steradian. The output from the sun sensor amplifier was limited with a soft limiter. This attitude error limiter had the effect of a rate limit on the system for large errors. Depending upon the deadband chosen, the rate range was nominally 0.55 $\pm$ 0.5 degrees per second for the 2.0 degree deadband and 0.55 $\pm$ 0.05 degrees per second for the 0.2 degree deadband. The time allowed for the sun acquisition, after Agena separation was limited to 20 min. by battery capacity. The narrow deadband was chosen for this task to assure that the acquisition rate would be at least 0.5 degrees per second. During Sun acquisition, the roll axis was operated in a rate limited mode, with the Canopus tracker "OFF" until after the spacecraft had passed through the Van Allen radiation belts.

3.1.2 Canopus Acquisition

Approximately six hours after launch, the Canopus Tracker was turned on. The planned Canopus Acquisition sequence shown in Figure 3-4 was initiated. A 360 degree maneuver was commanded during which time a star map was telemetered to the ground. This star map was compared to a computer calculated a priori star map from which the roll angle of the spacecraft relative to Canopus at the end of the 360 degree maneuver was defined. A second maneuver was commanded to orient the spacecraft with as small a roll error relative to Canopus as possible. The tracker switched from search to track and issued a "Canopus present" signal. The third maneuver in the sequence was an Acquire Canopus command which switched the control of roll error to the tracker.

3.1.3 Attitude Maneuver

The Lunar Orbiter maneuver sequence was different from any method used up to that time in a spacecraft. The sequence is outlined in detail in Figure 3-5. For the maneuver sequence, the deadbands for all axes were first narrowed to 0.2 degrees. This provided tight control on the vehicle rate during the maneuver and also guaranteed the vehicle position to be within 0.2 degrees of null at the start of the sequence. After a wait time of 51.2 seconds to stabilize in the new limit cycle, the maneuver was initiated by simultaneously applying a 0.55 $\pm$ 0.05 degree per second slew voltage to the maneuver axis, and switching the IRU rate signal to a voltage to frequency pulse converter which, in turn, was connected to the Programmer's pulse counting register. The control system nulled the slew voltage and rate gyro signal by firing the proper jets to bring the S/C to the commanded rate. The integration register in the programmer would count the output pulses of the V/F converter and compare the accumulated count with that stored in the memory for the commanded maneuver. When the maneuver was completed, the slewing signal was removed, and the gyro switched to the integrating mode. This fixed the null of the gyro to the desired end point of the maneuver. The vehicle was controlled during the maneuver hold period with jet pulses commanded by the inertial hold lead-lag compensation network operating on the attitude error output from the gyro.

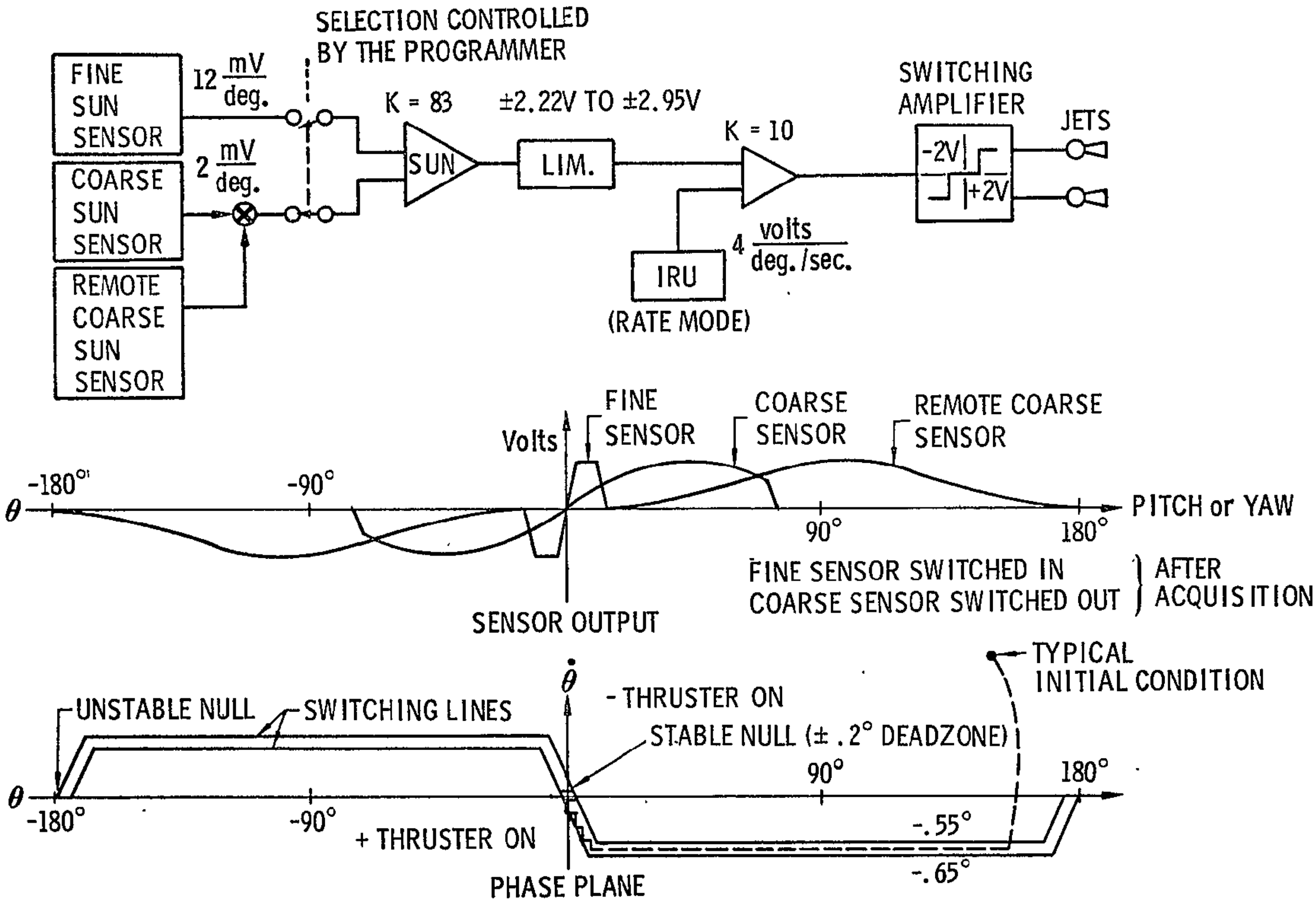


FIGURE 3-3 SUN ACQUISITION

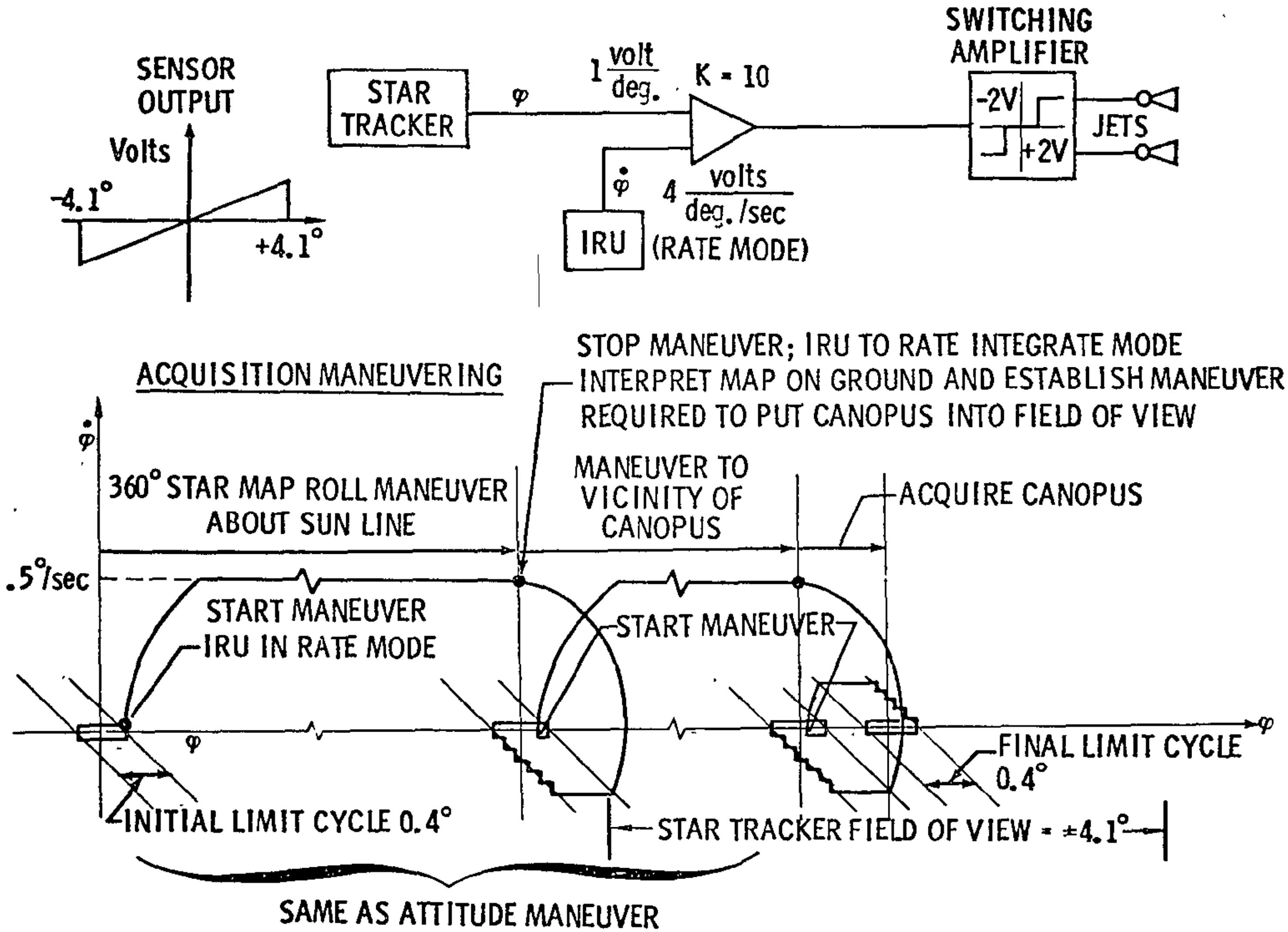


FIGURE 3-4 CANOPUS ACQUISITION SEQUENCE

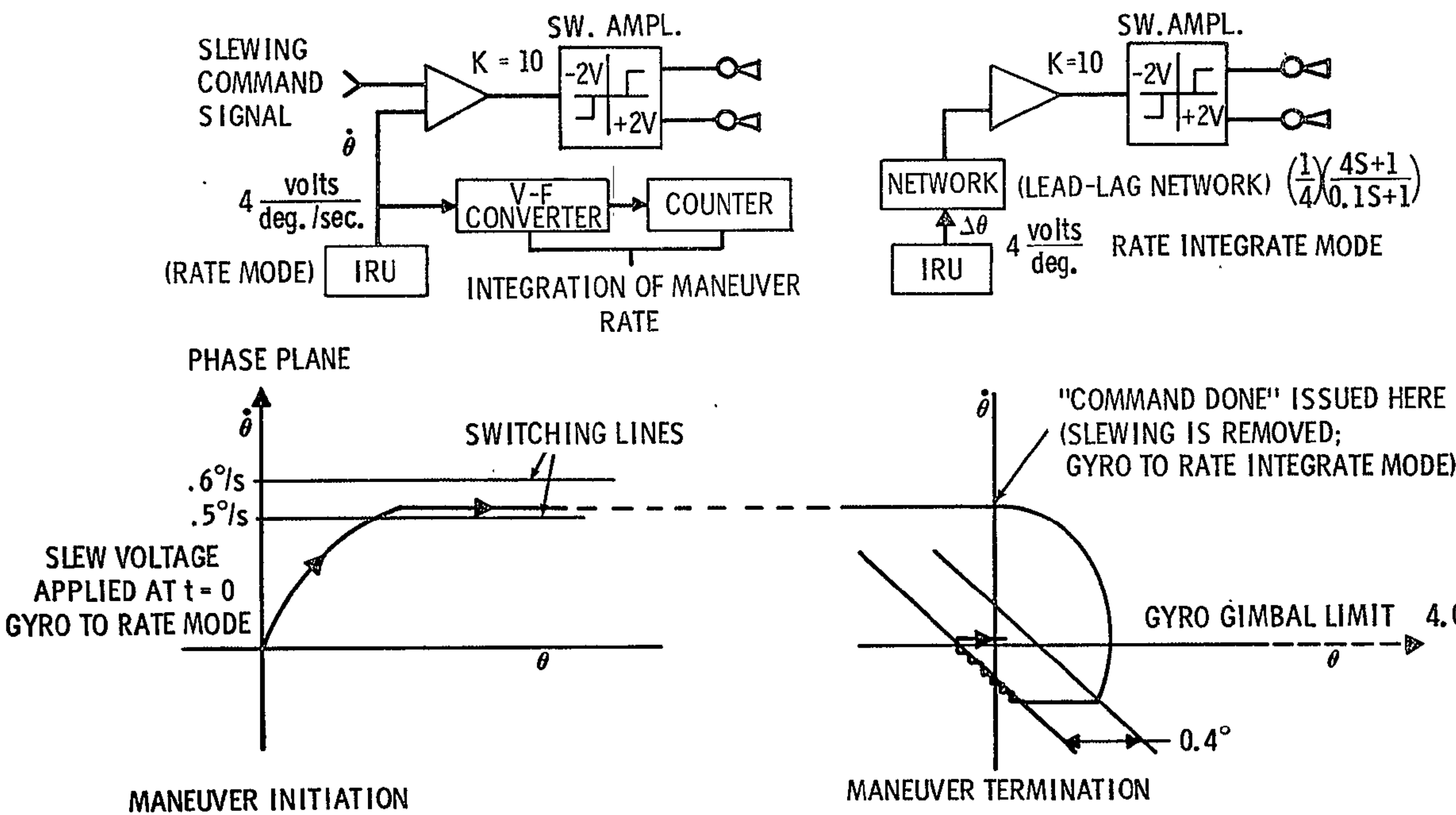


FIGURE 3-5 ATTITUDE MANEUVER

3.2 DESIGN REQUIREMENTS

The design requirements of the attitude control subsystem were developed from a nominal 30 day photo mission with a one year extended life capability. Figure 3-6 correlates the requirements with mission events. A qualitative discussion of the design features required of the attitude control subsystem is listed below. A discussion of component tolerances is contained in the respective component sections.

DESIGN FEATURE REQUIRED	BASIS OF REQUIREMENT
1. Three axis stabilized configuration.	To obtain the required 1 meter resolution from orbit demanded the choice of a photographic camera to obtain, store, and play back the topographic information rather than by a television system. Image smear during the photo exposure time was of prime concern. A spin or dual spin stabilized configuration would have been more complex to compensate for image motion.
2. Conventional ON-OFF Reaction Control but with "One Shot" on jet valve driver.	Cold gas N_2 reaction control was proven for long reliable life by JPL Interplanetary Spacecraft (Mariner). One shot for minimum impulse bit was selected to suit good performance state of art valves. Boeing laboratory experience was source of one-shot feature.
3. Inertial reference unit with rate and rate integrate modes from gyros.	Unique orbiter requirement was to hold reference attitude with either Sun or Canopus occulted. Accurate rate information was required to maneuver, to stop initial tumbling, and for damping with celestial sensor position references.
4. Celestial sensors: a) Fine sun sensor, coarse sun sensor, aft remote sun sensors.	Complete 4π steradian coverage was provided to assure sun acquisition. Coarse and remote eyes were switched out on command to prevent pointing errors from secondary radiation sources such as the moon or earth.
b) Canopus Star Tracker with magnitude telemetry and presence gates at 1/3 to 3X Canopus intensity.	Canopus is second brightest star; only 14.5° away from South ecliptic pole, with no other bright stars nearby. Therefore a star map generated by rolling 360° about sun line would provide unique Canopus identification by comparison with a priori star map. Thus a unique celestial referenced coordinate system is established by a line to the Sun (for pitch and yaw) and to Canopus (for roll).

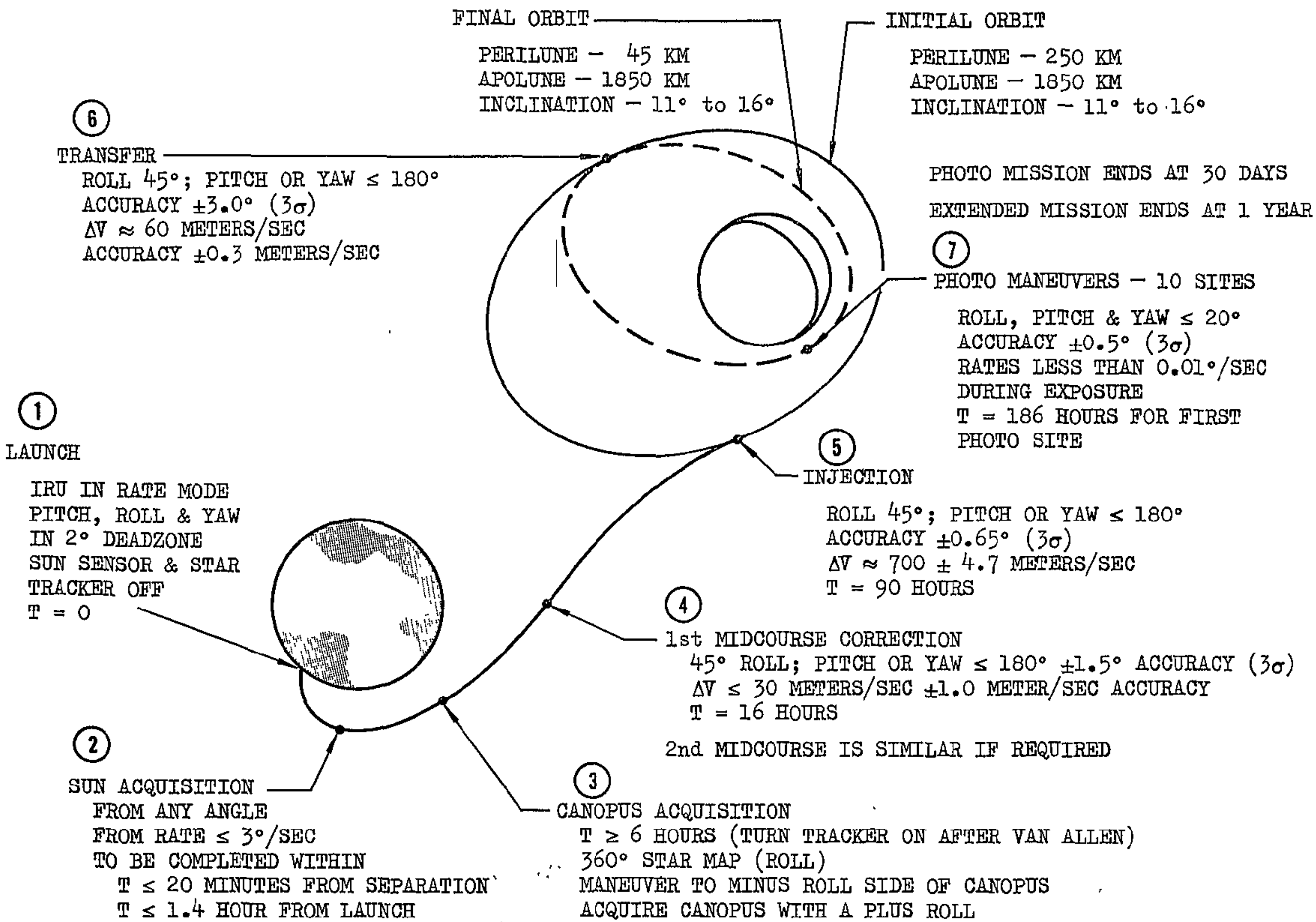


FIGURE 3-6 ATTITUDE CONTROL REQUIREMENTS

5.	Wide and narrow deadzone provided by "two step gain" change on switching amplifier (Gain of 1 or 10) as commanded by the Flight Programmer.	Accuracy for photo and delta velocity maneuvers demanded deadzone no greater than 0.2°; to prevent large limit cycle propellant penalty during coast periods required 2.0° or larger deadzone. Gain change method retained switch line slope with rate to position ratio of 4:1 and an overdamped response for propellant conservation.
6.	Lead-lag derived rate from inertial reference unit in the rate integrate mode with passive R-C networks.	Transfer function selected to provide near optimum time response for convergence from maneuver rate of $.5^{\circ}$/sec and be overdamped at lesser rates for propellant conservation.
7.	Slew plus or minus (for attitude maneuvers at 0.5°/sec.	To obtain photos at proper surface illumination angle following occultation (A.M. photography) required the maneuver rate to be approximately $.5^{\circ}$/sec. Other maneuvers could have been at lower rate. Both plus and minus maneuvers required to minimize total maneuvered angle and give most accurate orientation.
8.	Switch yaw control to photograph V/H crab angle control.	This feature, though provided, was never used in flight operations because it was not needed and did not work very well because of noise. Early requirement was based on anticipated inability to predict orbit track and spacecraft attitude to prevent lateral image smear as would result from yaw angle (crab) errors greater than 0.5°.
9.	Solar Detector (Solar panel array voltage).	A sun "not present", as indicated by lack of solar array voltage, initiated switching logic to place IRU pitch and yaw axes into rate integrate mode so as to hold inertial attitude during sun occultation periods. A similar logic was provided by Canopus "not present" signal.
10.	Provide attitude control for primary mission and extended mission with 10 LB of N_2 gas.	Weight limit of spacecraft limited control gas available.
11.	1.0 Volt per degree and 4 volts per degree per sec. for sensor scale factor.	Selected on basis of electrical noise level expected with solid state electronics and to make noise effect small on a $.2^{\circ}$ deadzone.

12. Only solid state switches used throughout this subsystem.	Reliability for attitude control subsystem could not be achieved with relays, reliability required was .88 for 30 day mission.

3.2.1 Gas Budget Requirements

The impulse requirements for attitude control were determined by accounting for every detail item of impulse. These were grouped in the following 5 major categories:

1. Major rate increments

   a) Initial damping of booster tip off rates
   b) Initial Sun and Canopus acquisition
   c) All maneuvers; midcourse, injection, orbit change and photo maneuvers.

2. Limit cycle

   Deadzone width and time were the variable factors. Because of the uncertainty in ability to obtain limit cycle rates of .001°/sec. corresponding to a single 11 millisec. pulse, .0025°/sec. was arbitrarily assigned as the minimum predicted rate.

3. Disturbances

   a) Solar pressure acting to produce torque over the mission time was determined.
   b) Gravity gradient torque time history was determined.
   c) The net disturbance torque was integrated over the mission time to give the impulse requirement.

4. Celestial reference reacquisition following an occultation

   The consequence of occultation and re-appearance of the celestial references is a unique problem of an orbiter. Reacquisition may impose severe gas requirements if the design fails to anticipate the problem.

5. Leakage

   This is the total system leakage and was treated as a random quantity producing no net torque on spacecraft.

   The conservative way of obtaining gas requirements is to simply add them up for each individual axis as impulse requirements. To the extent that limit cycle and disturbances strongly interact this is an over simplification of the problem. The experience of Lunar Orbiter was that gas usage was less than the simple addition of predictions for limit cycle and disturbance.

3.3 DEVELOPMENT AND OPERATION

There were no changes to the attitude control hardware design for any of the 5 flight spacecraft. This was due in part to the maturity of the design as a result of a thorough developmental and qualification test program; also due to the ingenuity of the designers, analysts, test engineers, and flight operations people in devising work-around methods or in exploiting inherent capability of the existing hardware. To expose this information in an orderly manner the listing of design changes will be discussed for periods between major mileposts related to the program schedule.

3.3.1 Proposal Submittal to Contract Award

One change to the spacecraft mission which had a major impact on the attitude control was to be capable of continuing in operation for one year. The primary photographic mission required approximately one calendar month. The extended life goal of eleven additional months was desired to provide extended periods of time for obtaining lunar environment and selenodetic data by means of tracking through the DSIF. The proposed design solution was to simply allow the spacecraft to continue in operation in much the same manner as for the primary mission except that there was no requirement to maneuver and point the high gain antenna. This meant no firm requirement for an absolute roll reference. An additional 2.6 lb of Attitude Control gas was allocated to provide coarse stabilization to the sun line. The gas requirement was based on a simple extrapolation of the propellant requirement for a wider deadzone limit cycle for an additional eleven months. The inadequacy of this estimate will become evident in the discussions which follow.

3.3.2 Contract Award to Preliminary Design Review

Changes during this period occurred as a result of detail design and analysis efforts. It was also during this period that the total spacecraft weight became a major problem and forced the decision between alternatives in favor of the lighter method.

The first class of changes to be discussed are those resulting from problem areas that were not anticipated in the proposal but which were identified before the Preliminary Design Review (PDR).

1. Reacquisition of Celestial References

A unique problem of an orbiter spacecraft is how celestial occultations are handled. Orbits for the Lunar Orbiter photo mission resulted in typical Sun occultations of 45 minutes out of each orbit period of 200 minutes. Canopus could be occulted with similar frequency. The Lunar Orbiter design concept was to automatically switch the gyro to inertial hold when the celestial sensor was not present. Likewise the celestial sensor error signal was to be switched back into the control loop when the occultation period was over.

The probability of being outside the switch lines at sunrise was not properly assessed. Gyro drift had been considered but not the null shifts from being at an unknown random position within the $\pm 2^{\circ}$ deadband at the time of occultation. Figure 3-7 illustrates the sequence for the pitch axis; yaw would be the same. Roll would be a variation of the same idea. The penalty for one reacquisition illustrated above would be twice the $\dot{\theta}$ expended at sunrise to get back inside the switch lines. A 1°/sec $\dot{\theta}$ used approximately .007 lb. of N_2 gas.

The total gas penalty for each axis would be the sum of these individual amounts which would be dependent on a more or less random position in the deadzone at both sunset and sunrise.

2. Maneuver Initiation

The sequence proposed for initiating a maneuver of the spacecraft was to:

a) Close the deadzone to $\pm$.2 degrees.

b) Wait for a zero crossing of the celestial sensor.

c) Slew plus or minus with gyro switched to rate mode.

Analysis showed that closing the deadzone on each orbit caused the expenditure of approximately .02 lb. of N_2 per 3 axis closure. Continuous operation in a $\pm$.2 degree deadzone used only .014 lb N_2 per orbit. Therefore, a saving of .007 lb of N_2 per orbit could be had by continuing in the .2 degree deadzone instead of closing the deadzone once per orbit from a 2° deadzone.

The "Wait for zero crossing" idea was originally proposed to eliminate the inaccuracy contributed by the position uncertainty within the deadzone. Analysis showed that there was a strong possibility that disturbance torques (gravity gradient and solar pressure) would be sufficient to cause an unsymmetrical limit cycle and if already in the .2 degree deadband, no zero crossing would occur. Thus it was concluded that maneuver time uncertainty of the zero crossing negated its value and it was eliminated from the design.

The final design provided for a deadzone closure of all axes upon command from the flight programmer followed by a 51.2 second wait time for settling, immediately followed by the slew command to the axis being maneuvered. Another feature which was incorporated in the pitch and yaw channel was a limiter on the sun sensor at 2.4 volts. The limiters prevented the reaction control from accelerating the spacecraft to a high rate when acquiring the sun from a large angle.

3. Wide Field of View Sun Sensor Errors

Pointing errors introduced into the sun sensor by secondary radiation from the moon could have caused pitch or yaw errors of approximately 1° to 2° because of the wide field of view of the sensor proposed. The solution was to provide a fine and a coarse sun sensor, selectable by command from the flight programmer, and to switch out the wide angle coarse sensor whenever the coarse sun sensors were not required.

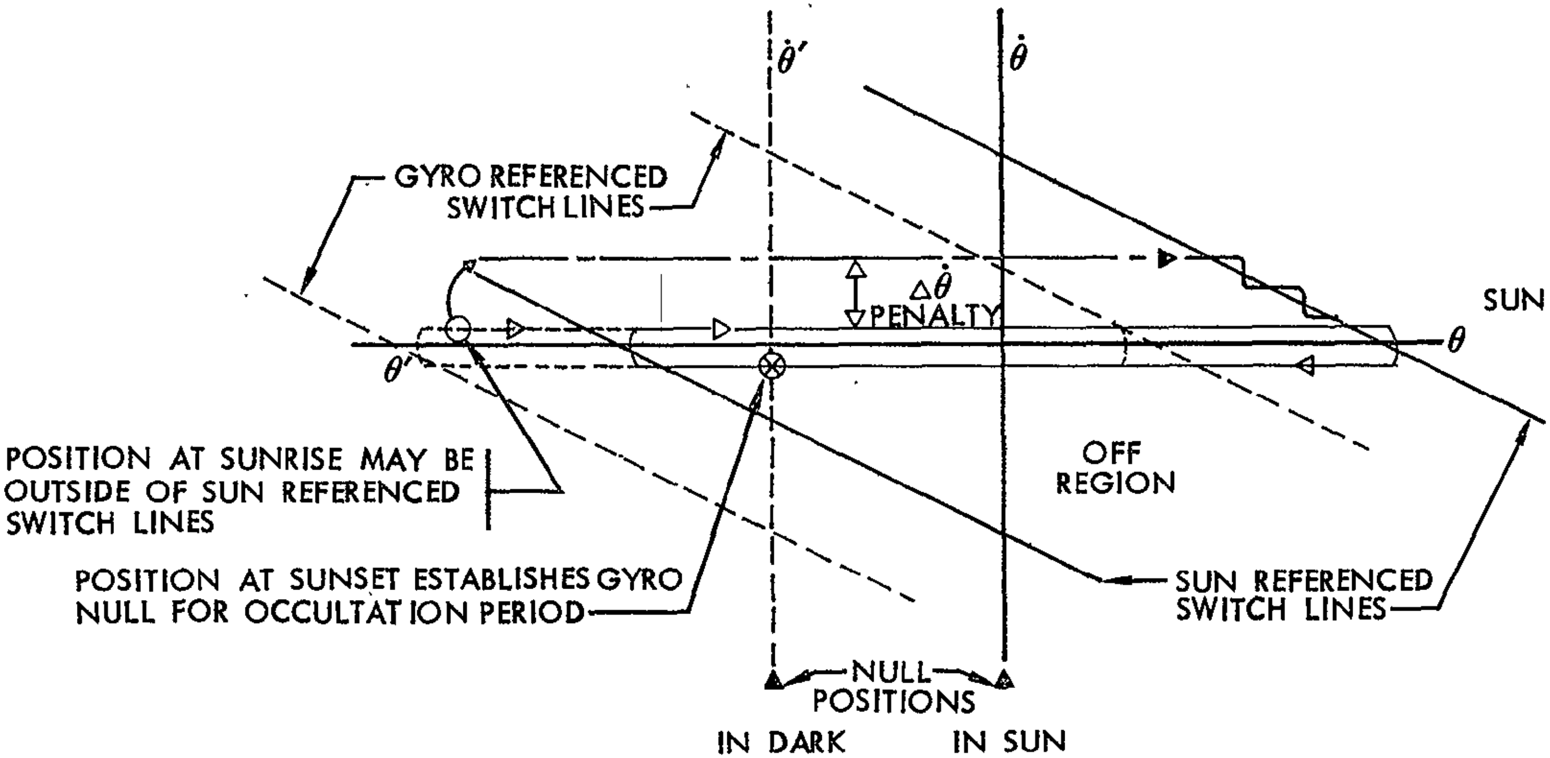


FIGURE 3-7 PHASE PLANE FOR SUN OCCULTATION AND REACQUISITION

4. Extended Life

The method of providing an extended life capability of eleven months following the primary photo mission of 30 days remained an unsolved problem for most of the time period preceding the PDR. Several alternatives were examined and rejected. One of the most promising was to operate the gyros in the rate integrate mode and to slave the gyros to the celestial sensor error signal. Projected gas requirements to continue operation in the same manner as during the 30 day photo mission, i.e., with Sun and Canopus reacquisition each orbit, were prohibitively heavy. The major item requiring the greatest propellant expenditure was the celestial reacquisition problem discussed under item (1) above.

The proposed design solution presented at the PDR involved an operational concept without much change in control hardware. The gas quantity allocated to extend the life to 1 year ranged from 2.6 lb to 5.90 lb. The 2.6 lb was simply an optimistic estimate of required limit cycle gas in a wide deadzone.

The 5.90 estimate was based on analyses that included probability of reacquisition, disturbances, and leakage requirements. The operational method proposed to be employed was as follows:

a) Control roll to ± 2 degree limit cycle with roll gyro in rate integrate mode.

b) Operate pitch and yaw in wide deadzone with coarse sun sensors only - this provided deadzones in the range of ± 10 degree to ± 16 degree. Also lowered the probability of requiring a celestial reacquisition each orbit.

Another important outgrowth of this analysis was the philosophy that the quantity of control gas was fixed at 10.58 lb. Reserves and safety factors became expendables available to extend mission life, but could be traded off for more maneuvers or for other mission options.

5. Thrust Vector Control

Thrust vector control by means of gimbaling the rocket engine instead of high level reaction control jets located at the tips of the solar panels was a major change and is discussed under Thrust Vector Control Subsystem, Section 5.0.

The low level reaction control jets were relocated on the rocket engine heat shield and are discussed in detail under Reaction Control Subsystem, Section 4.0.

6. Location of Antenna In View of Canopus Tracker

Initially the low-gain antenna was located in the direct field of view of the Canopus tracker. The antenna location was originally selected because of the desire to maximize the signal strength received at earth over the low gain radio link. Since the antenna pattern was torroidal the idea was to put the plane of the torrid near the earth moon or ecliptic plane. The antenna pattern was broad which allowed shifting the boom and antenna away from the tracker optical axis. In the final design the object nearest the tracker field of view was the tip of the antenna. It was 23 1/2° from the optical axis of the tracker. As can be seen from the discussion of the Canopus tracker operational problems in section 3.3.6 this was probably not far enough!

3.3.3 Preliminary Design Review to Critical Design Review

The emphasis during this period (approximately 3 calendar months) was on establishing the adequacy of the detail design. A series of development tests were run on the attitude control components and subsystem. Performance tests of the breadboard attitude control subsystem were conducted on an air bearing simulation of the spacecraft. Data obtained confirmed the soundness of the design concepts. Some potential problem areas were identified (e.g., electrical noise from sources such as the gyros, thruster operation, or other switching transients).

The principal design change was to reduce the roll thruster force level to .028 lb from .05 lb. This force level gave similar torque to inertia ratio capability to the roll axis as existed in pitch and yaw. Analysis also indicated a potential N_2 gas saving over the entire mission of 0.8 lb.

The pitch and yaw thrusters were oriented so that plus and minus jets thrusted in opposite directions. This jet orientation was selected to minimize the net accelerations imparted to the spacecraft by attitude control. It was desired to make the unpredictable accelerations on the spacecraft as near zero as possible because the effect would be to cause errors in the gravitational model deduced from orbit determination data.

3.3.4 Critical Design Review to System Design Verification Tests

The actual design changes to the attitude control during this time were minimal. Analyses refined the propellant requirements. Breadboard tests of the 3 axis control system were completed and a high level of confidence was obtained in the ability to predict the control performance in space.

Effects of component tolerances on system operation were re-evaluated accounting for all known tolerances. The impact on attitude control of component tolerances or of specification changes was assessed frequently as demanded by the difficulties in meeting component requirements as set by prior procurement specifications or by test requirements.

One important design philosophy was adopted in this time period which eased the component and spacecraft assembly problems considerably. Namely, it was decided to calibrate the spacecraft by measuring alignment of important items rather than require the precise installation of the item on the spacecraft. The resulting alignment data would be incorporated into the command software computer programs as necessary. Examples of this alignment process were:

- Sun sensor null to spacecraft X axis
- Canopus tracker reference mirror to spacecraft Y and Z axis
- Center of mass location relative to X, Y, and Z

- o IRU mirror relative to Sun sensor and Canopus tracker mirror
- o High gain antenna relative to reference sensors
- o Camera optical axis relative to X, Y, and Z

3.3.5 System Design Verification Test to Actual Flight Operations

System, subsystem, and component problems came to light as a result of ground tests. One set of flight type hardware designated "Set C" was used to assemble a complete spacecraft. A logical buildup of tests by component and subsystem preceded the system integration tests at the spacecraft assembly level. An important milestone in the development of the attitude control was the Subsystem Design Verification Test (referred to as SDV_1).

The purpose of the first of these tests was to demonstrate compatibility among components when connected together electrically. The Flight Electronics Control Assembly was checked with the Star tracker, the IRU, and the Sun sensor. Open-loop performance measurements were made whenever possible; for example, to check maneuver accuracy on the rate table.

Closed loop tests of the attitude control were conducted on an air bearing platform simulating the spacecraft. The purpose of these tests was to obtain verification of correct functioning of the flight hardware.

All of the major functions of the attitude control were observed. This was the first closed loop demonstration of a precise maneuver magnitude.

The unexpected events which occurred were:

1) Programmer would not execute stored programs.

2) IRU lost power whenever oscillator clocks were switched over from Clock A to B or vice versa.

3) Inaccuracies in performing maneuvers on the air bearing occurred.

Design changes resulting from these anomalies were:

1) Beat frequencies between the two Programmer clock oscillators were interfering with normal execution of stored program commands. Previous evaluation tests on the programmer had been done with a test set connected to be able to monitor operation. The air bearing test was the first time it was possible to observe the operation without extraneous equipment attached. Evidence of the problem was simply a hangup in the stored maneuver program where the simulated spacecraft would not maneuver back to reacquire the sun or continue in the stored program. The cause was isolated to extraneous bits appearing in the command word as a result of cross talk or beat frequencies between the two clock oscillators. Circuit changes required to correct this deficiency were incorporated.

2) IRU power was lost at clock switch over due to a state change in the programmer. This problem was tied up with item (1) above and circuit changes corrected the problem.

3) Maneuver accuracy was questioned because:

   a) 3 axis air bearing tests could not verify closed loop maneuver precision because measurement equipment was not precise enough. Also too many test effects obscured the real accuracy; e.g., earth rate, "g" sensitive drift and position instrument inaccuracy.

   b) Spike noise from the IRU was out of spec. and had a detrimental effect on maneuver accuracy. Changes to the IRU were incorporated to reduce the spike noise. Later tests on the precision rate table indicated maneuver accuracy requirements were met.

The conclusions reached from these test experiences were:

1) Overall system accuracy evaluations are the most difficult to measure in ground tests. Some short cuts initially implemented to save schedule time and money had to be reworked; for example, a precision rate table test with the programmer and IRU operating together was essential to evaluate maneuver accuracy.

2) Test specifications were based on space operating requirements and errors allowable in an earth test environment were greater. Insufficient effort was put into analyzing the effect beforehand. Whenever an inaccuracy (for out of tolerance test result) occurred, it was difficult to answer the question as to the cause; design deficiency, true failure of a component, or a test environment effect.

3) A test without ground system checkout equipment or cabling connected is essential to obtain a good picture of expected operation. The air bearing tests were an invaluable tool in validating the system performance.

4) The completed flight spacecraft tests in the 3 axis test stand were also an essential ingredient to the operational confidence in the attitude control system on all five missions. It gave the test crews an operating model which previewed all space maneuvers except actually hot firing the engine.

3.3.6 ATTITUDE CONTROL FLIGHT OPERATIONS

Before the first flight a major question was the extent and rate of degradation of the thermal coating of the equipment mounting deck due to exposure to ultra-violet radiation in space. If the feared degradation occurred, the equipment mounting deck and thermal environment was expected to exceed the upper design limit of +85°F during translunar flight when the spacecraft was in full sunlight. Later in the flight the continued degradation of the coatings would also cause excessively high temperatures. An operational solution was proposed to be employed if required by the thermal environment, namely: to maneuver off sun approximately 35 degrees to maintain the thermal equilibrium. The consequences of exceeding an average temperature of approximately 70 degrees F over a long time period prior to completion of photo processing would have been a progressive deterioration of the photo processing bi-mat. Mounting deck temperature above approximately 100 degrees F would have caused the temperature control limit on the gyro heaters to be exceeded with the resultant loss of fluid viscosity and scale factor change.

Flight Experience - The attitude control system operation for all five missions could be considered flawless had it not been for the consistent problem of "glint" with the Canopus tracker. The first six hours and 48 minutes of Mission I were routine and according to plan. At this point the Canopus tracker was turned on and a 360 degree roll was made according to plan to generate a star map in preparation for the first midcourse correction maneuver. Canopus could not be identified. The actual star map had no resemblance to the a priori map. Failure to obtain an accurate roll reference (Canopus or a reasonable substitute) would have been catastrophic to the mission. The second star map was executed three hours and 40 minutes after the first attempt and produced similar unintelligible telemetry data.

Four hours later (and after much analysis) a third star map was made simultaneously with an antenna signal strength map. The antenna roll map corroborated the location of the moon indicated on both previous maps and confirmed that one blip seen on the second star map was Canopus. The moon was used as the roll reference for the first midcourse maneuver 13 hours later (a total of one day and four hours into the flight). Detailed star map is shown in Section 8.3, which discusses the Canopus tracker.

The diagnosis of the Canopus tracker problem was that the star tracker was "tracking" glint reflected from the light baffles; hence was locked-up at full roll error. One major source of stray light was identified as the low gain antenna which was located 23 1/2° off the tracker centerline in roll. Canopus was finally tracked successfully when in the sun by maneuvering to the roll attitude which placed Canopus near the center of the field of view and turning the tracker off and back on by ground command. The basis for switching off and on was in the design of the tracker acquisition characteristics. The first bright object seen by the tracker in the search field of view would be tracked. This first object could be glint from the sun shade baffles, Canopus or some other star. The off-on cycle might have to be repeated several times to "get" Canopus because of the location of the star in the field of view, glint, and the unpredictable starting point of the instantaneous field of view in the scan cycle. Later in orbit when the sun was occulted by the Moon,the Canopus tracker operated correctly and star maps were generated which closely matched the a priori maps.

The attitude, conduct, and control philosophy of the spacecraft operations team changed radically as a result of this experience. The items which impacted future operations are discussed in the following:

1) The pre-planned mission, the timing of events, the rigorous and inflexible operational modes of the attitude control subsystem which had been planned for use were discarded from necessity.

2) The worst fears of the thermal control people about the spacecraft overheating were confirmed. Operation of the spacecraft at an angle to the sun line provided thermal relief and became accepted as standard required practice.

3) The flexibility of the Attitude Control System to operate for long time periods in modes not previously planned was clearly exposed.

4) The skill of the operations team in being able to fly the spacecraft, almost as if in real time, was demonstrated. Many of the flight operations people had been intimately involved in the design, analysis, and test phases of the program.

5) The confidence in the attitude control and spacecraft equipment was increased by an order of magnitude.

6) The attitude control propellant wasted in the initial portion of the flight (by numerous unscheduled maneuvers) demanded that subsequent operations conserve N_2 gas as the prime criterion. Excellent management of maneuvers, reduction of the number of reacquisition of celestial references, integration of thermal relief off-sun maneuvers with other maneuvers, extrapolation of telemetry indicated gyro drift data and exploitation of the wide deadzone mode for both limit cycle and sun acquisition were all used to complete the primary photo mission of Lunar Orbiter I with all photos read out and with 2.7 lb. of N_2 gas remaining for extended life mission.

3.4 CONCLUSIONS AND RECOMMENDATIONS

The attitude control subsystem performance on all 5 successful flights adequately demonstrated the quality and soundness of the design concepts and hardware used except for the Canopus tracker. In summary the major DO's are:

1) Provide 4π steradian coverage to ensure sun acquisition, but be able to narrow the field of view to avoid excessive errors from outside light sources.

2) Closed loop attitude maneuvers (one axis at a time) using strap down gyros with rate and rate integrate modes provide good precision. The reverse maneuver sequence (or unwind) as employed on Lunar Orbiter proved especially useful in avoiding unnecessary celestial reacquisitions or searches and provided a means of evaluating the maneuver accuracy via telemetry by means of the celestial sensor error when the sequence was completed.

3) Alternate and complementary modes of operation which can be selected by ground command are invaluable. Examples which contributed most to the Lunar Orbiter success were:

   a) Roll orientation could be determined by radio signal strength from the high gain antenna when pointed at earth and was used to corroborate the information from the Canopus tracker.

   b) Star map and roll attitude error signals were of great value to the workaround methods of achieving roll attitude even though the CST failed to consistently allow Canopus lock-on in the sunlight.

   c) Attitude hold using gyro references in inertial hold for long periods of time permitted the operation off sun to prevent overheating the spacecraft; also to solve the Canopus tracker glint problems. In space, calibration of gyro drift improved the efficiency of these workaround techniques.

   d) The reaction control gas supply was adequate for the missions because minimum impulse bit limit cycle was achieved and the operations team could trade off gas budgeted for extended life against extra maneuvers.

   e) The wide and narrow deadzone select option, in conjunction with the fine and coarse sun sensor select option, was used to maneuver at 0.055°/sec rather than the design rate of 0.55°/sec. These options conserved gas.

   f) Derived rate from lead lag networks provide good acquisition characteristics. Electrical noise problems were successfully solved before first launch and limit cycle performance with gyros in inertial hold using the lead lag circuits also achieved minimum impulse bit performance.

4) Many of the people manning the flight operations team had intimate knowledge of the attitude control system gained through involvement in analysis and test from the beginning of the program. Others were new to the design and so had no inhibitions about exploring and exploiting new capabilities. Without the ingenuity and skill displayed by the flight operations team in working around in-flight problems, the program could not have been a success.

Recommendations for "doing differently if one had it to do over" include the following:

1) Additional features which were desired by the flight operations team "if they had it to do over again" were:

   a) The ability to torque the gyros of the IRU or command spacecraft slew at some low fixed rates, e.g., increments $\pm$ .05 degrees/sec. Since gyro drifts were very predictable, they could have been corrected by torquing pulses controlled in duration through the programmer. Hence the effects of gyro drift could have been essentially eliminated.

   b) Gyro rate telemetry scaling was selected to confirm limit cycle rates and gave a saturated output during all maneuvers. An additional scale covering a range of $\pm$ 1 degree per sec. would have been very useful in gas budgeting, and in confirmation of maneuvered angle.

   c) A wide angle sun sensor calibration prior to flight would have been useful in flying off sun. Such a calibration was made in flight from maneuver data.

2) One disadvantage of the operating routine developed for Lunar Orbiter which became evident during the extended mission was that a Flight Operations crew was required to "baby sit" the spacecraft. There were essential required functions, tests, and maneuvers which were interspersed between hours of doing nothing. The result was boredom to the operations crew. This was probably a factor in incorrectly compensating for drift, loss of sun power and the unscheduled loss of N_2 gas on Spacecraft V. This required that it be crashed much earlier than planned.

4.0 REACTION CONTROL SUBSYSTEM

The Reaction Control Subsystem (RCS) was one of four subsystems that constituted the spacecraft G&C system. The relationship of the RCS to the total G&C system is illustrated in Figure 4-1.

4.1 SUBSYSTEM DESCRIPTION

The RCS, illustrated in Figure 4-2 consisted of the nitrogen storage tank, squib valves, filters, pressure regulators, plumbing, and reaction control thrusters. The N_2 storage tank served as the gas supply for the RCS, and as a source to pressurize the fuel and oxidizer tanks in the velocity control subsystem; Figure 4-1 illustrates the plumbing and valves associated with this function. The RCS generated the torque necessary to control the spacecraft about three orthogonal axes designated roll, pitch, and yaw, by use of eight reaction control thrusters. Thruster opening was controlled by the closed loop electronics. A supply of 14-1/2 pounds of nitrogen was stored in a spherical tank at 3500 psi. Ten pounds were budgeted for attitude control, and four for pressurizing the fuel and oxidizer tanks. The nitrogen was regulated to 19 psi for the thrusters and to approximately 200 psi for pressurizing the fuel and oxidizer tanks. The advantages of using space proven, off-the-shelf hardware were major factors in the selection of the nitrogen control gas system. The thrusters and regulator were modifications of those used on the Ranger and Mariner spacecraft.

4.2 SUBSYSTEMS DESIGN REQUIREMENTS

In addition to the general guidelines and constraints specified in section 2.0 which apply to the G&C system as a whole, there were the following specific requirements imposed on the RCS.

Subsystem Life	One Year
Nitrogen Supply	
Attitude Control	8.0
Reserve	2.0
Velocity Control Pressurization	4.0
Ullage and Leakage	1.0
TOTAL	15.0 lbs.
System Leakage	
First Month (including velocity control	<.061 lbs/month
pressurization)	<.037 lbs/month
Thruster Requirement	
Response	
On transport lag	<9.75 millisec
Off transport lag	<9.5 millisec
Cycles (endurance)	>100,000
Leakage	<2SCC/hr/thruster
Thrust Levels	.05 lb. pitch and yaw
	.028 lb. roll - in couples

2

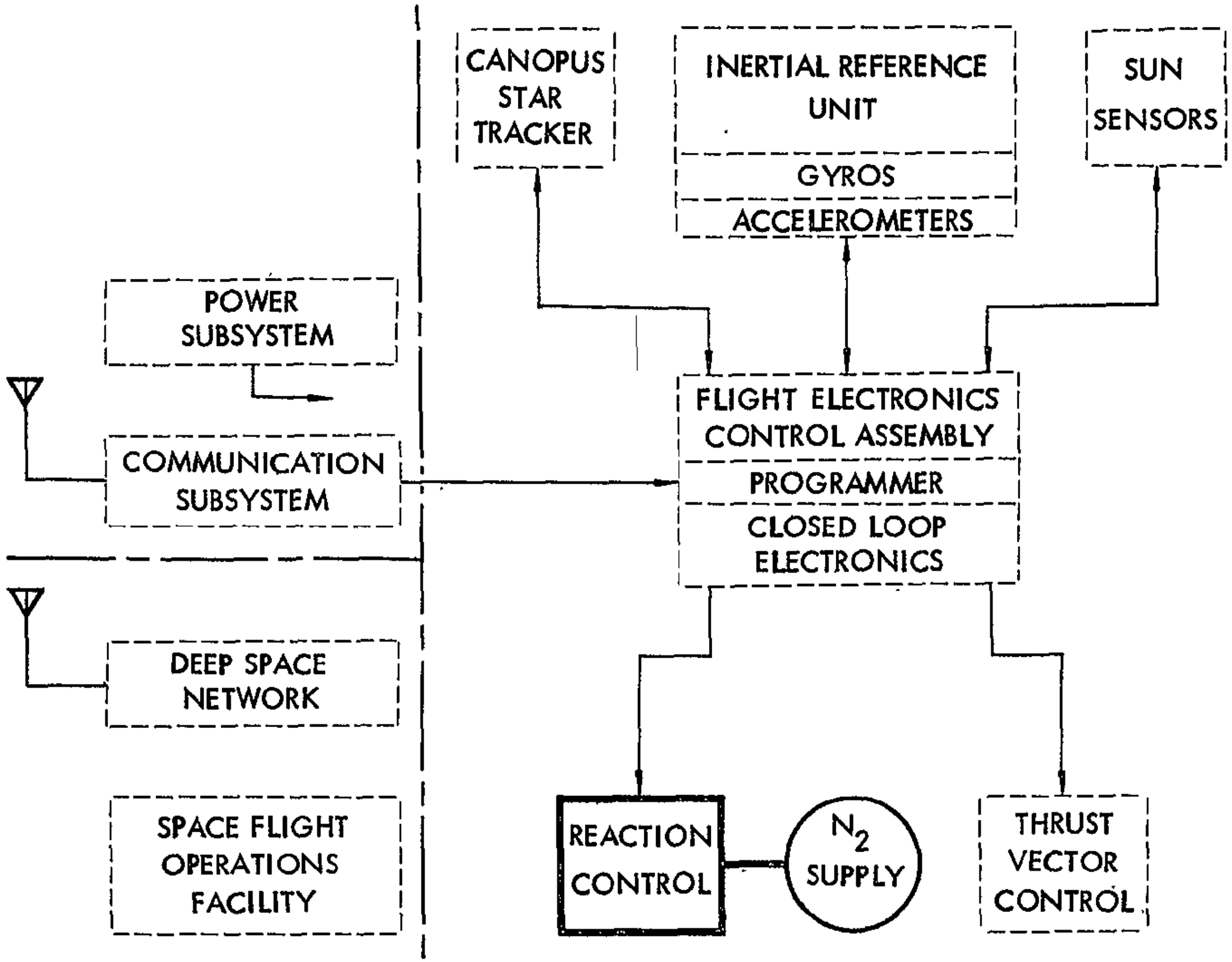
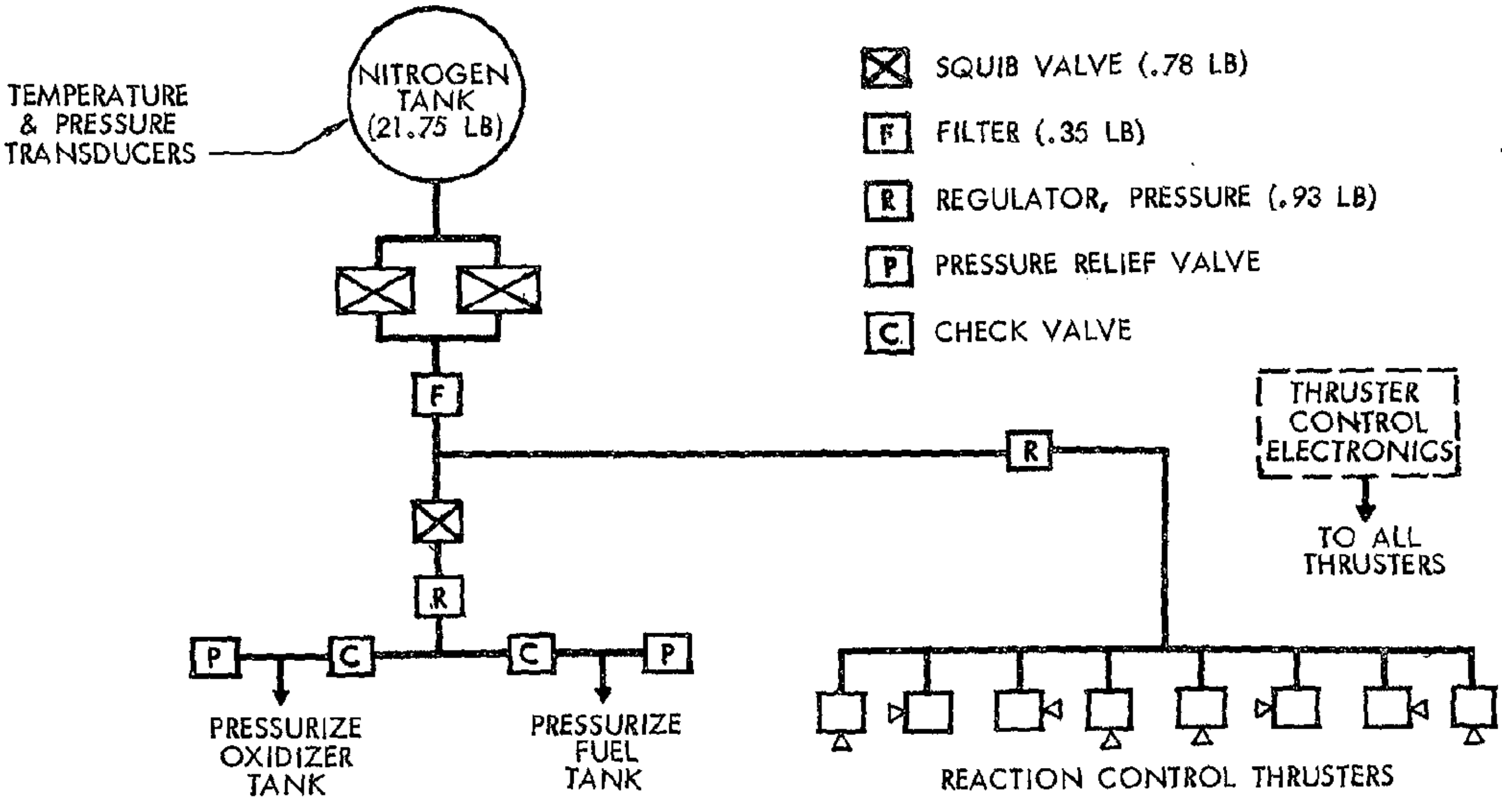


FIGURE 4.-1 G & C REACTION CONTROL SUBSYSTEM



TRANSDUCER WEIGHT

TEMP. = .05 LB
PRESS. = .56

THRUSTER WEIGHT

SINGLE = .23 LB
TRIPLE = .67 LB

FIGURE 4-2 REACTION CONTROL SUBSYSTEM

Temperature	
Thruster body - other components	35° to 85° F.
Thruster nozzle	-300° to $+300^\circ$ F.
Gas temperature	-65° to 85° F.
Pulse width	$11.5 \pm .5$ millisec
Recovery time of electronic driver	1 millisec
Regulator	
Adjustable pressure	16 to 24 psig
Maintain pressure	$\pm$ 1 psi
With inlet pressure 200 to 3820 psig	
and flow rates 0.5 to 2.0 SCFM	
Cycles - lockup to 2.0 SCFM to lockup	240,000
Cleanliness	
Maximum particle size	5 micron metallic 25 micron nonmetallic
Assembled in class 100,000 clean room	

The thruster performance was based on a specific impulse of 68 sec using N_2 gas at 70° F. This also was the basis of the conversion to amount of N_2 gas required.

4.3 DEVELOPMENT AND OPERATION

Cold gas Nitrogen reaction control subsystems had been used on several spacecraft prior to Lunar Orbiter. The Ranger and Mariner were similar applications and the hardware experience and techniques developed on those programs were used wherever possible. However, it became apparent that the particular Lunar Orbiter specifications for environment and performance precluded the automatic approval of the hardware based on "space proven" operation. Most items of hardware were required to be completely re-qualified and documented in a formal manner.

4.3.1 SUBSYSTEM DESIGN EVOLUTION

The stringent leakage requirements, necessary to meet a one year life requirement, played a dominant role in development and operation of the RCS. Many of the design considerations and minor problems encountered in both the development and mission operations were related to leakage considerations. The use of brazed fittings for all the plumbing, except the connection of the N_2 tank, and the use of a cluster of 3 thrusters to reduce manifold plumbing, are examples of the influence of leakage requirements upon configuration. "Modular" construction of the reaction control system was used to keep plumbing lengths short and allow testing at the subsystem (module) level rather than at the spacecraft level. Tests for leakage after installation on the spacecraft were accomplished using a mass spectrometer to sense leakage by means of a helium tracer. Thruster leakage was determined by using a special fitting over the end of the thruster and collecting the gas. Stringent component and assembly cleanliness requirements were imposed to keep contamination low and to help maintain the leakage requirement. Each component had to meet the requirement that there be no metallic particles exceeding five microns in size and no nonmetallic particles exceeding 25 microns in size. Component assembly in a class 100,000 clean room was required and acceptance test procedures included a test for cleanliness of the components. By purging the components with a gas which was passed through a 1.2 micron filter and collected on a 0.8 micron filter, it was determined if the component met the above requirements. If any particle exceeding the requirements was found, the component was purged again until the specification was met.

Several significant design changes occurred early in the RCS design phase. The original system proposal had two levels of thrust, a low level for attitude control and a high level for control during velocity maneuvers. In July 1964, a decision was made to use a gimbaled engine mount, controlled by thrust vector actuators (this trade is discussed in section 5.0). This change, made primarily to reduce spacecraft weight, eliminated the need for the high level one pound thrusters located on the solar panels. It also eliminated the associated structural coupling with the reaction control loop as a potential problem. As part of the weight saving effort a common N_2 storage tank was selected to serve as the RCS gas supply, and as a source of pressurizing gas for the fuel and oxidizer tanks. The several advantages of combining the two systems were: (1) weight saving of one tank instead of two, (2) weight savings by sharing the nitrogen residuals and reserves between the systems, and (3) considerable flexibility between the systems was available, e.g., a squib valve was provided to isolate the velocity subsystem. This could be used after the final orbit maneuver to conserve all remaining N_2 for attitude control (sufficient pressure remained in the velocity system to expel fuel and oxidizer in a "blow down mode" to accomplish a final engine burn to crash the spacecraft at the conclusion of the mission). The major disadvantage was the possibility of a severe leak in one system depleting the gas for the other system. If the attitude control system developed a leak it could not be isolated since the loss of attitude control was catastrophic on the mission. One requirement that resulted from a combined system was to specify materials in the attitude control regulator and thruster which were compatible with N_2O_4, since all leakage back from the propellant tank could not be prevented.

In the original proposal the low level attitude thrusters were located on the equipment mounting deck. Just prior to the preliminary design review the decision was made to move the thrusters to the engine mounting deck to improve lever arms and reduce nitrogen consumption, and to reduce plumbing lengths. At this time the thrust magnitude of the roll thruster was reduced from 0.05 lbs. to 0.028 lbs. to improve limit cycle efficiency in the roll axis.

4.3.2 THRUSTER DEVELOPMENT

The reaction control thrusters were designed to meet the requirements stated in paragraph 4.2. The original design requirements on response were much tighter (e.g., 5 milliseconds "On" transport lag) than final requirements. During the design evolution, it became apparent that the thruster vendor could not meet the response requirements over the specified range of temperature and voltage. Thruster control circuit design had to be modified to be compatible with thruster capability. Response time was set on the thruster

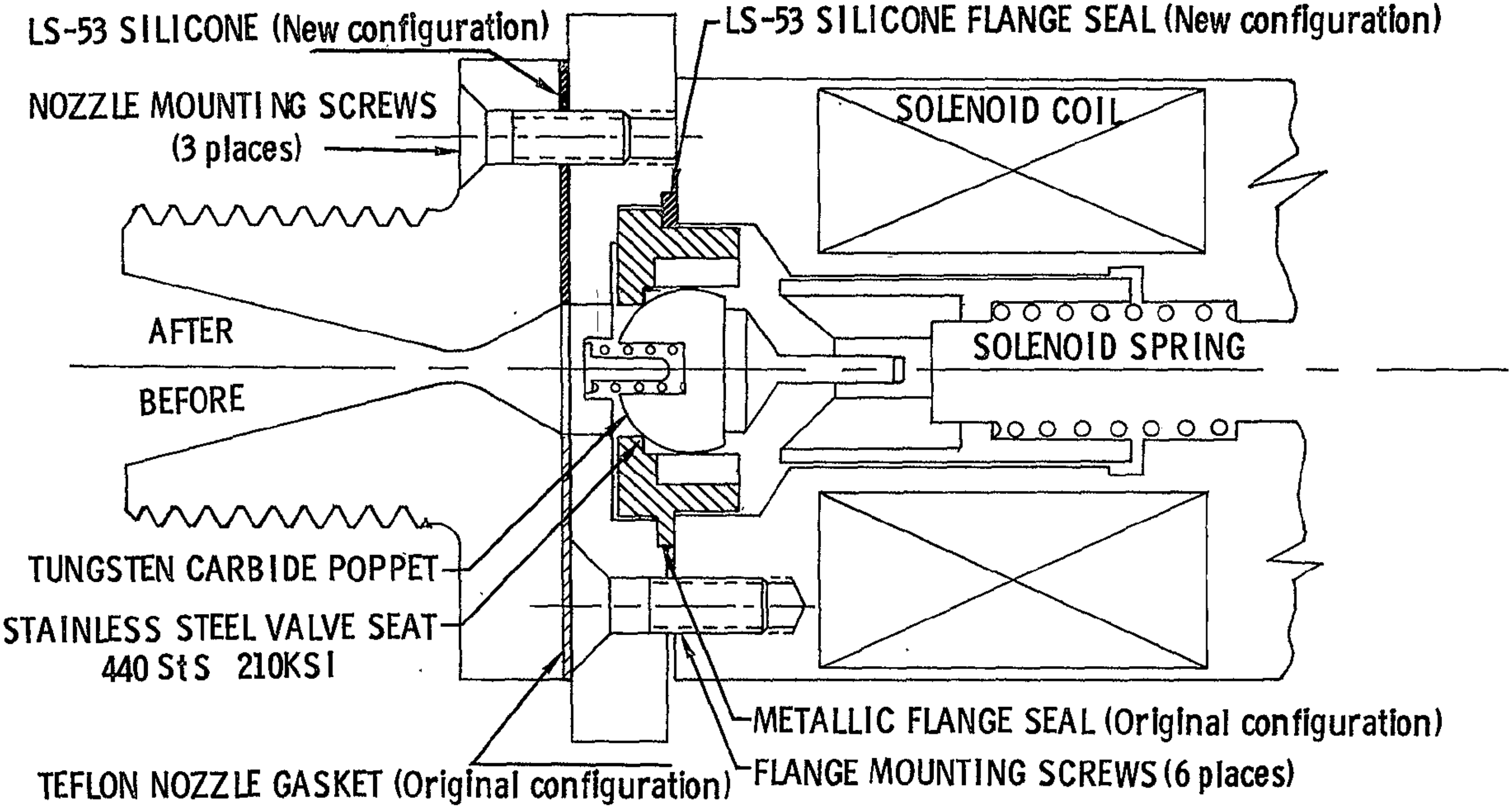


FIGURE 4-3 REACTION CONTROL THRUSTER SCHEMATIC

to minimize the nitrogen consumption for limit cycle operation. Limit cycle rate goals were one-fourth of the design rate used for fuel budget calculations and one-tenth of the design rate as set by photo smear requirements. This conservative requirement was set to assure that the rate requirements would be met even with multiple pulsing of the control system. Limit cycle rates during flight were almost always below the rate equivalent of two minimum thruster pulses. When the rates were higher, the major cause was disturbance torques.

Thrust level was sized by the attitude maneuver method and equipment. The spacecraft was maneuvered at 0.5 degree per second nominally and the reaction control thrusters had to be large enough to reduce this rate to zero in less than four degrees travel (gimbal limits of the gyro).

In order to meet the stringent leakage requirements of 2 scc/hr and have long life with low leakage, hard metal-to-metal seat and poppet were used in the thrusters. The seat was 440 stainless steel and the poppet was a tungsten carbide. Very satisfactory performance was achieved with this design in both test and flight.

Early in the program a leakage problem was encountered during assembly of the hard metal seats to the thruster body. The thruster design during that time period is shown in the bottom half of Figure 4-3. The leakage was being controlled by adjusting the torque on six assembly screws. Due to the very tight leakage requirement, minute distortions of the seat occurred resulting in leakage that exceeded specifications. The hard flange of the seat was modified to mate against a silicone flange seal as shown in the upper half of Figure 4-3. This redesign eliminated the original problem by sealing the seat without distorting it.

One pitch and a pair of roll thrusters were manufactured in a welded cluster of three with common manifolding to save weight, reduce leakage, and reduce the number of alignments required at the spacecraft level. Problems were encountered with the cluster of thrusters. During the assembly welding process, contamination was generated which caused thruster leakage. In many cases only two of the three thrusters would meet the specifications during thruster checkout testing. This resulted in a great deal of rework and schedule delays. In addition if one of the three thrusters was out of alignment, specification on spacecraft assembly realignment of that thruster would usually compromise the alignment of the other two.

In order to maintain moderate thruster valve temperatures, the body of the thruster was mounted inside the thermal shroud. In addition, an insulating silicone gasket washer was used between the nozzle and the valve body which reduced the heat loss through the nozzle to deep space. By necessity, the nozzle was exposed to the deep space environment with a temperature range of $\pm$ 300 degrees Fahrenheit. During testing it was shown that the valve operated successfully at temperatures as low as minus 65 degrees F. Also to prevent the possibility of thruster freeze up, the nitrogen gas was required to be dry enough so that moisture would not precipitate from the gas at temperatures down to minus 65°F.

4.3.3 THRUSTER CONTROL CIRCUIT

The circuit to control opening of the thruster was physically located in the closed loop electronics portion of the Flight Electronics Control Assembly. The "valve driver" controlled the thruster in response to error signals from the sun sensor or star tracker, inertial reference unit, and slewing amplifier.

A "one shot" electronic drive circuit was incorporated in the valve driver electronics to avoid thruster chatter caused by the driver responding to system noise of insufficient duration to cause the thruster to fully open. By preventing thruster chatter the number of cycles would be reduced to extend the life of the thruster for the one year mission. The on time of the "one shot" was selected to satisfy two conditions, (1) with worst case tolerances the thruster would be fully open prior to the one shot off time, and (2) minimum value consistent with the thruster response time to minimize the impulse bit capability of the system. In order to reduce the thruster off delay and protect the transistor drive circuit from excessive inductive kickback, a shunt zener diode was incorporated. The zener breakdown value was "optimized" as shown in Figure 4.-4 so that the off delay matched the on delay. The minimum impulse bit from the thruster with the one shot was 11 milliseconds nominal.

Current and voltage oscillograms were used to indicate actual physical motion of the valve plunger. A typical current trace for valve opening is shown in Figure 4-5a and for a valve closing in Figure 4-5b. These "signatures" were used to measure response time on component tests as well as to prove that the thruster was still functioning when installed on the spacecraft.

4.3.4 REGULATORS

The pressure regulator was designed to maintain a thruster supply pressure of 19.5 psig ± 1 psi from a maximum tank pressure of 3820 psig. The 3800 psi tank pressure was chosen near the maximum value that an available space proven pressure regulator could tolerate without a redesign. The 19.5 psig thruster pressure was chosen to give a good thrust coefficient while maintaining a compact unit and meeting the thrust requirement. The maximum pressure was kept as low as possible under the above design constraints to reduce leakage and ullage. In order to relieve downstream pressure surge resulting from the slam start, a relief valve was incorporated in the regulator. The slam start resulted from firing the pressurization squib following spacecraft separation.

4.3.5 PLUMBING

Stainless steel quarter-inch tubing was used for the plumbing. The quarter-inch tubing was of adequate size to allow flow for a single thruster operation without excessive pressure drop. Because the spacecraft was maneuvered about one axis at a time, only a single thruster would operate for a long duration (not in a pulse mode) at a time. The quarter-inch tubing had pressure drops which resulted in out of specified thrust if all three thrusters were operated simultaneously. Since this mode of operation was not normal and the reduced thrust caused by three thruster simultaneous operation was still adequate for control, the tubing size was not increased.

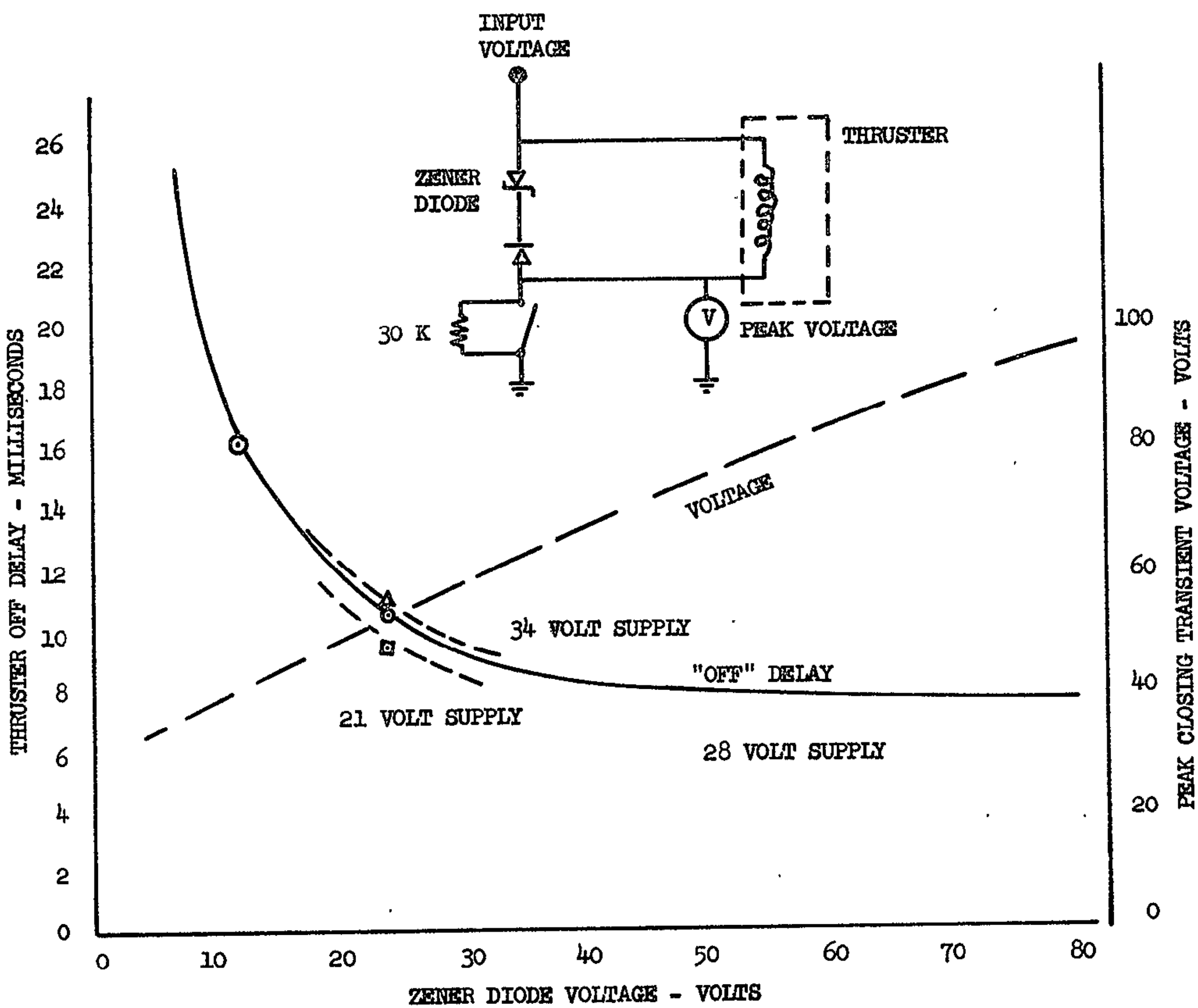
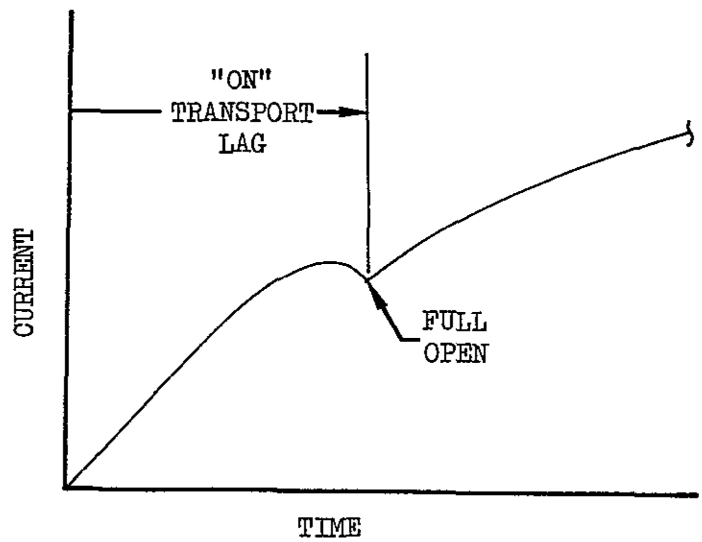
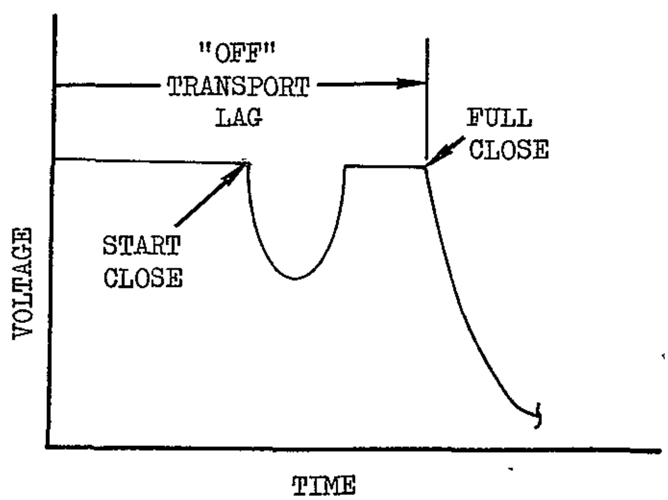


FIGURE 4.-4 EFFECT OF SHUNT ZENER DIODE ON THRUSTER "OFF" DELAY



(a) "ON" TRANSPORT LAG MEASUREMENT



(b) "OFF" TRANSPORT LAG MEASUREMENT

Figure 4.-5 USE OF VOLTAGE AND CURRENT FOR TRANSPORT LAG MEASUREMENT

To keep leakage to a minimum the joints were brazed. Stainless steel tubing was chosen because consistently good brazed joints were easier to achieve. The brazing process was chosen over welding because the plumbing could be assembled and brazed on the spacecraft without contaminating the system.

4.3.6 TESTING

Engineering evaluation tests of the thrusters were conducted using development models of the thrusters with sea-level nozzles obtained from Sterer in August of 1964. These were later incorporated on the air-bearing attitude control simulator.

One of the major questions about the reaction control subsystem from the critical design review of January 1965, was whether the specific impulse was degraded in the minimum pulse or limit cycle mode of attitude control.

An analysis was conducted to assess the total impact on the N_2 gas requirements of a specific impulse degradation. The analysis indicated that the gas requirements for short pulsing were relatively insensitive to I_{sp} over a range from 45 to 68 sec. The reason for this was that gas required for limit cycle went down with lower I_{sp} while the gas required to remove disturbances went up. The two effects tended to cancel out.

In addition, a simple unique method for testing valve thrust and impulse degradation was devised. The test thruster was mounted at the end of a 61 inch piece of 1/4-inch tubing as a cantilevered beam which had a free period of 1.0 sec. (See Figure 4-6). Due to the low damping of this system, the first peak or the steady state value of the deflection is proportional to thrust level. Steady state thrust levels were measured using this test set up. In addition, the same test set up was used to determine impulse when the thruster was driven by single pulses of 5 to 100 milliseconds duration. As a result of this testing, it was concluded that there was no significant impulse degradation due to short pulsing. These were some of the tests which added confidence in the adequacy of the design.

During spacecraft level testing the thrusters were operated for several thousand cycles without gas flow. Throughout spacecraft testing, when the spacecraft was not in a clean room area, the thrusters were kept clean by means of a dust cover over the end of the thruster. As a result, only internally generated conatmination could cause the thrusters to exceed their leakage requirements. By cycling the thrusters several thousand cycles with N_2 flowing, the leakage could be reduced to an acceptable value, <3 scc/hr. The hard metal poppet and seats and gas flow would eliminate any contamination without scoring and return to an acceptable leakage state. This was a very useful advantage of the metal to metal seat design.

4.3.7 MISSION PERFORMANCE

The Lunar Orbiter Mission Operations proved to be very flexible,in part due to an adequate supply of nitrogen control gas. The requirement to provide the capability for control of the spacecraft for one year provided a substantial amount of N_2 that could be traded for modifications to the primary (30 day photographic) mission.

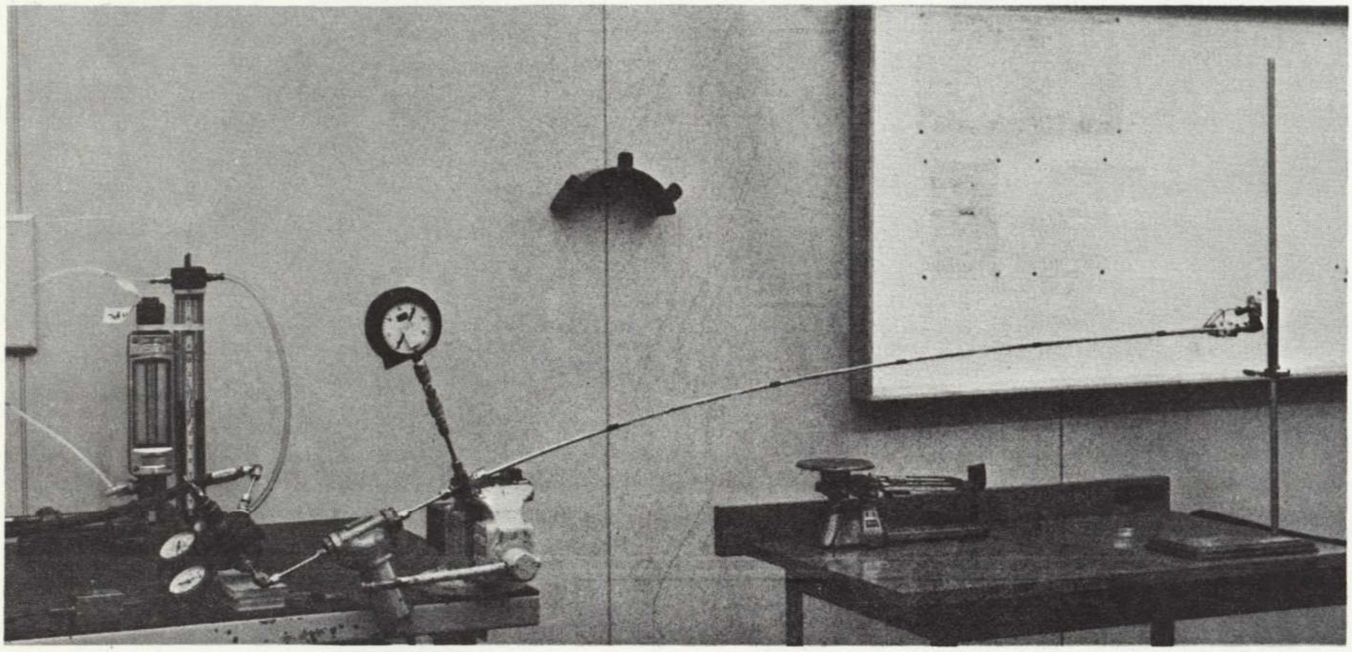


Figure 4-6: REACTION CONTROL THRUSTER FORCE AND IMPULSE TEST SETUP

During Mission I, an internal leakage did occur in the velocity subsystem regulator. In ten days approximately one pound of N_2 vented overboard through the propellant tank pressurization relief valve. A decision to isolate the system even though there was propellant left was made to conserve the attitude control nitrogen. The isolation squib stopped the major leak. Later the velocity control engine was fired successfully in a blow down mode. Following the photographic mission (30 days into the flight), a detail analysis was made to account for all control gas used. The difference between the actual usage (as indicated by telemetry pressure and temperatures) and this impulse analysis was 50 scc/hr. Leakage through the isolation squib of the velocity control subsystem was not evident in tank pressure rise. Since the analysis only covered 2 orbits, approximately 7 hours, the leakage value is at best an estimate due to the coarseness of the telemetry data and inaccuracy in assessing the nitrogen used in the limit cycle mode. The significant thing is that the design leakage requirement was based on approximately 21 scc/hr with the isolation squib completely closed.

Performance during Mission II was as expected. The first clear evidence of leakage through the velocity control isolation squib was indicated by the propellant tank pressure rise after the isolation squib was blown. Methods of conserving nitrogen gas were improved and refined.

Figure 4.-7 is a comparison of the design nitrogen budget and the Mission III actual usage. It is evident from the comparison that the basic photographic mission was greatly expanded from the design mission by comparing the design and actual photo maneuvers. Also, an additional 1.1 pound of nitrogen was used to circumvent problem areas encountered with the spacecraft. It is evident from Lunar Orbiter experience that the control propellant budget for future spacecraft should include contingency gas.

During the last four of the Lunar Orbiter missions there was a small internal leakage through the velocity subsystem isolation squib. The leak was estimated to be 50-100 cc/hr and occurred until the velocity control regulator went to "lock up" and then the leak stopped. An improved design of this isolation squib with a conical shaped plug and seat, rather than a cylindrical shape would probably have reduced this leakage source.

There were no catastrophic failures of any of the components during the five Lunar Orbiter flights which accumulated 920 days of flight time in the five flights. A summary of thruster cycles and total mission time is presented in Figure 4.-8.

4.4 CONCLUSIONS AND RECOMMENDATIONS

The major conclusions of the study of the Lunar Orbiter Reaction Control design are:

1) The cold gas N_2 reaction control proved to be highly reliable, and with operating characteristics closely predictable. There was never any evidence in flight that the delivered specific impulse differed from the theoretical value of 68 to 70 seconds.

PRIMARY PHOTO MISSION (30 Days)	Design Budget	Actual (Mission III)
Initial Acquisition	0.22	0.30
Attitude Maneuvers for Velocity Change	0.55	0.28
Photo Maneuvers		
12 - 3 axis (design total)	1.19	
41 - 3 axis, 8 - 2 axis, 1 - 1 axis (actual)		
Actual Total		3.84
Photo Transmission		
10 days (design)	0.68	
8 days (actual)		0.40
Extra Maneuvers	0.0	1.11
Limit Cycle & Automatic Celestial Reacquisition	1.36	1.07
	4.00	7.00
EXTENDED MISSION		
TOTAL (11 months design, 7 months actual)	4.02 lbs	3.7
Usage Rate	0.012 lbs/day	.0172* lb/day
Reserve	1.98	
Ullage	1.00	
Available when destroyed		1.0
N_2 Velocity Control		
Primary	4.00	2.85
Extended	.0	.65
Total Velocity Control	4.00	3.50
Total Supply	15.00	15.20

*Average usage during 215 days. Included limit cycle and pitch maneuvers for thermal relief plus several special tests and training maneuvers for Mission IV.

FIGURE 4.-7 DESIGN N_2 BUDGET

Compared to Mission III Actual Usage

Mission	I	II	III	IV	V
S/C	4	5	6	7	3
Test Cycles	697,380	497,100	231,500	74,800	96,400
Photo Mission Cycles	26,830	15,260	17,000	19,180	17,110
Extended Mission Cycles	N. D.	N. D.	N. D.	N. D.	214,400
Total Cycles	724,210 [1]	412,360 [1]	248,500 [1]	93,980 [1]	327,910
Cycles Per Thruster [2]	120,700 [1]	85,390 [1]	41,310 [1]	15,660 [1]	54,610
Flight Time Total	79 days 18 hrs	338 days 8 hrs	246 days 9 hrs	73 days 8 hrs	182 days 9 hrs

N.D. - no data

[1] Total of test and photo mission cycles.

[2] Assuming each thruster operates approximately one sixth of the total.

FIGURE 4.-8 THRUSTER CYCLE AND TIME HISTORY

2) Prevention of leakage for the long life requirement had the major impact on the design. Welding and brazing proved to be good solutions. Cleanliness requirements for contamination to be less than 5 micron for metallic and 25 micron for nonmetallic particles was justified. Even so, the velocity control N_2 pressure regulator experienced internal leakage (failing to lock-up completely) on Lunar Orbiter I and wasted approximately 1 lb. of gas.

3) An adequate gas supply permitted the flight operations to be conducted with a flexibility greater than was anticipated at the design stage.

4) A one-shot or minimum pulse bit command to the thruster is recommended to avoid valve chatter problems.

5) Hard metal valve seats are recommended for valves expected to last more than 6 months in space or operate more than 100,000 cycles.

6) Combining the N_2 gas supply for reaction and velocity control proved to be good on Lunar Orbiter and added to the mission flexibility.

7) Valves or other moving parts tend to generate their own contamination particles which may accumulate during testing and cause excessive leakage. The thruster valves on Lunar Orbiter proved to be self cleaning when N_2 gas was allowed to flow from a make-up supply during final spacecraft check-out.

8) Thruster nozzles with external threads allow caps to be installed for protection and sealing. System regulator checks may also be performed without added test valves.

9) Oscillograms of current and voltage provide an adequate signature of physical motion of the valve plunger for functional test purposes.

10) The N_2 tank supply was sealed off from the reaction and velocity systems by squib valves until the squibs were actuated in space. This permitted end to end system testing from the time of final spacecraft propellant and N_2 tank loading until actual launch without being constrained by the amount of N_2 gas which would be wasted if the tanks had not been so isolated. This was a good feature and added confidence in proper hardware operation.

Recommendations for doing differently "if one were to do it over" are:

1) Welding of thrusters into a cluster of three caused problems with manufacturing and delivery schedule slides because the process caused contamination of thrusters previously completed satisfactorily.

2) The isolation squib of the velocity control system was inadequate - it leaked; it could not be actuated against normal system pressures early in the mission. A conical seated squib valve should have been used.

5.0 THRUST VECTOR CONTROL SUBSYSTEM

The Thrust Vector Control (TVC) subsystem is one of four subsystems that constitute the spacecraft G&C system. The relationship of the TVC subsystem to the total G&C system is illustrated in Figure 5.-1.

5.1 SUBSYSTEM DESCRIPTION

The TVC subsystem, illustrated in Figure 5.-2, consisted of 3 principal elements for each axis, the thrust vector actuator, the gyro, and the compensation network. The TVC subsystem controlled the spacecraft pitch and yaw attitude during velocity maneuvers. The Attitude Control Subsystem initially oriented the spacecraft to the attitude desired for the thrusting maneuver, and established a 0.2° deadband. During the velocity maneuver, error signals from the gyros operating in the rate integrate mode, commanded actuator position through the lead-lag compensation networks. The velocity control engine was mounted on a set of gimbals and positioned by the thrust vector actuators. The gimbal assembly utilized flexural pivot bearings, which were selected because of their inherent insensitivity to prolonged space storage, and ability to withstand high temperature resulting from engine heat soakback. The compensating networks were located within the closed loop electronics portion of the flight electronics control assembly. Roll control was maintained by the .028 lb. roll thrusters of the Reaction Control Subsystem. The pitch, yaw and roll reaction control deadbands were opened to $\pm 2.0^{\circ}$ during engine firing to avoid wasting reaction control gas.

5.2 DESIGN REQUIREMENTS

In addition to the general guidelines and constraints specified in section 2.4 and 2.5 that apply to the G&C system as a whole, there were the following specific requirements imposed on the TVC subsystem:

Orient and hold the thrust vector with a total error of 1.3° (including attitude control two axis maneuver errors).

Maintain control with c.m. offsets from actuator null of 1.5°

Recover from engine start burn misalignment conditions:

- Full propellant tanks 3.8°
- Near empty propellant tanks 3.0°

Have 6.0 db gain margin for all flight conditions.

Environment:

- Pressure from sea level to hard vacuum

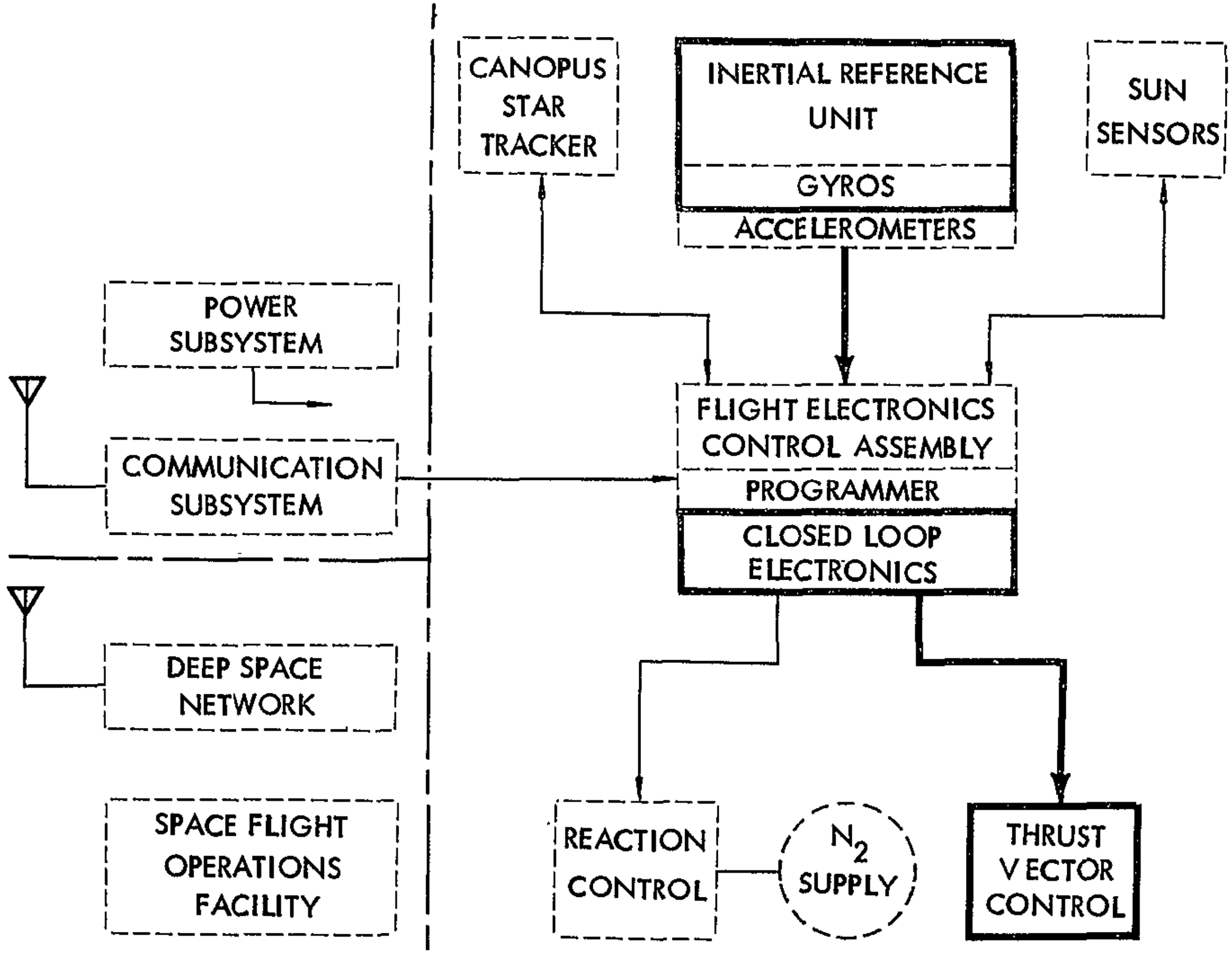


FIGURE 5.-1 G & C THRUST VECTOR CONTROL SUBSYSTEM

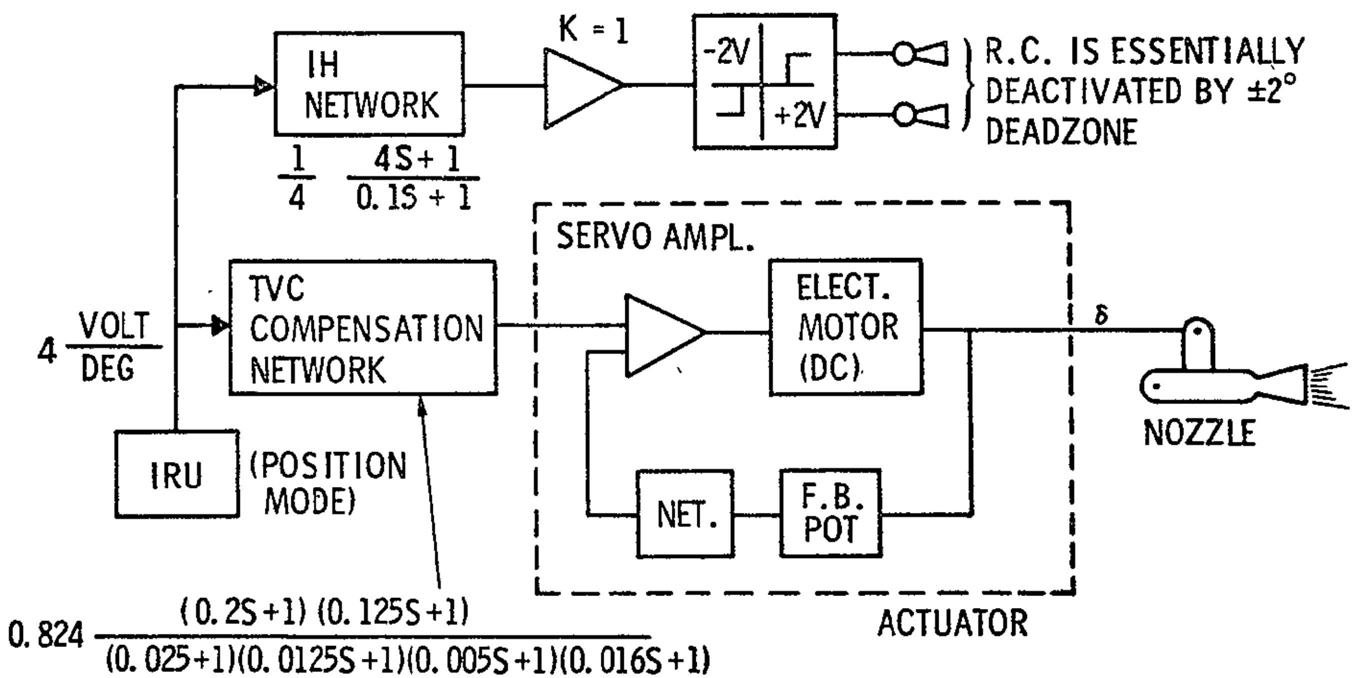


FIGURE 5.-2 THRUST VECTOR CONTROL SUBSYSTEM

Operating temperatures 35° to 85°F.

Actuator rod end soak back - non-operating 175°F.

Vibration - A combination of 10.0 g sinusoidal between 20 and 2000 cps and random vibration with a power spectral density of 0.02 g^2/cps between 15 and 400 cps peaking to 0.1 g^2/cps between 1000 and 1400 cps at 11 g RMS overall for Flight Acceptance Tests. Qualification tests were similar to FAT, but with 16 g RMS random for 4 minutes per axis and 15 g sine sweep at 4 octaves per minute.

The requirements to cope with a c.m. offset and mistrim were based on two values of initial conditions, one a start-burn flight condition with full propellant tanks, and the second an end-burn flight condition with near empty propellant tanks. These initial conditions were:

SOURCE OF MISALIGNMENT	START BURN	END BURN
Initial spacecraft error with respect to the gyro null	$\pm 0.2^{\circ}$	$\pm 0.2^{\circ}$
Initial actuator position	$\pm 2.8^{\circ}$ (stroke limit)	$\pm 1.5^{\circ}$
Engine position for c.m. alignment	$\pm 1.0^{\circ}$	$\pm 1.5^{\circ}$
Initial nozzle angular error from trim	3.8°	3.0°

The large c.m. offsets and shifts from trim used in the design were calculated from the effects of the possible propellant migration between tanks plus initial alignment tolerances. Actuator hard-over conditions were considered for the start-burn flight condition because of the remote possibility that an actuator could overshoot to this condition if the velocity maneuver was short and the actuator power was shut off during an actuator high rate transient. A possible gyro error of $\pm$ 0.2 degrees could exist at velocity engine ignition since these are the deadbands of the attitude control system prior to a burn.

5.3 DEVELOPMENT AND OPERATION

The thrusting control configuration originally proposed on the Lunar Orbiter spacecraft consisted of a fixed engine pre-aligned as close as possible to the average predicted center of mass. High thrust level (1.0 lb) reaction control jets located at the tips of the solar panels were provided to maintain control of pitch and yaw during engine firing. The gimbaled engine configuration was also considered and presented as a backup in the proposal. The high level reaction control was initially selected to avoid the problems of a new actuator hardware development. Early in the spacecraft design phase, total spacecraft weight dictated a vigorous weight reduction program. The most important factors which favored the change to the gimbaled engine was the resultant weight reduction and allowed relaxation of c.m. location requirements. A tabulation of major trade factors are summarized in Figure 5.-3.

A major engineering design decision was made based on this evaluation to change to a gimbaled engine using electromechanical actuators to provide pitch and yaw control during the engine burn. The decision made prior to the PDR permitted incorporation of necessary design changes and the development of a qualified actuator without a delay to the program. Problems arising during actuator development and subsequently in system integration did, however, cause delays in completion of qualification and reliability demonstration tests beyond initial schedules. The TVC actuators were the last items of the guidance and control hardware to receive qualification approval.

5.3.1 THRUST VECTOR SUBSYSTEM DESIGN EVOLUTION

The design concept selected for the thrust vector control loop was based on booster control technology, except that aerodynamic forces were not present in space. The initial transient conditions of engine thrusting direction relative to c.m. alignment were most important in sizing the actuator rate requirement for closed loop stability. The inability to predict c.m. location was due primarily to uncertainty in knowledge of where the propellant was located in the paired tanks during flight.

Two potential problems with c.m. shifts due to unequal propellant in the paired propellant tanks were considered in the TVC subsystem design. The first problem results from propellant flow imbalance during the velocity maneuver. Small imbalances in tank pressures or in outlet pressure drop could result in one propellant tank depleting before the other. Since it could not be proved that the flow imbalance would not occur, matched orifices were installed in tank outlet lines and were sized to be the predominant pressure drop in the propellant feed system which assured that propellant (both fuel and oxidizer) usage from paired tanks would be nearly equal. The second problem was possible migration of propellant from one to the other between the paired tanks during the long coast times between the engine burns. It could not be proved that migration would not occur, so the TVC subsystem was designed to recover from the predicted c.m. shifts due to this phenomenon.

Another problem which strongly influenced the TVC subsystem development was the inability to precisely model the spacecraft dynamics or the actuator dynamics while both were being designed and constructed. The approach taken was to keep the design as simple as possible while maintaining a conservative approach of considering worst case situations for those parameters that could not initially be definitely specified.

Early in the thrust vector design, an attitude trimming network was considered. A simple block diagram of this network is shown in Figure 5.-4. The output of this network was summed with the gyro position. Since the actuator must align the thrust vector with the c.m. for a trimmed condition, a steady offset of the thrust with the spacecraft centerline results. The

ADVANTAGES OF GIMBALED ENGINE

1. Weight Savings - 20 lbs.
2. Eliminates Skewed High Thrust Control Axes Due to Thruster Location on Solar Panel.
3. Eliminates Reaction Control Coupling with Solar Panel Flexibility
4. Eliminate Flexible Plumbing to Panel Mounted Thruster.
5. High Level Thruster Valve Location Problem Eliminated.
   - o Performance is Best With Thrusters on the Tip of Solar Panels
   - o Solar Panel Temperatures are Extreme
6. Simplify Reaction Control System Mechanization and Allow Modular Design.
7. Mission Performance Less Sensitive to C. M. Change.
   - o Relax c.m. Control Requirement
8. Gimbal System Provides Better Growth Capability.

DISADVANTAGES OF GIMBALED ENGINE

1. Actuator Development Required - Time Short
2. Structural Redesign to Provide Gimbaling Compatible with:
   - Space Environment
   - Engine Heat Soakback
3. Actuator Development and Qualifications Cost is a Major Addition.
4. Introduces TVC Coupling With Solar Panel Flexibility.

FIGURE 5-3 TRADE BETWEEN GIMBALED ENGINE CONTROL AND NITROGEN THRUSTER CONTROL

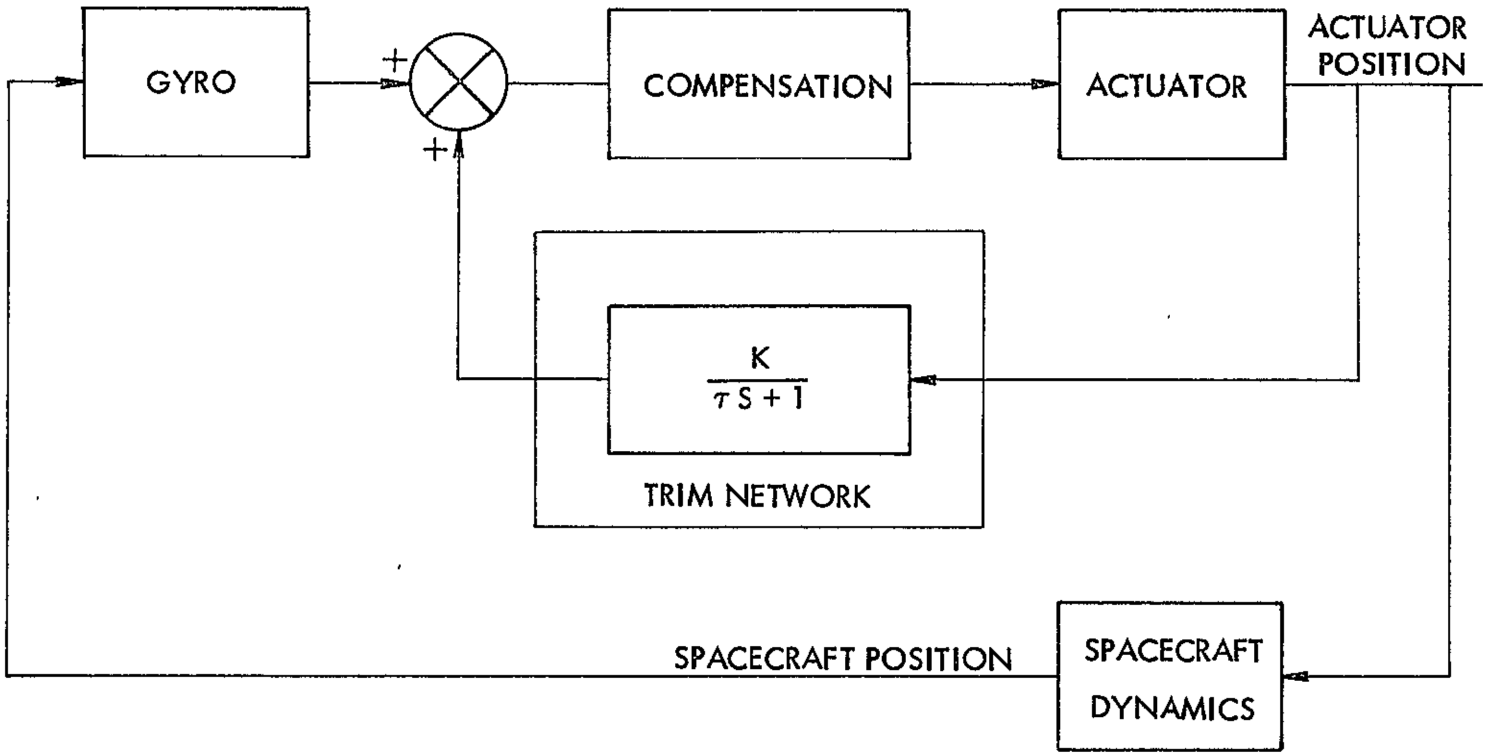


FIGURE 5-4 ACTUATOR WITH TRIM NETWORK

trim network had a long time constant but would eventually cause the spacecraft centerline to move away from the initial thrusting orientation so that the resultant thrust vector would align itself with the nominal desired direction. This network was discarded because it increased the system complexity and reduced reliability. Also, it reduced over-all system stability margin for a given compensation design since this feedback was positive.

5.3.2 SUBSYSTEM ANALYSIS AND SIMULATION

Thrust vector control subsystem analysis and analog computer studies were initiated early in the design phase of the program to define component and system performance requirements. Initial analyses and computer simulations prior to the CDR were based on a simple structural model with variable hinge stiffness of each appendage. Later analysis refined the model to include flexible body parameters as determined from structural dynamic model tests. Linear and non-linear structural effects with propellant slosh were evaluated on the analog simulation.

Even though damping effects of the propellant expulsion bladder was not included (a conservatism assumption), slosh dynamics were found to be insignificant compared to structural coupling associated with the solar panels and antenna booms. Qualitative estimates of the slosh damping provided by the bladders was estimated to be .2 to .3 by observing movies of those dynamic tests. It is also significant that all of the propellant was located between the engine and the spacecraft center of mass and divided among four tanks. This was an ideal arrangement from a slosh stability standpoint.

During the design phase the analog simulation was used to determine compensation requirements, system response, transient response, etc. Early in the design phase, actuator characteristics (i.e., nonlinear effects) were simulated in detail to determine effects on system performance. From the analog simulation, detailed actuator requirements were generated so that overall system requirements and stability could be met. Close coordination between the TVC subsystem analyst and the actuator vendor was required during actuator development to keep the analog simulation up to date and evaluate effects of actuator design changes on system performance.

5.3.3 THRUST VECTOR ALIGNMENT

Compensation for known pre-launch, center of mass offsets was accomplished by aligning the thrust vector with the known c.m. using a mechanical adjustment in the actuator length prior to flight. During the adjustment the actuator was set at its electrical null. This procedure would reduce pointing error required to align the thrust vector through the c.m. Since the thrust vector was not aligned with the spacecraft centerline after these adjustments, the attitude maneuvers were adjusted to compensate for this known offset.

In order to reduce startup transients at each engine ignition, the actuator was designed to be irreversible and the electrical power was switched off the actuator between velocity maneuvers. During each velocity maneuver (if the maneuver was of long enough duration) the actuator aligned the engine thrust vector through the center of mass. At the next velocity maneuver the engine would be trimmed up with the c.m., providing there was no change, and the resulting transient minimized. This technique increased the stability margin since the system stability was a function of initial conditions because of actuator nonlinearities.

5.3.4 COMPENSATION NETWORK SELECTION

Structural design of the Lunar Orbiter solar panels, communication antennas and the deployment mechanisms resulted in a significant amount of structural coupling with the TVC subsystem. The structural frequency of the appendages was between 2. and 3.5 Hz and was primarily the result of the deployment and hinge mechanism. The rigid body control frequency was between 0.4 and 0.6 Hz. Analysis also indicated that the structural coupling was a maximum with frequency separation between the appendages of about 20%. The structural coupling was accentuated by the engine start up thrust exciting the appendage oscillations. Differential frequencies between the appendages resulted in asymmetric (out of phase) oscillations which were sensed by the gyro and fed into the thrust vector loop. Because of this coupling, dual lead-lag compensation was employed to phase stabilize the structural frequencies. A high forward loop gain was required with the position actuators to reduce system droop and meet system pointing accuracy requirements. As a result of dual lead-lag and high forward loop gain, significant noise problems were encountered during the design. Subsequently, changes in this compensation were made to reduce the actuator command signal noise to a tolerable level by adding a double lag filter to attenaute high frequencies. A reduction in forward loop gain caused a slight increase in pointing error but this was acceptable based on system analyses. Noise filtering was also added to the actuator itself to reduce the noise levels.

5.3.5 ACTUATOR

The gimbaled engine configuration with thrust vector control required an electromechanical actuator to be developed for a hard vacuum environment. (Refer to section 8.5 for a design review summary of the actuator). A position actuator rather than a rate actuator was chosen for the TVC design because: (1) simpler compensation in the forward loop was possible with a position actuator, (2) precision position feedback was easier to implement in the actuator, (3) a position actuator eased the initial conditions for engine start up transients, (4) performance of the subsystem was easier to monitor and (5) a position actuator simplified the interface between Boeing and the subcontractor.

Approximately one month after the decision to use a gimbaled engine, the actuator procurement specification was released. During this time period the structural model was undefined and structural frequencies of 1.5 and 1.65 Hz were assumed for the solar panels. The actuator frequency response limit was specified to be greater than the expected structural frequency. The stroke limit was determined by calculation of the maximum travel required by the actuator to align the thrust with the center of mass and adding an additional 1.5 degrees of nozzle deflection for control purposes. The additional 1.5 degrees was determined on an analog computer to give adequate control acceleration for system response. Rate limits were set high to assure adequate dynamic stability margins with mistrim conditions of two degrees and stroke limits of 2.5 degrees. Later these stroke limits were increased to the final value of 2.8 degrees.

5.3.6 TEST

Subsequent to the decision to gimbal the engine for thrust vector control, studies were initiated on a definition of a test and design verification program needed to demonstrate compatibility of the thrust vector control system when coupled to the spacecraft dynamic model. Consideration was given to various levels of hardware sophistication from an actual 6 degrees of freedom tethered spacecraft flight in a vacuum chamber, as had been done on the Mariner spacecraft, to a computer simulation which was believed to be a minimum possible level. Because of program timing, it was apparent that serious consideration could only be given to doing the task at Boeing. A spacecraft level hot firing in a 19-foot diameter vacuum chamber was proposed to Program Management. A summary of significant considerations generated at that time is included in Figure 5.-5. A Management decision was made <u>not</u> to conduct a closed loop test of this magnitude. It was felt that the test could not be justified in view of added cost when considering the inaccuracy of results. Principal limitations of the simulation were caused by gas flow disturbances in the chamber and effects of the suspension on the dynamics of the controlled spacecraft. The JPL conclusion derived from Mariner experience was that such a test served to corroborate analysis techniques. This provided added confidence in the decision. The program that finally evolved was accomplished by a series of separate tests. The first of these tests was a demonstration of actuator performance when installed in the development propulsion module to gimbal the engine during an actual engine firing in a vacuum test chamber. Acceptable actuator and gimbal performance in space environment was demonstrated by a series of open loop tests conducted during a simulated mission engine firing sequence. These tests included a frequency response of the actuator servo loop to various amplitude input commands.

Final subsystem evaluation and design verification of the thrust vector control subsystem was demonstrated during the closed loop test program. For this test, actual thrust vector control hardware as shown in Figure 5.-6 was coupled to an analog simulation of the inertial reference unit and the spacecraft dynamics. Acceptable performance was demonstrated for all expected flight conditions except for the following:

1. Actuator oscillation was experienced due to coupling between the actuator and control electronics. This development problem was corrected by modification to actuator electronics and adding filtering to the TVC compensation network.

2. Gain margin was only 3.6 db for conservatively assumed structural damping of .01. Actual damping was subsequently established by test to be at least .03. Gain margin for this condition exceeded the required value of 6.0 db.

1. The effect of engine vibration and the resultant control system noise generated by the inertial reference unit and amplified by the control electronics must be investigated to demonstrate acceptable system performance. Pro and Con arguments of a spacecraft level live firing in a vacuum chamber are presented.

PRO

a. A vacuum chamber spacecraft firing test is best, next to actual space flight firing, it would include:

   - o Engine thrust characteristics
   - o Actual S/C with flexibility and dynamic coupling
   - o Fuel Slosh
   - o Power supply and wiring
   - o Actual control performance including system interfaces, sensors, electronics, actuation.

b. Superposition holds for linear items only, presence of all effects at once may cause separation.

c. Non-linear effects have not been fully analyzed.

d. A complete simulation of the system could become more complex, difficult, and time consuming than to do the closed loop test using actual hardware.

CON

a. Chamber test and suspension have limitations which will modify vehicle response. Proposed test suspension was constrained to limited freedom in 2 translations and 1 rotation. Engine exhaust would impinge on solar panels and high gain antenna.

b. Work around methods can be employed to investigate each problem separately. Superposition holds, therefore individual examination is valid.

c. Analysis methods for problems are established, proven valid by past programs.

d. Analog simulation can produce better and cheaper answers.

2. Results of 6 degree of freedom tests of the Mariner S/C in a vacuum chamber provide a significant precedent.

PRO

- o Such tests are planned for future spacecraft developed for JPL because "yielded information unobtainable by other means."

CON

- o It was concluded that test results corroborated the analytical approach.

FIGURE 5-5 THRUST VECTOR CONTROL

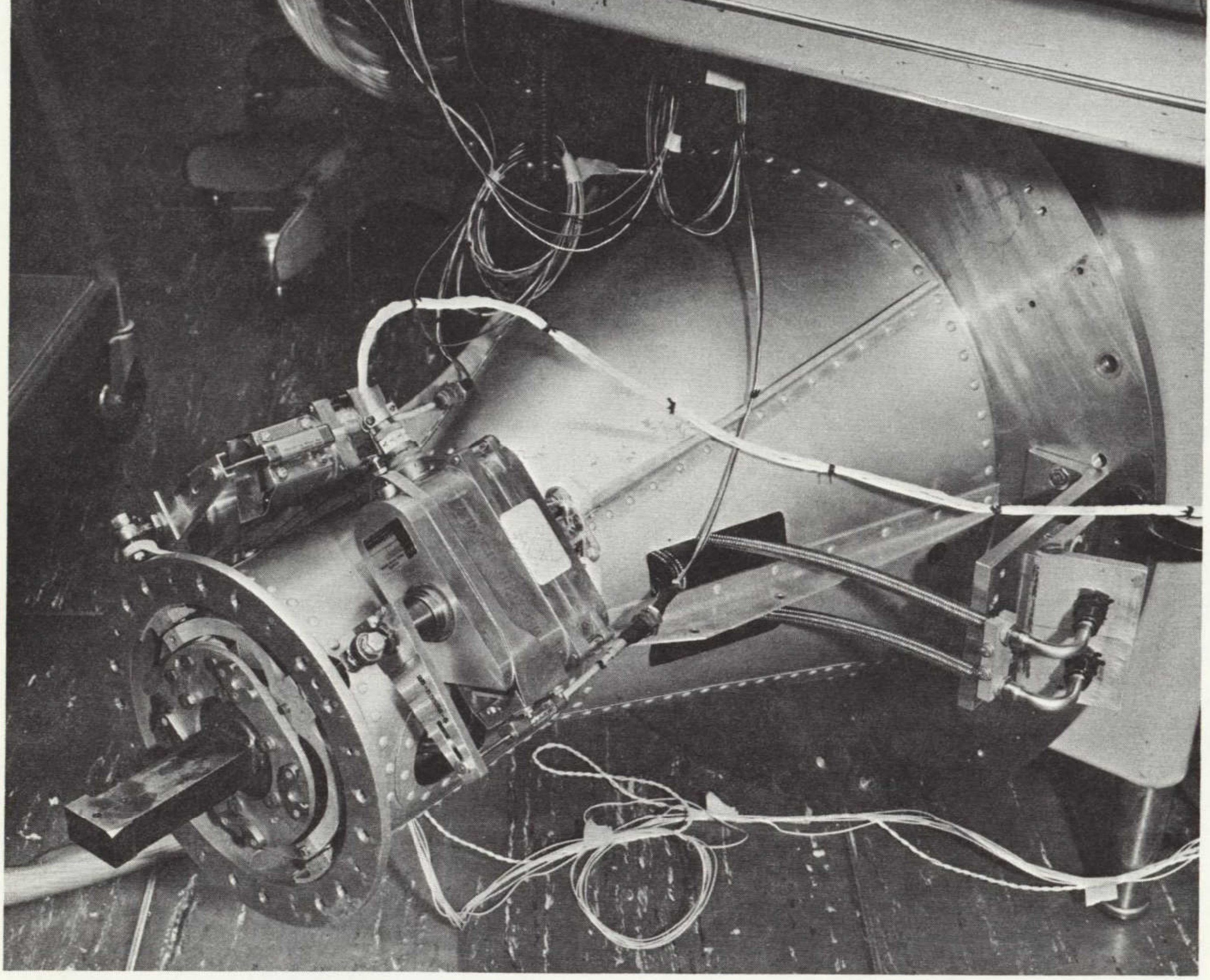


Figure 5-6: CLOSED-LOOP TEST OF THRUST VECTOR CONTROL SYSTEM

3. Command signal electrical noise caused actuator power to exceed allowable values. In addition to the compensation filtering, the actuator electronic switching deadband range was increased to provide additional safety margin.

Closed loop test results verified that analog simulation of the actuator dynamics was very good, and close correlation between results with analog and test actuators were obtained.

5.3.7 MISSION PERFORMANCE

In the five Lunar Orbiter missions the velocity control subsystem was operated a total of 29 times, 16 being in direct support of the primary photographic mission and the remaining 13 being conducted during the extended mission. Although the design operating life requirement for the Thrust Vector actuator was 18 days, successful performance was demonstrated during engine firings as long as 338 days after launch. Figure 5.-7 summarizes all velocity maneuvers. Figure 5.-8 summarizes the operational experience.

Thrust vector control performance in terms of actuator position telemetry data is shown in Figure 5.-9 for both actuators of typical flight spacecraft. These data show that equal propellant flow did occur during all the engine burns and there was no migration during the static periods between engine firings.

Flight verification via telemetry of thrust vector short period dynamic response and stability margins could not be made. Telemetry sampling rate (1 sample every 23.04 seconds) was too slow to assess the transient behavior of the system. The only indication from flight data of thrust vector dynamics and stability were the residual spacecraft rates following engine shutdown. For all the velocity maneuvers, the residual spacecraft rates were very low, indicating good stability of the thrust vector system.

5.4 CONCLUSIONS AND RECOMMENDATIONS

The major conclusions resulting from the study of this thrust vector control system are as follows:

1. The subsystem was more than adequate for the task and all flight results indicated excellent control and stability.

2. Dual lead lag compensation of a position signal leaves the designer most susceptible to both electrical and mechanical noise problems.

3. Structural appendages should have a frequency separation of approximately 5 to 1 above the rigid body controlled frequency.

4. Structural frequency separation between appendages is a critical parameter to the TVC subsystem. Design specifications should keep separation either very small ($< 10\%$) or large ($> 50\%$).

	PLANNED [1]	ACTUAL	ADDITIONAL	TVC OPERATION DAYS AFTER LAUNCH	REMARKS
L. O. I	4	5	o 2nd Orb Trim	15	Lower Photo Altitude
			o Impact		
L. O. II	5	7	o Inclination Change	30	L. O. III Selenodesy
			o Orbit Phasing	157	Solar Eclipse
			o 2nd Orbit Transfer	231	Life Time Adjustment
			o Impact	338	
L. O. III	5	7	o Orbit Phasing	67	Solar Eclipse
			o Perilune Change	162	Apollo Type Orbit Adjustment
			o Apolune Change	207	Apollo Type Orbit
			o Impact	246	
L. O. IV	3	4	o Perilune Change	32	L. O. V Selenodesy
			o Apolune Change	35	L. O. V Selenodesy
L. O. V	6	6	o Orbit Phasing	70	Solar Eclipse
			o Impact	182	
TOTAL	23	29			

[1] 2 MIDCOURSE, ORBIT INJECTION, ORBIT TRANSFER PLANNED.

ONLY ONE MC WAS REQUIRED ON EACH MISSION. IMPACT WAS ADDED REQUIREMENT AFTER L. O. I

FIGURE 5-7 VELOCITY MANEUVER HISTORY

SPACECRAFT	OPERATING TIME GROUND TEST ACCUMULATED HOURS	FLIGHT SEC	FLIGHT HOURS	TOTAL HOURS	VACUUM* EXPOSURE LIFE (DAYS)
L. O. I - S/C No. 4					
Act. S/N 14	237	730	.2	237.2	79
Act. S/N 15	244	730	.2	244.2	79
L. O. II - S/C No. 5					
S/N 16	207	752	.21	207.2	338
S/N 18	208	752	.21	208.2	338 (8100 hrs.)
L. O. III - S/C No. 6					
S/N 21	39	752	.21	39.2	246
S/N 22	45	752	.21	39.2	246
L. O. IV - S/C No. 7					
S/N 23	9	715	.2	9.2	35
S/N 24	9	715	.2	9.2	35
L. O. V - S/C No. 3					
S/N 17	25	720	.2	25.2	182
S/N 26	19	729	.2	19.2	182

* Prior to final operation

FIGURE 5-8 OPERATING LIFE HISTORY - FLIGHT S/C

THRUST VECTOR CONTROL SYSTEM

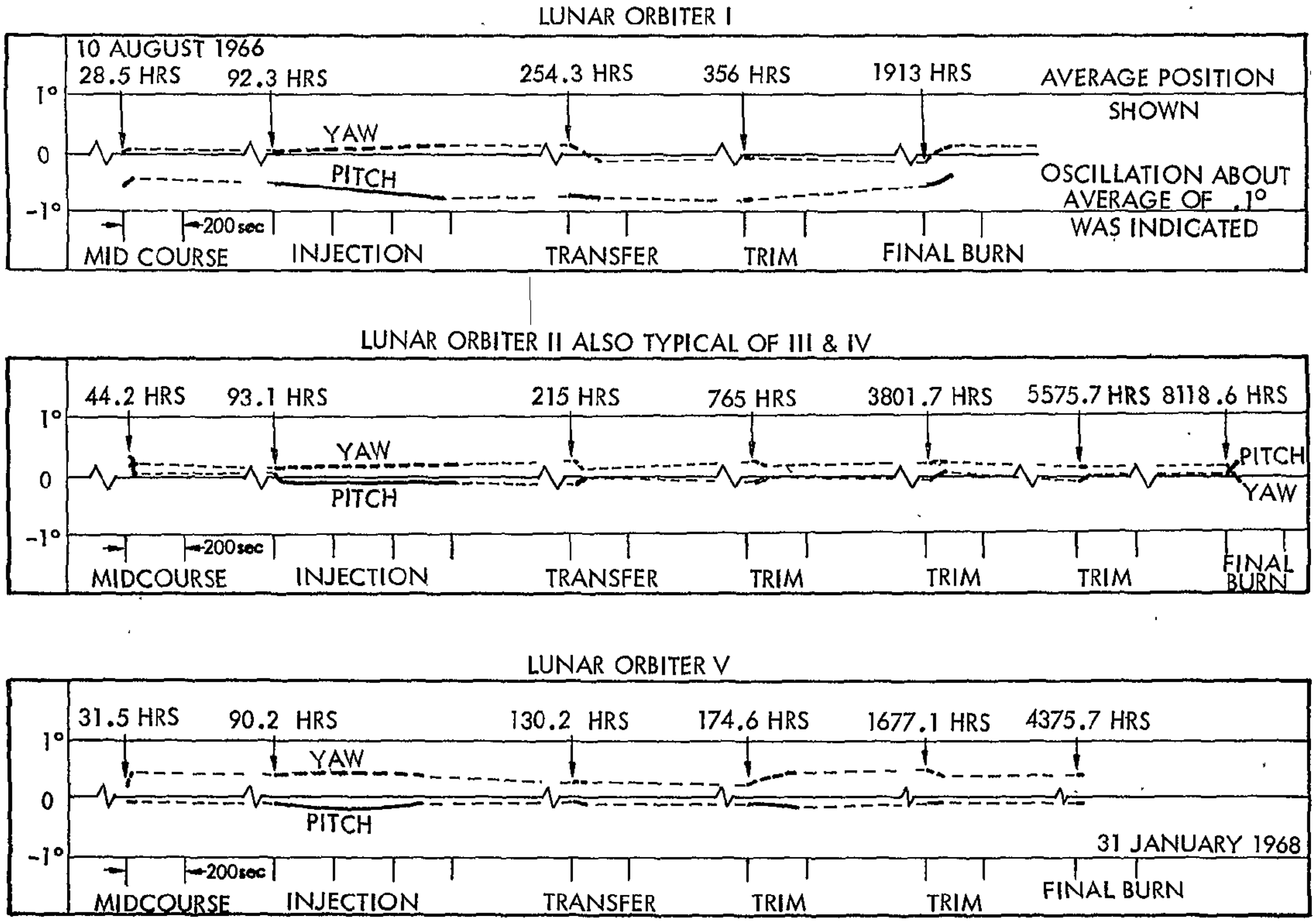


FIGURE 5-9 ENGINE POSITION TIME HISTORY FOR MISSIONS I THRU V

5. Use of an irreversible actuator to maintain the thrust vector aligned with the c.m. during non-thrusting periods will reduce startup transients and improve stability margin with actuator nonlinearities.

6. Subsystem closed loop stability tests can be adequately demonstrated by a hybrid simulation of spacecraft dynamics in a computer and actual hardware.

Recommendations for "doing differently if one had it to do over" are:

1. Interfacing impedance and electrical noise requirements should be specified to each component.

2. Requirements for c.m. offset should be critically scrutinized and methods developed to support or invalidate claims such as "propellant migration" between burns.

3. Other methods than requiring a completely irreversible actuator should be considered for pre-aligning the engine to CM for repeated burns.

6.0 VELOCITY CONTROL SUBSYSTEM

The Velocity Control Subsystem (VCS) is one of four subsystems that constitute the spacecraft G & C system. The relationship of the VCS to the total G & C system is illustrated in Figure 6-1.

6.1 SUBSYSTEM DESCRIPTION

The VCS, illustrated in Figure 6-2, consisted of the following: (1) a pulse integrating pendulous accelerometer (physically located in the inertial reference unit), (2) a velocity summer and magnitude comparator (part of the programmer,) and (3) switching assembly to operate the engine propellant valves. The function performed by the VCS was control of the magnitude of the velocity maneuver. The accelerometer measured the spacecraft longitudinal acceleration and generated a series of pulses with each pulse equal to a velocity increment of 0.1 ft/sec, which were summed by the flight programmer and compared to the commanded velocity change. At the instant the "integrated" acceleration was equal to the commanded velocity change the programmer issued a command to the switching assembly which switched off the power to the engine solenoid valves completing the velocity maneuver.

The 0.1 ft/sec per pulse scale factor was chosen to allow an accelerometer saturation level of 0.6 g's with the 200 pulse per second repetition rate which the accelerometer employed. The accelerometer had two lines with a pulse repetition rate of 100 pulses per second on each line at zero acceleration. The ΔV was the difference in pulses between the two lines which required an "up-down" counter in the flight programmer to feed the programmer counter with plus velocity pulses only. The programmer was designed to have the capacity for a velocity maneuver of 3276.8 ft/sec (998.7 meters/sec) with the accelerometer scale factor of 0.1 ft/sec per pulse.

6.2 DESIGN REQUIREMENTS

The guidance concept dictated that the velocity magnitude be controlled on board the spacecraft in a closed loop manner based on measuring the velocity change made good. A timed burn, for example, was not predicted to be accurate enough. The two critical ΔV maneuvers were (1) lunar orbit injection and (2) orbit change to final photo orbit perilune to an accuracy of $\pm$ 5.5 km (out of 46 km nominal perilune altitude). The consequences of a gross error in ΔV magnitude would have been a failure to achieve a useful orbit or a crash of the spacecraft on the moon.

Several "state-of-the-art" integrating accelerometers used on missile programs appeared to be easily able to meet the accuracy requirements. Since the thrust to weight ratio of the Lunar Orbiter ranged from 1/9 to 1/5 "g", (an uncommonly low "g" level) the scale factor needed to be sized for this particular application.

The strong desire to use an existing off-the-shelf accelerometer caused the detail requirements for the VCS to be based on what could be obtained. The approach was to perform an error analysis for the subsystem using the vendor provided performance capabilities and to compare this with the mission

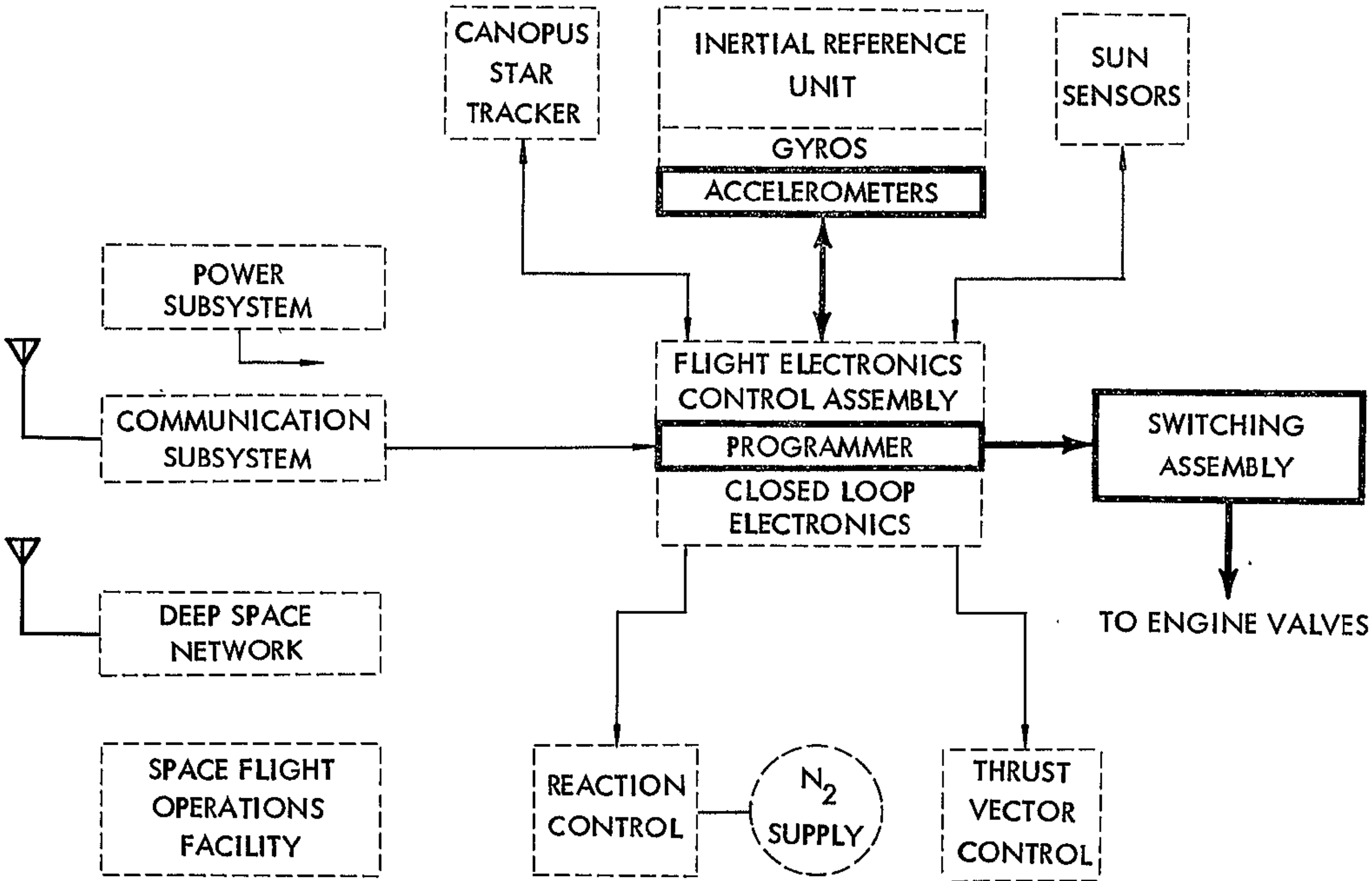


FIGURE 6.-1 G & C VELOCITY CONTROL SUBSYSTEM

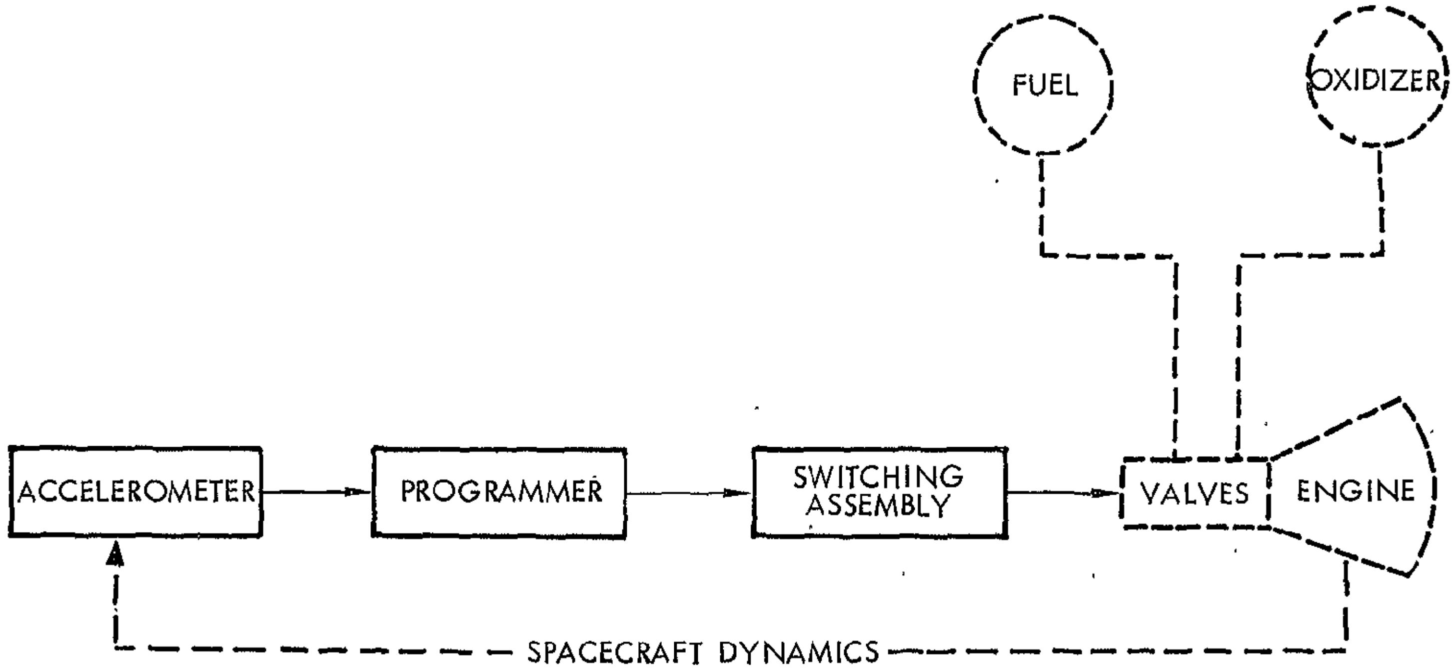


FIGURE 6-2 VELOCITY CONTROL SUBSYSTEM

requirements. Final error analysis values obtained were 0.062 meters/second plus 0.0018 ΔV for the lunar orbit injection. The primary contributors to the error in decreasing order were: 1) accelerometer, 2) programmer, 3) engine tail off.

The initial null bias specification for accelerometer error was set at 100 micro-"g" which was the "quoted" capability of an off-the-shelf unit. As the program progressed it became evident that the original requirement could not be met within the scheduled time and allocated cost. The accelerometer error specification was relaxed following an error analysis that showed that mission requirements could be met with the larger accelerometer errors including a null bias of 300 micro-g's. This relaxed error specification was used in the error analysis quoted above.

The programmer portion of the velocity control subsystem was designed to the specific Lunar Orbiter requirements. The design specification of this unit required a computation accuracy of $\pm$ 0.19 ft. per second for a 3000 ft. per second velocity maneuver and a resolution of 0.1 ft. per second.

6.3 DEVELOPMENT AND OPERATION

In addition to the component level tests, a subsystem compatibility test was performed as part of each spacecraft test sequence. The test was accomplished by orienting the spacecraft to a known attitude in the 3 axis test stand with respect to the acceleration due to Earth's gravitational field so as to be in the operating range 1/9 to 1/5 g. This provided a precisely known value of acceleration acting upon the accelerometer to simulate a velocity maneuver and proved to be a very good end-to-end test.

A failure analysis showed that failure to shut off the engine at the proper time was the most critical failure. This could have caused the spacecraft to crash on the Moon's surface or to deplete the fuel supply prematurely. The second most critical failure was not to ignite the engine at the proper time for lunar orbit injection. To preclude such gross errors from occurring, operational procedures were devised and used on each engine burn. The procedure was to assure engine ignition by monitoring the telemetry link. Real time commands via the DSIF could have been used if required. Also for each engine burn a real time command was sent to terminate the burn at a time which would be after the accelerometer had terminated it with normal operation, taking into account all the tolerance and the delay between command transmission and execution. This procedure was used to insure an engine cut off if any failure occurred in the accelerometer or programmer magnitude comparison circuitry. With this technique it was also possible to perform a timed burn if a failure in the system was discovered. During the 29 velocity maneuvers in five flights the velocity control system operated normally and the backup command was never needed.

Although there was no direct way to measure the actual accuracy achieved for each velocity change maneuver, all indications were that the accuracy requirements were met. As measured on-board the spacecraft and read back via telemetry each delta velocity maneuver was within .1 ft/sec (one pulse) of the programmed value. The minimum programmed ΔV was 5.4 meters/second (burn time of 3.0 seconds) on Mission I and was used to trim the photo orbit perilune altitude. The maximum programmed ΔV was 829 meters/second, on Mission II (a burn time of 611.6 seconds) for lunar orbit injection. In light of the total number of maneuvers performed by the L.O. velocity control system, the accuracy requirements could probably have been relaxed by planning additional engine burns if this had been necessary.

Two midcourse corrections were planned for each mission; however, the accuracy obtained with the first correction in each case was sufficient to preclude a requirement for a second correction. Additional information on accuracy can be inferred from the data contained in section 7.0 which discusses the total guidance concept.

6.4 CONCLUSIONS AND RECOMMENDATIONS

The VCS successfully used available off-the-shelf equipment. Of significant importance was the use of error analysis evaluation instead of holding rigidly to early specifications for the accelerometer. This approach is recommended when it is necessary to impose the constraint of using off-the-shelf equipment. Where possible it would be better to obtain mission error analysis results early enough to develop specifications based on mission requirements rather than vendor performance data. The capability to perform velocity maneuvers by ground command was also a desirable asset. Although the back up capability was never required, the capability was available which improved the flexibility of the system and improved overall probability of mission success.

7.0 GUIDANCE CONCEPT DESCRIPTION

Lunar Orbiter guidance was accomplished by (1) determining the spacecraft state from ground tracking and telemetry data, (2) calculating maneuvers to control the trajectory and to obtain photographic data, (3) transmitting to the spacecraft the guidance commands required to execute these maneuvers, and (4) determining the new state from ground tracking and telemetry data to ascertain that commands were properly executed.

7.1 GUIDANCE REQUIREMENTS

Firm numbers for the guidance and control system hardware were not available during the formative design phase of the Lunar Orbiter program. The hardware development evolved along the lines of "do the best you can" within the state-of-the-art for this equipment. The reason for this approach was that the end result was photographic coverage of 1 meter or better resolution of the lunar surface and with both forward and side overlap of at least 5%. Resolution was a function of many different variables. Given a nominal mission, the resolution degradation effected by a change in one parameter at a time could be determined. Although this gave a feel for the sensitivity of a change, very few were truly independent. At the time of submittal of the proposal, such a set of state-of-the-art hardware parameters were postulated to yield a system giving satisfactory end results. In the time period between contract go ahead and preliminary design review, the hardware specifications were written based on the proposal numbers generated earlier and further analyses allocating errors to individual components where necessary.

Some tolerances were set in an arbitrary manner because they could be achieved with little added cost.

Late in the program, just before the first flight, a comprehensive error analysis was made. Several components could not meet their specifications, for example the IRU drift rate on LO I was 1°/hr. The requirement was waived on the basis of this analysis because the impact was extremely small and could be made up in altered operational procedures.

7.2 SPACECRAFT STATE

A knowledge of the spacecraft state is required prior to the calculation of the guidance maneuvers. The spacecraft state is defined by position, velocity and attitude and is determined from ground tracking and telemetry data.

The position and velocity of the spacecraft were determined by tracking the spacecraft transponder radio frequency signals using the deep space tracking network. The doppler tracking data was processed and used by an orbit determination computer program to generate a spacecraft state vector. Basically, the problem was to determine the trajectory which best fit the tracking data over a given data arc. The output of this program was the spacecraft state vector; the three components of position and the three components of velocity at a given time. A more complete discussion of the orbit determination problem is beyond the scope of this report.

The attitude of the spacecraft was defined relative to the Sun and Canopus. Maneuvers away from the celestial references were in the gyro coordinate system. Each spacecraft was measured for alignment of the Sun sensor, Canopus tracker, inertial reference unit, camera optical axis, high gain antenna, and center of mass. All of this data was used to generate the desired precise maneuver commands. When gyro drift was a factor because of the time off celestial references, it was also taken into account.

7.3 CALCULATION OF SPACECRAFT MANEUVERS

When the spacecraft state was known, the required maneuvers were calculated to provide for the accomplishment of the mission objectives. These maneuvers were velocity control maneuvers required to change the trajectory and photographic maneuvers required to obtain the desired photographic coverage. Attitude maneuvers were also required for subsystem operation such as for thermal control and for star maps but are not discussed here because they were not guidance maneuvers.

7.3.1 VELOCITY CONTROL MANEUVERS

Velocity control maneuvers were required at midcourse, at orbit insertion, and in orbit for orbit adjustments. These maneuvers were calculated using Flight Operations Orbit Determination software at the Space Flight Operations Facility and were also calculated independently at the Seattle Operations Center using mission design software. The calculation of the maneuvers consisted of the determination of the time of the maneuver and the magnitude and direction of the corrective velocity vector. (Maneuver angles were then calculated to rotate the thrust axis from the celestial reference to the required orientation.) These velocity control maneuvers were calculated in accordance with mission control criteria which were defined by pre-flight mission design. These criteria defined the desired end conditions and allowable control tolerances on each parameter. A typical set of these desired end conditions and tolerances is contained in Figure 7.-1.

The Flight Operations software was designed to provide for real time trajectory design. That is, a maneuver was calculated to satisfy some mission objective, rather than to correct the trajectory back to some pre-mission nominal. For example, in the calculation of the first midcourse maneuver, neither the lunar arrival conditions nor the initial orbit conditions were specified. Rather, the conditions over some photo site in the final orbit were specified and the post-midcourse trajectory was designed to satisfy these conditions. In forwarding the state vector from lunar arrival to the photo site, the Flight Operations software used for designing the maneuver was limited to the use of a relatively simple lunar gravitational model. As a result, some inaccuracies were incurred in the calculation of the initial orbit conditions which were used for design of the midcourse maneuver.

A measure of the overall guidance accuracy is given in Figure 7-2. The trajectory parameters actually achieved following a maneuver are compared with the desired values of the parameters established prior to launch. The differences between the desired and the actual values are the errors in the overall guidance concept. These errors are due to (1) errors in knowledge

PARAMETER	NOMINAL VALUE	ALLOWABLE TOLERANCE	LIMIT IMPOSED BY
Apolune Altitude (km)	1500	100 -100	Nearside Resolution V/h, Farside Coverage
Perilune Altitude (km)	100	10 -8	Nearside Resolution, 30 Degree Cross-Track Tilt
Orbit Period (hr)	3.19	± 0.30	± 5 Deg. Cross-Track Tilt at V8A (1P1)
Orbit Inclination (Deg.)	85	± 0.6 Deg.	± 5 Deg. Cross-Track Tilt at 50 Deg. Latitude
Argument of Perilune (Deg.)	0.5	± 10	Minimum V/h of 0.005 Rad/Sec At 50 Deg. Latitude
ASC. Node Longitude (Deg.)	71.41	± 0.28	± 5 Deg. Cross-Track Tilt
True Anomaly (Deg.)	0	--	--
Time (S/C at Perilune)	Aug. 9, 1967	± 2 hr.	± 1 Deg. Illumination

Order of Preference of Solutions

There may be several solutions with values within the allowable range of tolerances listed. If this is the case, it is desirable to get solutions with some of the parameters close to their nominal values. The order of priority of establishing nominal values is as follows:

(a) Node Longitude
(b) Orbit period
(c) Apolune and perilune altitudes
(d) Orbit inclination
(e) Argument of perilune
(f) Time at perilune

FIGURE 7.-1 LUNAR ORBIT CONDITIONS AT START OF FINAL ORBIT

MISSION V

	LO I		LO II		LO III		LO IV		LO V	
	DESIRED	ACTUAL	DESIRED	ACTUAL	DESIRED	ACTUAL	DESIRED	ACTUAL	DESIRED	ACTUAL
MIDCOURSE ΔV	0	38	0	21	0	5	56	61	5	30
AIM POINT										
B.T. km	6402.	6459.	6010.	6044.	5005.	5607.	709.	726.	385.	352.
B.R. km	-1171.	-1119.	-391.	-373.	-2465.	-2479.	9755.	9808.	5701.	5696.
ORBIT INJECTION ΔV	796	790	831	830	705	704	655	660	645	643
Perilune Alt. km	200	188.9	200	196.1	200	210	2700	2706.3	200	194.5
Apolune Alt. km	1850	1835.5	1850	1871.4	1850	1801.6	6110	6114.2	6050	6018.3
Orbit Inclin. deg.	12.04	12.16	11.95	11.94	21.0	19.9	85.5	85.54	85.05	85.01
Argument of Perilune	182.4	180.2	164.41	161.71	175.73	174.43	0.04	1.15	1.37	1.55
Node Long. deg.	326.1	215.4	341.2	341.65	211.41	310.00	131.0	131.03	117.76	117.75
1st ORBIT TRANS. ΔV	26	40	26	28	26	51			12	16
Perilune Alt. km	57.9	57.4	50.2	49.7	54.9	54.9			100	100.4
Apolune Alt. km	1850	1853.9	1844.1	1852.6	1840.0	1847.4			6128.7	6107.0
Orbit Inclin. deg.	12.04	12.0	11.91	11.89	21.05	20.91			85.00	84.67
Argument of Perilune	187.02	181.20	163.3	162.83	178.34	178.88			0.90	1.26
Node Long. deg.	233.74	234.00	273.6	273.3	257.93	257.86			95.90	95.90
2nd ORBIT TRANS. ΔV									234	234
Perilune Alt. km									100	98.9
Apolune Alt. km									1500	1499.4
Orbit Inclin. deg.									85.0	84.76
Argument of Perilune									0.50	1.88
Node Long. deg.									71.38	71.38
OTHER MANEUVERS		2		4		4		2		2

(Not planned in detail before launch -- used to trim orbit, adjust orbit to minimize effects of eclipses, and to crash spacecraft.)

FIGURE 7-2

MANEUVER PERFORMANCE

of state (orbit determination errors and attitude errors), (2) errors in forwarding the state (due primarily to the uncertainty in the lunar gravitational model), (3) off nominal conditions due to previous maneuver errors, (4) errors in calculating the maneuver, and (5) maneuver execution errors. The significance of this data is that it shows how well the guidance concept worked. There were additional velocity maneuvers made as noted during each mission, but they were not defined prior to flight, therefore no comparison can be made. These maneuvers were for additional orbit trims, orbit phasing maneuvers to reduce the effects of lunar eclipses, and maneuvers made to crash the spacecraft.

The procedures used for the different velocity control maneuvers will be discussed separately.

Midcourse Maneuvers

Midcourse maneuvers were required for two reasons; first to correct trajectory errors resulting from launch vehicle cut-off errors and second to retarget the trajectory to the required lunar arrival conditions for the specific mission objectives. Generally, the launch vehicle targeting commenced before the photographic objectives, and hence the lunar arrival conditions, were defined. Therefore, it was necessary to target the launch vehicle to a preliminary aiming point. In the case of Lunar Orbiter IV, the launch vehicle was targeted for an orbit inclination of 21 degrees, while the final mission design was for an orbit inclination of 85 degrees.

With the state vector known at some reference time and the desired end conditions specified (in this case conditions at some time after insertion into lunar orbit), there was one specific maneuver for a given maneuver time and lunar arrival time. This is true because in forwarding the state vectors into lunar orbit, the minimum ΔV orbit insertion maneuver (impulsive) was used. The decision which had to be made then was when to make the maneuver and what arrival time to use. Generally, the maneuver was made as late as possible while staying within the ΔV budget but not later than 50 hours before lunar arrival to allow time for a possible second midcourse maneuver. The later the maneuver was made, the lower the arrival dispersions but also the higher the maneuver ΔV. The ΔV budget was defined as follows:

$$\Delta V \text{ budget} = \Delta V \text{ available (nominal)} - \Delta V \text{ required for later maneuvers (nominal)} - \Delta V \text{ tolerances}$$

The ΔV tolerances included allowances for engine performance (I_{sp}), fuel loading, and trajectory errors.

The arrival time was selected to minimize ΔV (midcourse plus orbit insertion) while observing a constraint to provide for viewing by two tracking stations at lunar arrival.

No second midcourse maneuver was necessary on any of the missions as it was always possible to obtain an orbit within the specified limits.

Lunar Orbit Insertion

At the time the decision was made that there would not be a second midcourse maneuver, it had been determined that a satisfactory orbit could be obtained; that is, one which was within the limits defined in the mission control criteria. Normally, the nominal design orbit could not be obtained because the approach trajectory was not exactly nominal. In the calculation of the orbit insertion maneuver, the criteria was to obtain the "best" orbit. The control criteria which were specified the nominal values and acceptable limits on each of the orbit parameters, and also the order of priority of each of the parameters. The problem then was to compute a maneuver which would result in an orbit with the highest priority parameters at their nominal values while keeping all the parameters within their specified limits.

The procedure used by Flight Operations at the SFOF is described as follows. A "desired" orbit was determined by assuming an impulsive maneuver. This was done by allowing one of the orbit parameters to change to satisfy the impulsive maneuver's constraint of ellipse-hyperbola intersection. Generally, the parameter of lowest priority was allowed to vary and a solution was obtained with the other four parameters at their nominal value. If the value of the variable was outside the allowable limits, additional solutions were searched for with other parameters as the variable. This procedure was continued until a satisfactory solution was obtained. The ΔV was also required to be within the ΔV budget, which was determined in a manner similar to that for midcourse.

Once the "desired" orbit was defined, a finite burn orbit insertion maneuver was calculated to obtain an orbit as close to the "desired" orbit as possible. In this calculation the four maneuver parameters (ignition time, burn time, and two attitude rotations) were used as variables to search on "desired" values for four of the five parameters of the orbit. The fifth parameter never differed significantly from its "desired" value, so there was no problem in satisfying the control criteria.

There were two limitations of the Flight Operations software that became evident during the Lunar Orbiter program. First, some of the orbit parameters which were used in the software were not the same as those specified in the control criteria document. (The software was designed long before any control criteria were conceived). The target conditions used in the program for calculating the insertion maneuvers were given as perilune latitude and perilune longitude. The control criteria specified values and tolerances for node longitude and argument of perilune. The reason that these parameters were specified in the control criteria is that node longitude was a very critical parameter in that it reflected directly to illumination of the photo sites. On the other hand, argument of preilune could have relatively loose tolerances as it had only a small effect on photographic altitude. As a result, argument of perilune generally had a large tolerance and was also of the lowest priority in the control criteria. But because of the characteristics of the Flight Operations program, this parameter could not conveniently be treated as the free variable in the calculation of the orbit insertion maneuver.

The second limitation was that the Flight Operations software required an impulsive solution before a finite burn solution could be obtained. This meant that in the finite burn solution the target conditions were those of the "desired" orbit which was constrained to intersect the approach hyperbola; therefore, it differed from the nominal design orbit. Had the finite burn calculation been targeted to the nominal design orbit, the resulting orbit would have been closer to nominal.

In the calculation of the orbit insertion maneuver for Lunar Orbiter II, the initial Flight Operations solution resulted in a perilune altitude which was about 50 km off nominal, while argument of perilune was essentially nominal. Although this was a satisfactory solution in that all the parameters were within their allowable limits, it was highly desirable to get perilune altitude closer to nominal while allowing argument of perilune to vary from nominal. (Argument of perilune was of lower priority in the control criteria). The backup calculation obtained at the Seattle Operations Center provided a more satisfactory solution.

By performing a finite burn search and allowing argument of perilune to be the free variable, a solution was obtained with perilune altitude near nominal and argument of perilune within its allowable limits. These results were then used by Flight Operations in their final calculation of the maneuver.

Orbit Adjustment

Orbit adjustment maneuvers were generally included in the mission design to lower perilune altitude to that required for photography. At this time adjustments were also made to correct certain orbit errors. Again control criteria had been specified and the maneuver was calculated to get as many of the orbit parameters at their nominal value as possible while remaining within a ΔV budget. (Although this was usually the last planned maneuver for the photographic mission, a ΔV capability to crash the orbiter was retained.) There were no significant problems on any of the missions in obtaining satisfactory solutions.

7.3.2 PHOTOGRAPHIC MANEUVERS

Attitude maneuvers were calculated to provide for photographic coverage of the designated areas. The problem consisted of two parts: 1) the determination of time when the spacecraft would be at the correct position relative to the photo site (this information was derived from the orbit determination results), and 2) the determination of the 2- or 3- axis maneuver which would orient the spacecraft for the required photographic frame coverage. A special flight operations software computer program was available to generate the attitude commands necessary. A complete discussion of these maneuver calculations is beyond the scope of this report.

The overall accuracy of the photographic maneuver guidance is indicated in Figure 7.-3, where the error in the camera axis intercept is shown in terms of its down-track and cross-track components for several of the

Photographic Site	Error in Camera Axis Intercept Degrees on Lunar Surface	
	Down-Track	Cross-Track
LO I-1	-0.33	---
-5	-0.26	---
-8.1	0	---
LO II-2	+0.07	---
-5	+0.17	---
-8	-0.33	---
-12	+0.66	---
LO III-2a	+0.09	+0.12
-6	-0.09	+0.41
-76	+0.18	+0.10
LO V-19	-0.13	-0.08
-31	+0.13	0
-43.1	-0.03	-0.02

FIGURE 7.-3 PHOTOGRAPHIC LOCATION ACCURACIES

photo sites from four of the five missions. This error is due to (1) error in knowledge of state (orbit determination error and attitude error), (2) error in forwarding the state (due primarily to the uncertainty in the lunar gravitational model), (3) error in calculating the maneuver, and (4) maneuver execution error. Only the down-track component is shown for Lunar Orbiters I and II since vertical photography was designated on these missions. For the other missions, both the down-track and cross-track increments are shown as errors because it was intended to aim the camera directly at the photo site on these missions.

The accuracy indicated in Figure 7-3 should not be confused with the accuracy in knowledge of photo site location (selenographic location of a specific lunar feature). This subject is covered in Section 7.5.

7.4 GUIDANCE COMMANDS

When the maneuvers were calculated and approved by the Space Flight Operations Director, the guidance commands were transmitted to the spacecraft to perform these maneuvers. The commands were first coded and transmitted up to the spacecraft, re-transmitted down from the spacecraft, verified, and then the command-execute signal was transmitted to the spacecraft.

The discipline of the command system which was developed at the Spaceflight Operations Facility was as essential a part of the Lunar Orbiter guidance and control system as the hardware. It was this rigid system of pre-mission planning, control of software, flight path analysis and command computational routines, subsystem performance monitoring and analysis, command generation and transmission routines for both real time and stored program commands, and tight overall management by the Space Flight Operations Director's staff that ensured minimum error. The system allowed flexible real time modification to mission plans and provided extensive review and cross checking.

Planning and development for flight operations took about 2 years. The operational software; i.e., the computer programs to be used for orbit determination, command generation, stored program formulation and transmission, star identification, telemetry data handling, etc., were checked out by formal testing and were not allowed to be changed after certification except for very grave reasons.

Specific procedures were worked out for preparation, approval, communication, and verification of both real time and stored program commands to the spacecraft via the DSIF station. These procedures were verified during operations team training which occurred over the six months prior to the first flight.

Prior to each mission the command programmer analysts assembled (from several sources):

a) A simplified timeline sequence which included DSIF rise and set, sun rise and set, photo site times, and stored program transmission times.

b) A photography budget plan which included frame number assignments for photo sites, film processing time allocation, and photograph readout time allocations.

c) A core map plan based on items (a) and (b) which showed what would be included in each stored program transmitted to the flight programmers.

During the mission a planned sequence of events was issued about once a day by mission control (SFOD's staff). This was a computer generated 2 or 3 day schedule of activities which included ephemeris and photography events noted above, preliminary and final command conferences for reviewing proposed stored programs, special events such as RTC camera adjustments, attitude control updating, Canopus tracker cycling, command activity features such as the time FPAC must complete and submit for review their maneuver data, and the transmission time for each stored program.

The Flight Path Analysis and Control (FPAC) group performed orbit determination calculations using tracking data, computed 2 axis attitude maneuvers, finite burn duration, and ignition time for velocity maneuvers, computed 3 axis attitude maneuver magnitude and camera actuation times, computed data such as cone and clock angles which were used by SPAC for generating antenna pointing commands. The Subsystem Performance Analysis and Command (SPAC) group provided attitude control mode switching requirements, update maneuver requirements, antenna pointing and communication subsystem switching requirements, etc. These were reviewed at a preliminary command conference then used by the command programmer analyst to generate a specific core map.

The Command Generation Program (COGL) was used to convert this core map into a command assembly and store it in the SFOF for automatic transmission to the spacecraft. The command assembly, which was a listing of the command words for the core map in spacecraft computer language, was printed out and used to check the telemetry readout of commands as they were transmitted. These commands were also punched on paper tape and read into an engineering test model of the spacecraft computer where they were independently checked. The engineering test model then ran at real time duplicating the spacecraft functions which helped to verify the spacecraft computer. COGL also printed out a time tagged sequence which resulted from simulating the stored program and any real time commands that were included. The program recognized and indicated in the printout errors such as invalid command combinations and errors in commands.

The COGL printout was taken to the final command conference where it was reviewed by SFOD staff, representatives from SPAC, FPAC, and NASA mission advisors. Once approved, the commands were transmitted to the spacecraft.

The handling and checking of real time commands was also very formalized. Requirements for RTC's were usually generated by a subsystems analyst for housekeeping purposes such as antenna adjustments or attitude update maneuvers. Frequently they were used to make last minute changes to stored program maneuver magnitudes. Each RTC was written in spacecraft computer language by a command programmer analyst. It was checked by the SPAC director and approved for transmission by the SFOD. After approval it was read to the DSIF over voice link. After the command was set into the transmission equipment at the DSIF and before it was executed, it was read back by voice for verification. After verification it was executed.

The significant thing about this elaborate system was that no real time or stored program command was ever sent without extensive checking. Before an error could pass through, it would have to be independently approved several times. During the five Lunar Orbiter flights less than 10 incorrect commands were transmitted and executed. Examples of the kind of errors which did occur were:

a) An incorrect maneuver magnitude was transmitted.

b) A mistake was made in one extended mission program in which an RTC rather than an SPC code was placed on a stored program sequence. This resulted in the spacecraft trying to obey approximately 15 maneuver commands. This eventually caused loss of power, tumbling, and temporary dropout of the programmer. Fortunately, "random recovery" occurred.

c) A core map was transmitted which failed to interface properly with the current core map. The result was missing a maneuver and improper incrementing of a maneuver magnitude. A quick thinking command programmer analyst was able to generate a sequence of RTC's which returned to the proper program sequence.

d) After a series of experiments during extended mission, the spacecraft was erroneously left at an "off-sun" attitude with the spacecraft drifting away rather than toward the sun. This caused loss of power and tumbling. Here again "random recovery" occurred but with a serious loss of nitrogen gas.

e) An emergency RTC was set into the "command drawer" at the DSIF during a maneuver for execution immediately after the planned maneuver. The command was executed too early and interrupted the maneuver angle resulting in too small a maneuver angle.

7.5 PHOTO SITE ACCURACY ANALYSIS

A post flight data analysis has been conducted under Contract NAS 1-7954 to determine the accuracy in the knowledge of the locations of the photographs taken during the five missions. It was determined that the best accuracy in knowledge of site location was one km. The principal cause of these location errors was due to orbit determination errors, and the secondary cause was due to attitude control errors.

Orbit determination errors could have been reduced with a better modeling of the universe. This applies particularly to the lunar gravitational potential model. A gravitational series expansion about the surface rather than about infinity might prove helpful.

Attitude control errors could have been reduced with a tighter operational control of attitude. Long periods of operation on gyro references rather than on celestial references resulted in significant body axis drift.

A calibration procedure which included a direct measurement of the geometry of the camera's field of view while installed in the spacecraft would have simplified the post flight data analysis. Although the camera optical axis was measured as installed on the spacecraft by a mirror attached to the center of the camera lens, this did not define the actual alignment or location of the photo frame mask. For example, the corner locations of the photo frames could be obtained only by analysis of the drawings and assembly dimensional tolerances.

7.6 CONCLUSIONS AND RECOMMENDATIONS

The conclusions from this study of the Lunar Orbiter guidance concept are:

1) The Guidance concept employed on Lunar Orbiter was adequate for the task.

2) The approach taken to establish tolerances on components was more expedient than wise in a number of cases and involved added cost to the program.

3) All velocity control maneuvers were accomplished within the specified control criteria.

4) Operational software used in calculating orbit insertion maneuvers should have the capability to target to a wide range of orbit parameters. Where there is a discernible difference between an impulsive and a finite burn maneuver, the finite burn calculation should be targeted to the nominal design conditions.

5) A better definition of the lunar gravitational potential model would significantly reduce orbit determination errors and errors in forwarding the spacecraft state.

6) Attitude control errors could have been reduced with a tighter operational control of attitude.

7) Flight operation and command generation activities which involve many people make the result vulnerable to human error. A major problem is boredom during routine operations. A well disciplined and highly motivated crew are essential ingredients for success. They must work within a framework of checks and balances; provided with communication procedures and management with rigid control of changes to the basic computer programs.

Recommendations for "doing differently if one had it to do over" are:

1) Guidance hardware requirements should be established prior to releasing procurement specifications by an error analysis to show the impact of gyro drift rate, maneuver integration accuracy, limit cycle deadband, and sensor alignment.

2) Back-up procedures which are devised to meet emergency conditions should not be adopted as standard operating practice on subsequent spacecraft flights without assessing the effects. For example, long periods with attitude references to gyros instead of Sun and Canopus, introduced errors in photographic coverage and inaccuracy in photo site locations.

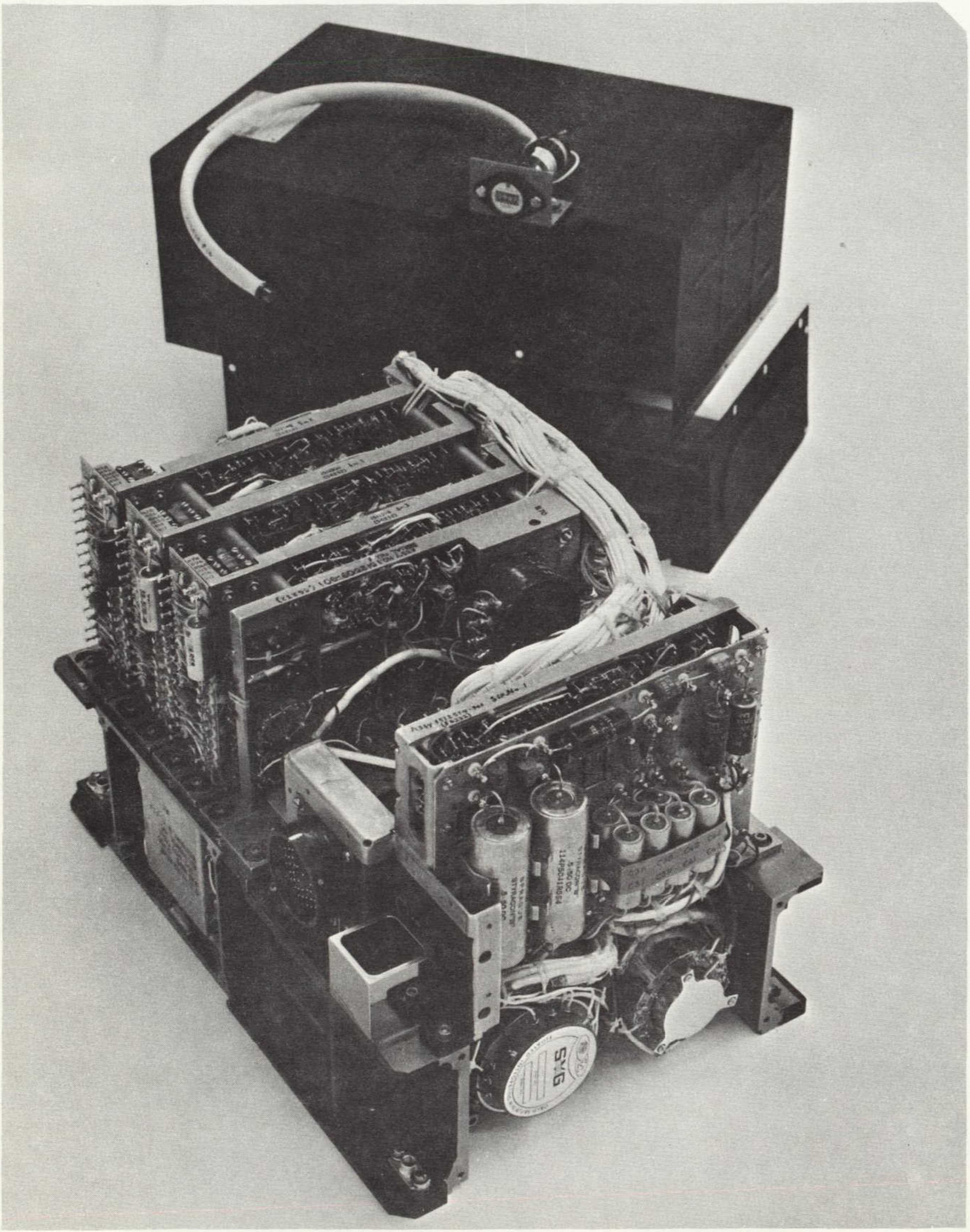


Figure 8-1: INERTIAL REFERENCE UNIT

8.0 COMPONENTS

A component level review of the guidance and control system was made. Included were the major hardware items as follows:

1. Inertial reference unit

2. Sun sensors

3. Canopus star tracker

4. Flight electronics control assembly-programmer, switching assembly, and closed loop electronics

5. Thrust vector control actuator

Since each of these components functioned as a part of one or more of the subsystems, the discussion which follows emphasizes the component aspects not covered previously. It is convenient to divide the discussion of the flight electronics control assembly into two sections: one covering the programmer functions including the switching assembly; and the other covering the closed loop electronics.

8.1 INERTIAL REFERENCE UNIT

The inertial reference unit (IRU) was an integral part of the attitude control, thrust vector control, and the velocity control subsystems described in Sections 3.0, 5.0, and 6.0, respectively. It was mounted on the spacecraft equipment mounting deck and aligned to the spacecraft control axes.

8.1.1 IRU DESCRIPTION

The Inertial Reference Unit (IRU) for the Lunar Orbiter spacecraft was a package containing three strapped-down gyros for attitude measurement, an accelerometer for thrust cutoff, and their associated electronics (see Figure 8-1). The gyros were single-degree-of-freedom, floated, rate-integrating units which provided angular rate or position error signals in three primary orthogonal body axes. The accelerometer was a pulsed integrating pendulous unit which provided integrated acceleration information for the spacecraft along the roll axis. The IRU provided rate information for celestial reference acquisition, maneuver measurement, and celestial reference limit cycle. The IRU provided position error information during operation at maneuvered angles; e.g., a photo sequence, velocity maneuver, thermal relief or during celestial reference occultations. During velocity maneuvers the IRU also provided a signal proportional to the linear acceleration of the spacecraft to provide an accurate measure of velocity change during engine thrusting. Among the design criteria for the unit, reliability received top consideration.

Parts and components were selected that had been proven in earlier space or missile applications. The SYG-1000 gyro, used on an Air Force satellite built by General Electric, and the pulsed integrating pendulum accelerometer

(16PIP) are examples. The 16 PIP was rated for manned spaceflight by NASA for the Apollo program. The minority of parts for which no experience existed were individually qualified specifically for the IRU. Materials and processes, too, were individually screened for reliability in the design. The next criterion was low weight and power. This led to the beryllium frame, extremely compact electronics packaging, and the design of a number of low-power circuits.

8.1.2 DESIGN REQUIREMENTS

The attitude, reaction, thrust vector, and velocity control subsystems set the requirements for the functional operating modes of the IRU. These are summarized as follows:

1) Rate Mode - provide an electrical signal proportional to the spacecraft angular rate about the three orthogonal spacecraft axes.

2) Rate Integrate Mode - provide an electrical signal proportional to the spacecraft angular displacement about the three orthogonal spacecraft axes.

3) Velocity Mode - provide an electrical signal proportional to the linear acceleration of the spacecraft along the spacecraft roll axis.

The specific IRU performance requirements for these modes of operation are listed in Figure 8-2.

8.1.3 DEVELOPMENT AND OPERATION

Floated rate integrating gyroscope and integrating accelerometers capable of meeting the performance requirements for Lunar Orbiter were within the state-of-the-art at the time of the proposal. However, there was no "on-the-shelf" package which contained the three gyros, accelerometer, and electronics within the desired weight and power limitations. Initially the weight objective was 9.1 lbs; the power was 22 watts. Both were approximately half of what was available from existing equipment. (The actual weight turned out to be 13.4 lbs and the upper limit on power was 30 watts with heaters on.)

It was necessary to request proposals from inertial instrument vendors for an IRU tailored to these requirements. There were five vendor replies to the Inertial Reference Unit request for proposal. Those responding were Kearfott, Minneapolis-Honeywell, Nortronics, Reeves, and Sperry. Each input was evaluated on the basis of performance characteristics, predicted reliability, electrical power, mechanical characteristics, quality assurance, and cost. Both the Sperry and Kearfott proposals were judged to satisfactorily meet the requested design objectives. The Sperry Gyroscope Company of Great Neck, New York, was selected to build the IRU even though there was no Boeing experience with the SYG-1000 gyro. Their proposed cost was lower which was decisive in the selection. Conservative, proven design techniques were proposed to be used. The difficulties later encountered with manufacturing yield in the instruments were not anticipated. Boeing's research, development, laboratory, and test work on attitude control systems prior to the

	RATE MODE	RATE INTEGRATE MODE	VELOCITY MODE
Nominal Scale Factor	4 Volt/Deg/Sec	4 Volt/Deg	0.1 Ft/Sec per pulse
Linear Range	$\pm$ 3 Deg/Sec	$\pm$ 3 Deg	$\pm$ 20 ft/Sec^2($\leq$ .6"g"
Saturation	3 Deg/Sec Also withstand $\pm$ 5°/Sec for 30 minutes	$\pm$ 4 Deg	.6"g"
Error	.8°/Hr @ 0 Deg/Sec 2.0°/Hr @ .5 Deg/Sec 108°/Hr @ 3.0 Deg/Sec	.01° @ 0° .01° @ .25° .18° @ 3.0°	Total accelerometer Bias not to exceed 3 x 10^{-4} g in range of .087 g to .23 g
Threshold	.2°/Hr	.01°	
Resolution	1.0°/Hr @ .5°/Sec	.01° @ $\pm$ .5°	$\pm$ 1 pulse
Drift Rate	G sensitive $\leq$ 2.0°/Hr/G nor change $>$.5°/hr/g between measurements	G insensitive $\leq$ .5°/hr @ 0° $\leq$.7°/hr @ $\pm$ 2.0°	---
Dynamic Response	300 radians/sec if under damped 110 radians/sec if over damped	Time constant .0017 sec Nominal gain of 1	200 pulse/sec N.A.
Electrical Noise	123 MV RMS	1 MV RMS for f = .1 to 1000 R/Sec 10 MV RMS for f > 1000 R/sec	

Power 30 watts at 21 VDC & 36°F mounting flange temp. (includes gyro heater & 16 PIP power)

Weight 14 lbs maximum, 9.1 lb goal (actual was 13.4 lbs)

Alignment -.01°; orthogonal $\leq$.01° gyros & accelerometer input axes within .14°

Hi-Rel solid state electronics to be used.

3 SYG 1000 gyros, one 16 PIP accelerometer. Back-up IRU's used C70-2564 Kearfott Alpha gyros.

FIGURE 8-2 INERTIAL REFERENCE UNIT REQUIREMENTS time of the Lunar Orbiter proposal provided a strong engineering preference for the Kearfott Alpha gyro. The JPL Mariner experience with the Kearfott gyros had been satisfactory. The confidence gained from these experiences was put to good use later in the program.

8.1.3.1 MAJOR DESIGN EVENTS

A preliminary design review was held in September, 1964. Significant findings of the PDR meetings included the following items committed for further action: (1) IRU parts list and qualification status, (2) thermal tolerance study, (3) IRU mounting interface thermal and alignment considerations, (4) reliability analysis at the module level, (5) interpretation of soldering requirements, (6) primary power source impedance, (7) accelerometer and gyro traceability requirements, and (8) review of test plans.

A critical design review was held in December of 1964. During these meetings it was brought out that design changes had been made to the SYG 1000 gyro and to the 16 PIP accelerometer. The changes were as follows:

SYG 1000 Gyro

<u>Internal Changes</u>

- Fluid changed due to change in operating temperature
- Fluid change resulted in a change in the internal balance weights which increased the g sensitive adjustment resolution to $\pm$ 1°/hour.
- Change in limit stops added a pin

<u>External Changes</u>

- Larger flange diameter
- Changed heater winding
- Changed nameplate

<u>16 PIP Accelerometer</u>

- Enlarged volume compensator
- Secondary seal
- One temperature probe removal

All internal changes

As a result of these changes, Sperry was directed to supply a list of all drawings, engineering bulletins, and test specifications, designating latest revision letters and differences between the past and the Lunar Orbiter version of gyro and accelerometer. Also, because of the gyro fluid change, a description of the controls for the acceptable contamination level of the gyro fluid was required.

The other major item evolving from the Critical Design Review required Sperry to submit a deviation request for non-compliance of the gyro, accelerometer and IRU electronics to the 200-Y specification on soldering techniques.

Assembly and test of the first Inertial Reference Unit built to the flight hardware drawings began at Sperry early in 1965. The first serious out-of-specification condition occurred on the engineering prototype unit while it was undergoing Flight Acceptance Test (FAT) at Sperry. The unit showed large changes in its "Rate Integrate Mode" scale factor. A series of investigative tests were performed on the breadboard IRU at Boeing in order to determine an explanation for the changes. The breadboard gyro "RIM" scale factor was found to be sensitive to thermal environment even though the gyro temperature control circuit was functioning as designed. This sensitivity was attributed to changes in the internal gyro temperature distribution, in response to environmental changes, which caused gimbal gain changes. It was predicted that a significant change in RIM scale factor would occur during the thermal-vacuum portions of IRU FAT.

By August 1965, a pattern had been established which indicated that a significant problem existed in the performance of the gyros used in the Inertial Reference Unit. At that time, the IRU's had failed to meet performance requirements in g-insensitive drift rate, g-sensitive drift rate, noise, rate mode scale factor, and rate integrate mode scale factor. The gyros had exhibited bad pivots, excessive noise, high mass unbalance, high g-insensitive drift rates, heater loop oscillations, and excessive contamination of damping fluid. Null shifts and gain changes were also occurring within the 16 PIP accelerometer and its associated electronics.

A series of Boeing-Sperry meetings were held to resolve the performance problems. While action was being taken to solve the problems of the "mainstream" IRU program, Boeing also began to consider a "backup" IRU program in which Sperry SYG 1000 gyros would be replaced by Kearfott Alpha 2564 gyros. Technical justification for considering this action was as follows:

1. The Sperry SYG-1000 and Kearfott Alpha 2564 gyro both derived their salient design characteristics from a G.E. specification relating to an Air Force spacecraft program, for which Sperry had delivered 155 units, and Kearfott, at the time of backup IRU gyro selection, had delivered 105 units. The Kearfott gyro therefore was similar to the SYG 1000, and could be incorporated in the L.O. IRU with only a change in thermal insulating washer and a change in envelope emissivity.

2. The Kearfott Alpha 2564 had an outstanding record of 95 percent yield on the above-mentioned G.E. program.

3. The Kearfott Alpha 2564 gyro had 2.2 times the angular momentum compared to competitive instruments in its weight, power and size class, and was therefore fundamentally less sensitive to production imperfections affecting major performance parameters (drift, stability).

4. Use of the Alpha 2564 gyro would result in a total power saving predicted to be 4 1/2 watts in the backup IRU, due to a smaller spin motor power consumption and higher floatation temperature. The

thermal design required that the equipment mount deck serve as the heat sink for the IRU. The deck temperature varied from 35°F to 85°F (nominal design) primarily depending on outside radiation. The gyro thermal control concept was for the heater to be off at the top deck temperature, 85°F, and full on at 35°F. The key item in the thermal resistance path, which was different for the Sperry gyro (145°F) and the Kearfott gyro (165°F), was the insulating washer through which the spin motor power was dissipated with the deck at 85°F. Since the Kearfott gyro spin power was less and the delta temperature from gyro to deck was greater, the thermal resistance was also greater for the Kearfott gyro. Hence, the heater power required at 35°F was also less for the higher floatation temperature.

In December 1965, Boeing recommended a backup IRU with substitute gyroscopes for the first flight spacecraft. The budgetary commitment was limited pending verification of compatibility of substitute gyros and/or continued loss of confidence in the SYG 1000 gyros. In March 1966, NASA directed Boeing to go ahead with the backup program. This program was implemented by diverting three (3) Inertial Reference Units from the mainstream SYG 1000 gyro program to the backup program. These units were to be modified by Sperry to incorporate the Kearfott gyro and replace the 16 PIP accelerometer with the accelerometer designed by Sperry for the Polaris program.

During this same period of time (January 1966 - June 1966) solutions were being found to many of the mainstream IRU problems. Test procedures were being revised and test procedure tolerances widened as a result of specification requirements being relaxed.

A summary of problems and solutions associated with the SYG 1000 gyro is given in Figure 8-3, the 16 PIP accelerometer in Figure 8-4, and the IRU Electronics in Figure 8-5.

As the backup IRU program reached production, units were found to suffer from the same heater loop instability at critical operating conditions as had occurred in the mainstream IRU's. Excessive gain in the temperature control loop was the cause of that trouble. However, the backup IRU's showed larger and unacceptable electrical noise on the output of the rate mode because of coupling of the heater oscillations through the power supply and into the pick-off excitation which was at 4.8 KHz. To eliminate these oscillations the heater modulation frequency was changed from 2.4 KHz to 9.6 KHz.

Failure of a "Polaris" accelerometer in a backup IRU and one in component test resulted in a recommendation from Sperry that this accelerometer model not be cooled below 75°F when inoperative. This limitation disqualified the "Polaris" type PIP from the Lunar Orbiter program due to a conflicting temperature constraint of the camera system which governed spacecraft temperature during handling operations prior to launch. The backup IRU program therefore reverted to using a 16 PIP accelerometer.

PROBLEM	CAUSE	EFFECTS	SOLUTION
o Mass Unbalance (Ground Test Only)	Change of fluid for 145°F float. Fluid absorption into float displacing air.	Results of ground tests changed from one test to the next.	New mounting insulator on gyro. Impregnate dynasyn to hasten fluid wetting of float. Recalibrate frequently to check on result of technique.
	Balance weight adjustment only in $\pm$ 1° per hour increments.	Original spec value of $\pm$ 1°/hr could not be met.	Increase tolerances based on realistic requirements.
o Gimbal Hangup (Ground and Space)	Fluid contamination Damaged Pivots	Loss of inertial hold reference	Revise filling procedure Revise balancing procedure Revise quality control
o Temperature Control (Ground and Space)	Heat sink varied from +35°F to +85°F nominal	Heater power required to maintain temp.	Increased allowable power
	Convection caused additional cooling at ambient test conditions	Power used was a function of the test conditions	Tolerances increased to allow for effect.
	Heater control loop gain caused instability	Electrical noise in gyro output	Lowered gain, added filters.
o Fixed Torque Drift (Ground and Space)	Unknown	Ramps in output Inertial hold or maneuver errors.	Screen the gyros and eliminate offenders. Increased tolerances.
o Life, ground and space	Bearing wear	Ground test time Reliability low for extended life	Lower operating temperature Float life demonstration test initiated. Frequent rundown test . Gyro and IRU burn-in. Limit maximum hours at launch to 2000 hrs.
o Total Yield		Launch delay	Implement backup IRU

FIGURE 8-3 SYG-1000 GYRO PROBLEMS

PROBLEM	CAUSE	EFFECTS	SOLUTION
o Null Shifts & Gain Changes	Improper Warmup Improper Orientation (Pendulum should hang down)	ΔV Accuracy Same	Close loop after warmup prevented excessive fluid forces on pendulum suspension. Store and warmup with output axis horizontal. This prevented pendulum from resting on stop which caused bias shifts to occur in the null.
o Slow Response (several minutes)	Bellows leak (bubble)	Same	Redesign bellows to lower stress. Bellows life test - low confidence in realism and results.
Total Yield	----	Launch delay	Increase tolerances

FIGURE 8-4

16 PIP ACCELEROMETER

PROBLEM	CAUSE	EFFECTS	SOLUTION
o Rate Mode Spikes	EMI	Maneuver accuracy	Filters
o Extra ΔV Pulses	Same	ΔV error	Same
o Temperature Hunting	Loop Gain	Gyro Noise	Lower gain; change PWM frequency

FIGURE 8-5

IRU ELECTRONICS

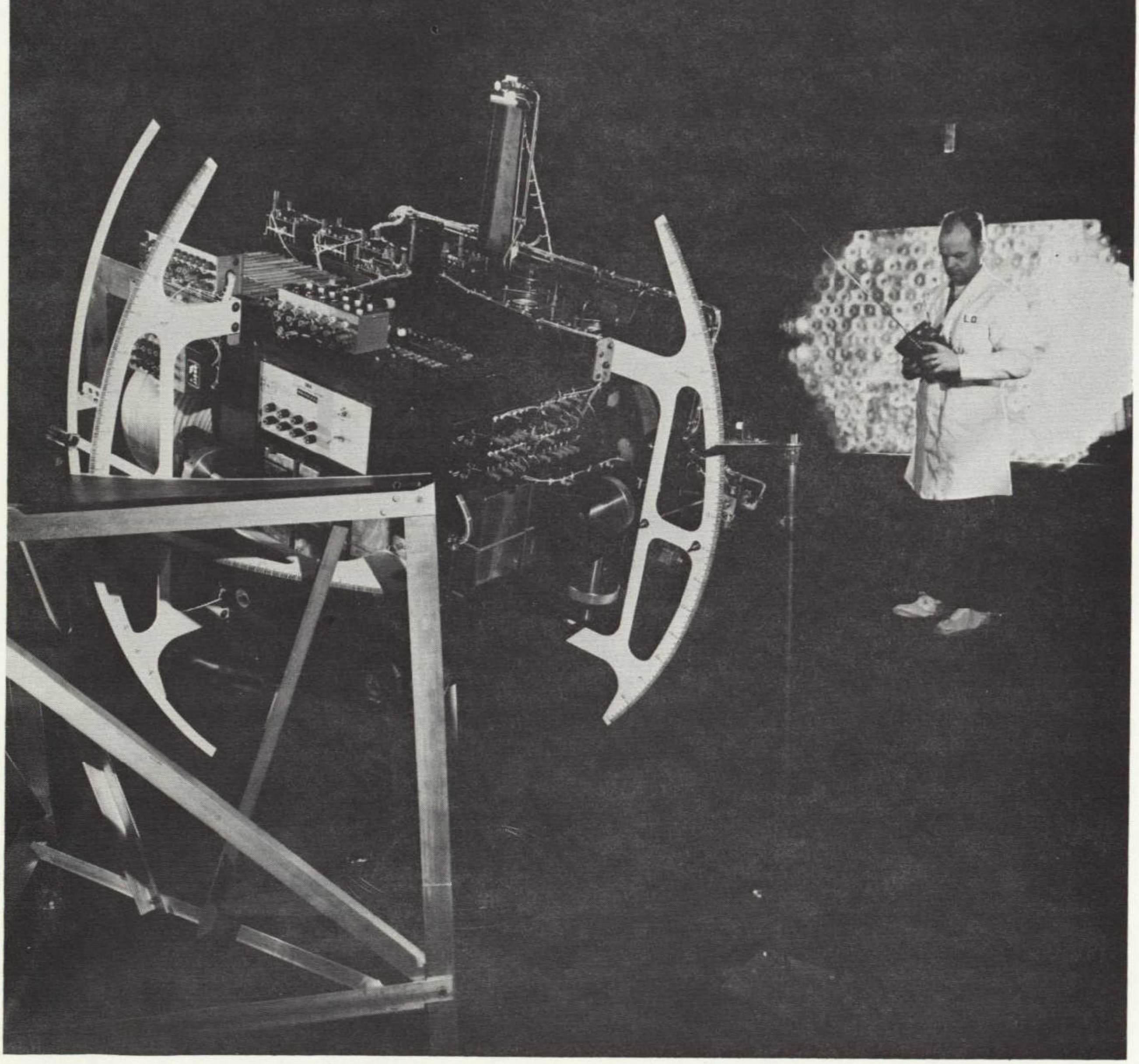


Figure 8-6: ATTITUDE CONTROL SUBSYSTEM
Airbearing Simulator

8.1.3.2 TESTING

An extensive test program was initiated that encompassed all levels of test. The formal portion of the test program included: (1) Flight Acceptance Tests, (2) a Qualification Test, (3) a Reliability Demonstration Test, and (4) an Incoming Acceptance Test. In addition to this formal test program there were gyro life tests, breadboard tests, air bearing simulator tests, and special post-mission tests. These tests and some of the significant results are presented in the following paragraphs.

A single axis breadboard IRU was made available to Boeing for test in late August 1964. It was evaluated in component and subsystem level engineering tests. Because of the question of life, a gyro life test program and a float life demonstration test was started by Sperry prior to the CDR in December 1964. The float life demonstration tests continued throughout the Lunar Orbiter program. During this same time period, Sperry built a three axis IRU breadboard for use as an engineering evaluation tool and for 3 axis development tests. Sperry conducted tests on the 3 axis IRU breadboard during the period 31 October through 19 November 1964. Test results and procedures were presented at the CDR in an effort to demonstrate that the IRU design would meet the specification requirements for rate mode and rate integrate mode. A 200 hour Boeing test program on the single channel IRU breadboard indicated that the unit would not meet all of the performance requirements. Later tests of the 3 axis breadboard at Boeing showed that the unit performed within specifications and large improvements in the electronics over the single axis breadboard were noted. IRU output noise and gyro heater oscillations appeared to remain as problem areas.

The 3 axis breadboard IRU was next used on a 3 axis air bearing platform simulation in conjunction with development testing of the Lunar Orbiter attitude control subsystem. The air bearing platform simulator (shown in Figure 8-6) contained all of the ACS hardware, including sea level thrusters, was mounted on a "friction-free" platform which simulated the vehicle inertias in pitch, yaw, and roll. These tests are discussed in more detail in Section 3.0.

The average time required by Sperry to perform a flight acceptance test on an IRU was originally 474 hours. After Boeing and Sperry teams revised these procedures, this figure was reduced 30 percent to an average time of 329 hours per unit. Many of the tests, such as the RIM Bias Drift Test, simulated the actual use of the IRU, while others, such as the tumble tests and low speed rundown time tests, were not strictly performance tests but provided confidence in the life and quality of the IRU.

A summary of problems occurring during component vibration qualification testing and actions taken to obtain an acceptable design are discussed in the following. When the inertial reference unit was subjected to the qualification vibration levels initially called out in the Environmental Criteria Document for all spacecraft equipment, high internal resonances were observed. Parts most affected were gyros, accelerometer, and electronic cards. Electronic parts also broke. Redesign included stiffening the circuit cards, cementing parts in place, and providing flexible leads where necessary. The beryllium frame could not be materially altered in the required time schedule to improve the accelerometer and gyro environment. Retesting repeatedly indicated that the IRU design would have great difficulty in passing the required 15 g sine vibration levels. Because of the evidence that these arbitrary vibration levels would not exist in the spacecraft environment, special tests were conducted of a dynamic model of a spacecraft with an instrumented IRU mounted on the equipment mounting deck. Although the vibration level input to the spacecraft was at the level required by the spacecraft vibration specification, there was considerable attenuation at the location of the IRU. This was especially true at the higher frequencies. On this basis the vibration envelope required for the IRU was adjusted to what the IRU could safely pass with the partial fixes incorporated.

These tests demonstrated that the actual magnification factors on vibration environment for the IRU as mounted on the spacecraft equipment mounting deck were less than originally estimated and specified. Therefore, the final vibration levels used for sine wave and random vibration tests were reduced as shown in Figures 8-7 and 8-8, respectively. In addition, it should be noted that flight data recorded even lower levels of acceleration.

Qualification Tests were conducted to determine that the design and fabrication procedures on IRU's were adequate to allow for expected variation in individual articles and environments, and that these variations did not compromise the required performance. Further objectives were to locate possible failure modes, and determine the effects of varied stress levels and combinations of tolerances on equipment performance.

Reliability Demonstration Tests were conducted in order to demonstrate two-mission capability by subjecting the system to the vibration and thermal environmental conditions and excitation voltage fluctuations expected during a launch and mission in two cycles with performance tests between cycles. The period of each of the two mission simulations was in excess of 30 days, the expected time for the primary mission.

Boeing conducted Incoming Acceptance Tests (IAT) on all mainstream "flight quality" IRU's prior to installation on a spacecraft. Flight Acceptance Tests, Qualification Tests and Reliability Demonstration Tests were conducted by Boeing on all "backup" IRU's according to the same requirements and procedures used by Sperry for the mainstream IRU's except that the RDT time periods were reduced to result in two 15 day simulated missions.

Figure 8-9 shows the failure rate (failure density, Ø, in failures per hour) as a function of test hours to failure on the IRU. The failures considered include those attributable to design deficiencies, part failures and manufacturing errors and deficiencies. Examination of the curve suggests that approximately 600 hours of test time (or burn-in) was required to ensure that the infant mortality rate portion of the curve had been passed.

8.1.3.3 MISSION PERFORMANCE

The primary photo mission of the Lunar Orbiter spacecraft required a 30-day life which was reflected in the IRU specifications. In addition an extended

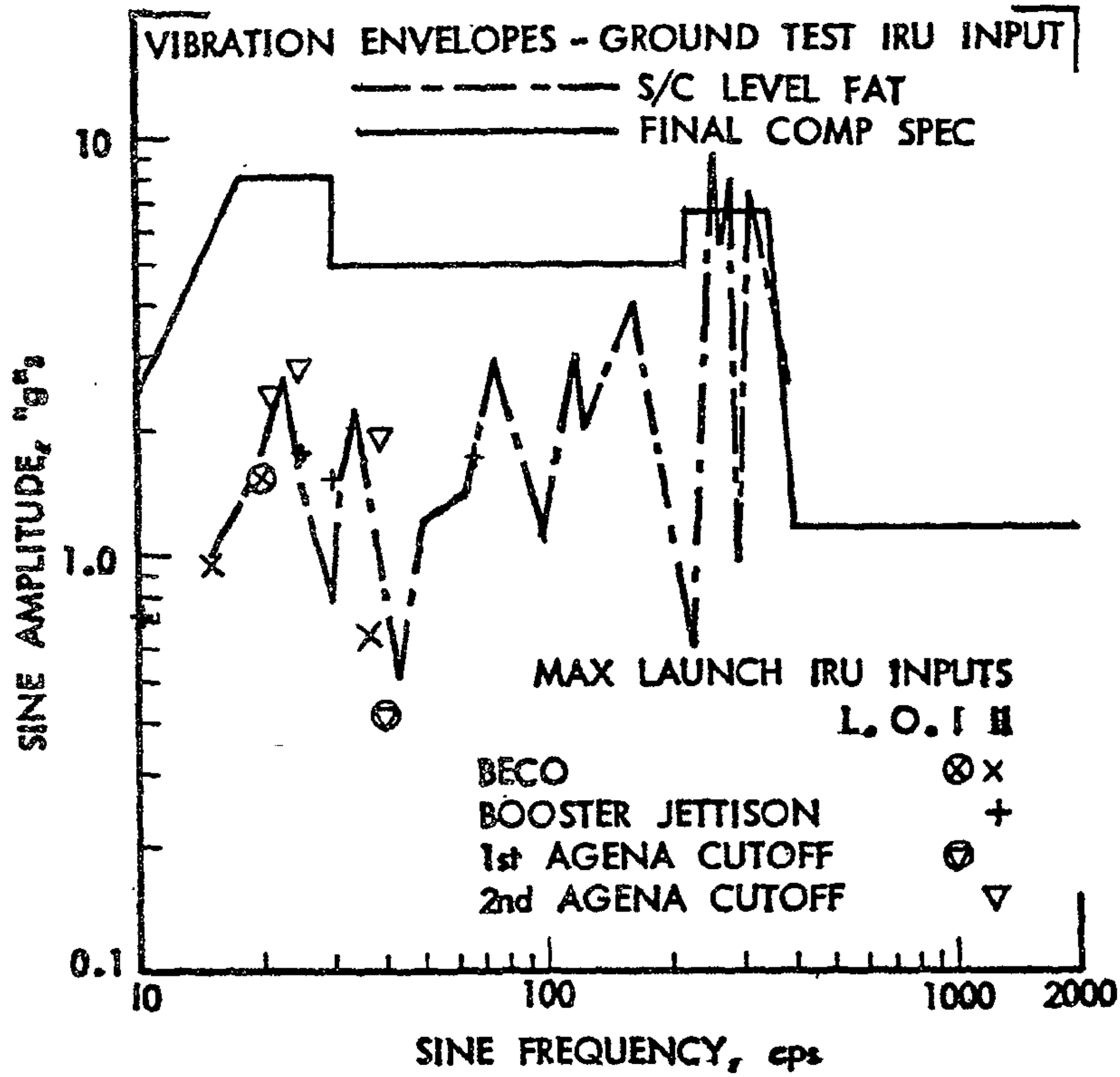


FIGURE 8.-7 IRU SINE VIBRATION ROLL (X) AXIS

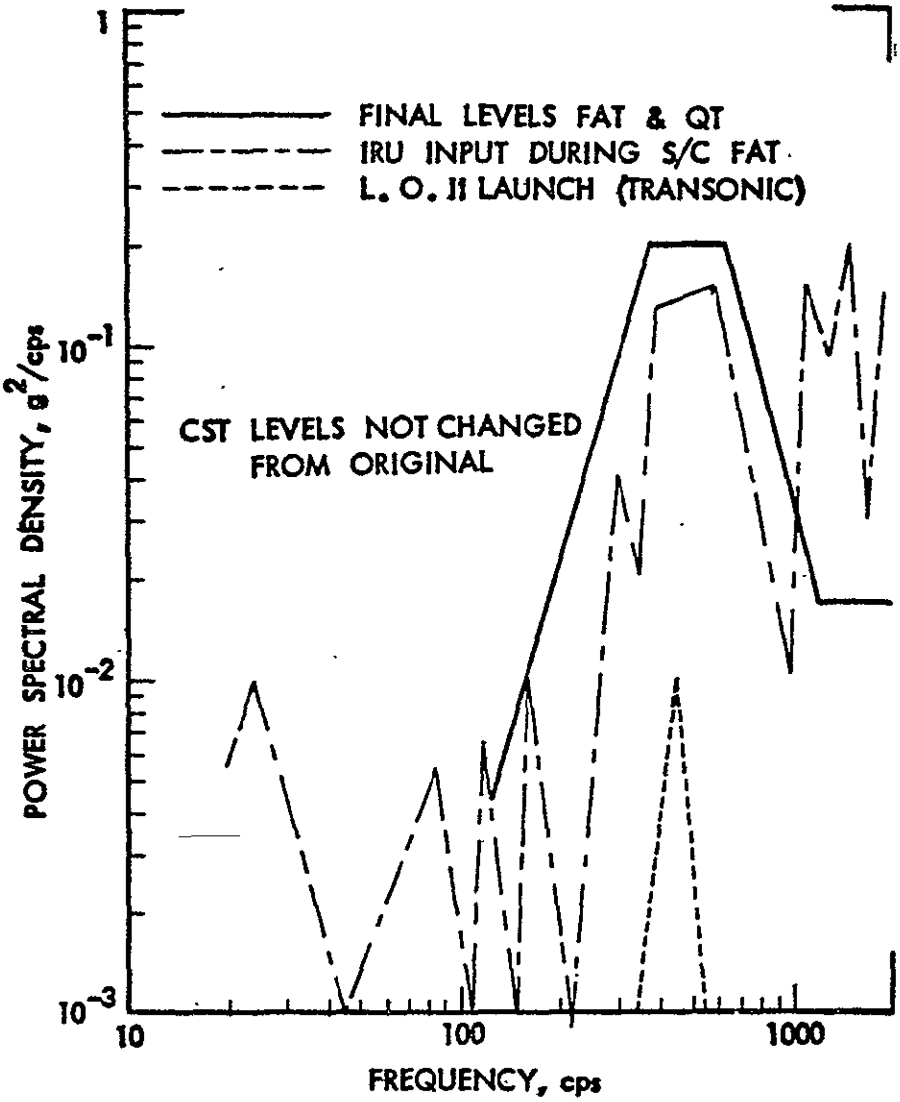


FIGURE 8.-8 IRU RANDOM VIBRATION

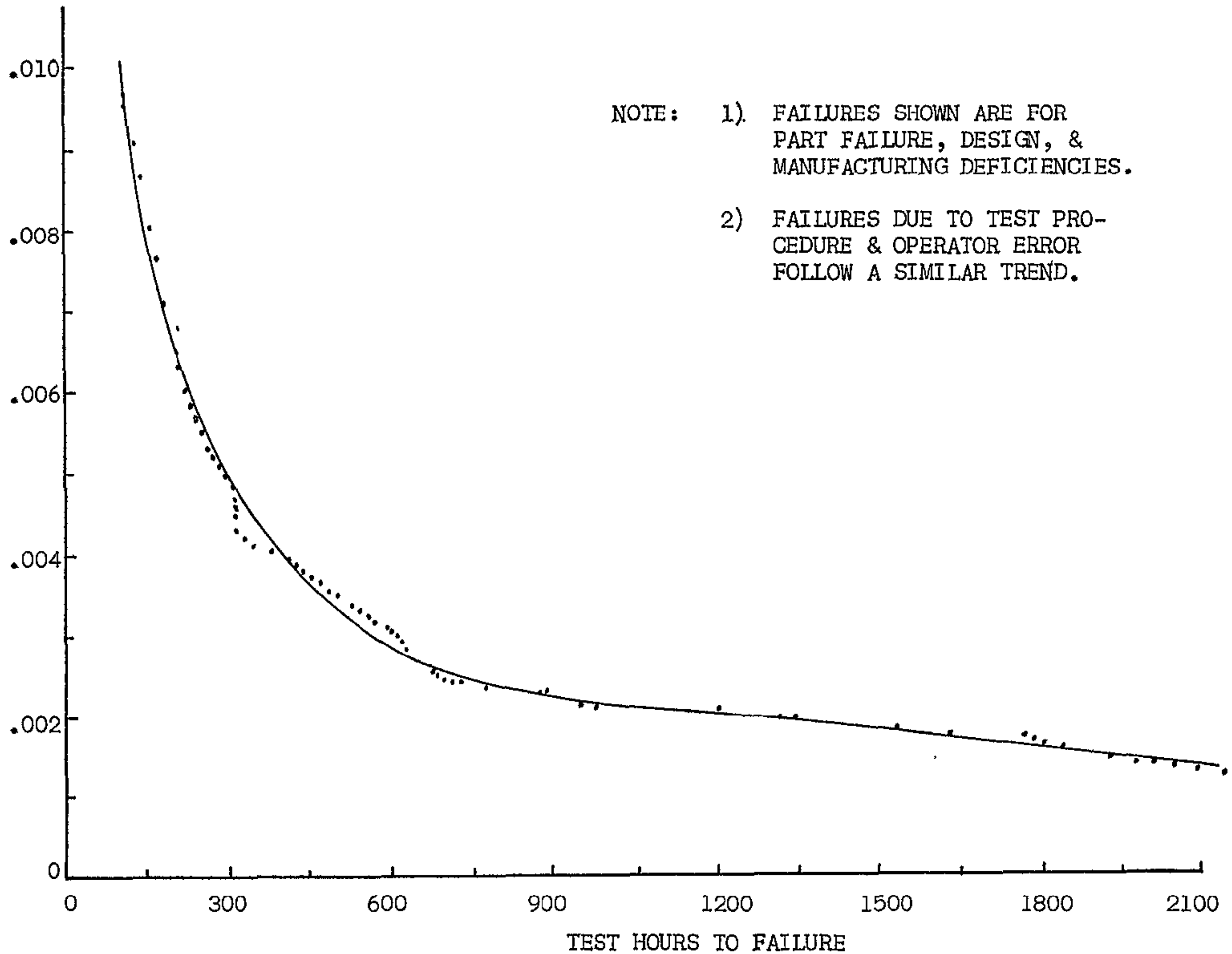


FIGURE 8-9

INERTIAL REFERENCE UNIT

FAILURE DISTRIBUTION WITH TIME

		OPERATING HOURS	START/ STOP CYCLES	GYRO MODEL
LO I	(SN 106 Mainstream) (Crashed 10-29-66)			Sperry SYG 1000
	o Ground test	864	349	
	o Space Flight	1914		
	o Total	2778	349	
LO II	(SN 109 Mainstream) (Crashed 10-11-67)			Sperry SYG 1000
	o Ground Test	1574	183	
	o Space Flight	8120		
	o Total	9694	183	
LO III	(SN 113 Backup) (Crashed 10-9-67)			Kearfott Alpha II
	o Ground Test	392	114	
	o Space Flight	5913		
	o Total	6305	114	
LO IV	(SN 111 Backup) (Last Communication 7-17-67)			Kearfott Alpha II
	o Ground Test	1087	212	
	o Space Flight	1703		
	o Total	2790	212	
LO V	(SN 110 Mainstream) (Crashed 1-31-68)			Sperry SYG 1000
	o Ground Test	1700	375	
	o Space Flight	4377		
	o Total	6077	375	

Total Space Hours

IRU - 22,027
GYRO - 66,081

Failures

Catastrophic - 0
Degradation - 1
(Gimbal hysteresis during spacecraft limit cycles with rates in the range of .001 deg/sec.)

Average Power used during thermal vacuum test

22 watts @ 85^{o}F
27 watts @ 35^{o}F
Sperry SYG 1000 Gyros

19 watts @ 85^{o}F
24 watts @ 35^{o}F
Kearfott 2564 Gyros

FIGURE 8-10 IRU OPERATIONAL LIFE

mission life was realized which substantially exceeded the minimum requirement. Space flight operating time ranged from a minimum of 1914 hours on L.O. I to a maximum of 8120 hours on L.O. II. A summary of IRU operational life, given in Figure 8-10 shows the individual IRU operating hours, total gyro space hours and type of gyro (Sperry or Kearfott) in each IRU. Because of the manufacturing and schedule problems with the Sperry SYG 1000 gyro, the Kearfott Alpha was used as a backup and flown on two of the five missions. There were no catastrophic mission failures and only one indication of performance degradation which was with a Sperry gyro. During the mission of L.O. II the roll reference attitude was lost by a slow drift while in the rate integrate mode. It was believed that this failure was due to an intermittent gimbal hangup. This gyro had been recommended for replacement following erratic behavior in Boeing tumble tests, but, because of schedule problems, it was flown. Mission success "proved" this to have been a good gamble.

There were two other flight anomalies which recurred on IRU's during flight but had no discernible effect on performance.

Telemetry data from L.O. II, IV, and V indicated intermittent heater loop oscillations similar to those observed during ground tests. Usually these were associated with transient heating conditions, such as following an eclipse. The other anomaly observed on L.O. IV was a yaw gyro wheel current oscillation and an indicated temperature oscillation. There was no evidence that this affected the performance.

Lunar Orbiter gyro drift rate history is shown in Figure 8-11. The gyro drift rate data tabulated are the Rate Integrating Mode (RIM) or attitude mode g-insensitive drift rate measurements on the ground (last bench value) and in space (space flight average value). The elapsed time between the last bench value column and the space flight average value column is about two to four months, and the actual operating time about 100 to 300 hours. The degree of correlation between the ground and space data is excellent. The one exception was the yaw gyro for L.O. I. The fixed torque drift continued to increase gradually with time from the value of .34°/hour shown to 1.2°/hour by the end of Mission I. Fourteen out of fifteen gyros showed a drift rate stability ranging from 0.01 to 0.16°/hour.

The total solar eclipse of October 1967 was predicted to have a severe effect on spacecraft subsystems which were sensitive to low voltage of the electrical power system or low temperature. The gyro's of the IRU were of special concern. Special laboratory tests were made on a Lunar Orbiter inertial reference unit in October 1967. The tests revealed that warmup times after heater turn on and switching transients are problem areas when re-activating IRU's that have been turned off. The tests were conducted to simulate environmental conditions predicted for Lunar Orbiter Spacecraft numbers II, III, and V which were all in orbit prior to the solar eclipse. The effects of turning the IRU off during the eclipse and then turning it on again after the eclipse were determined. The test results were long gyro motor spin up time to synch, a shift in drift rate, and switching transients. These transients can cause undesired spacecraft attitude changes by commanding the reaction jets on. A decision was made not to turn Iner-

			G INSENSITIVE DRIFT, RATE INTEGRATE MODE		
			LAST BENCH VALUE*	SPACE FLIGHT AVE**	MAX DEVIATION FROM AVERAGE
			DEG/HR	DEG/HR	DEG/HR
ROLL	LO I	IRU 106	-0.39	-0.50	0.36
	LO II	IRU 109	-0.03	+0.19	0.12
	LO III	IRU 113	+0.075	+0.11	0.03
	LO IV	IRU 111	+.048	+.064	0.03
	LO V	IRU 110	-.083	+.045	0.75
PITCH	LO I	IRU 106	-0.16	-0.20	0.20
	LO II	IRU 109	-0.09	-0.13	0.48
	LO III	IRU 113	-0.134	-0.16	0.09
	LO IV	IRU 111	+.145	+0.15	0.07
	LO V	IRU 110	-.093	-.078	0.34
YAW	LO I	IRU 106	-0.34	-1.20	0.50
	LO II	IRU 109	-0.47	-0.32	0.36
	LO III	IRU 113	+0.075	0	0.09
	LO IV	IRU 111	-0.018	-0.08	0.03
	LO V	IRU 110	-0.038	-0.071	0.05

*Measured at null, R.I.M. drift test

** Average value observed in ± .2 degree deadzone

FIGURE 8-11 GYRO DRIFT HISTORY

tial Reference Units OFF or On during critical periods of operation. Also to avoid the possibility of losing control of spacecraft II or III it was decided to crash these spacecrafts before the eclipse. However, the IRU on Spacecraft III was turned off after the last propulsive maneuver (designed to crash the spacecraft on the moon) and then turned on again. The "off" transient caused a spacecraft angular rate of approximately 110 degrees/hour, which corresponds quite well with expected spacecraft response to the IRU transient data obtained during the October ground tests. Unfortunately, the telemetry link was occulted after turn on and before the "on" transient data could be obtained. The results of these laboratory and space flight tests indicate that the switching transients of IRU's (gyros) are quite complex and difficult to predict without experimental data. Spacecraft V survived the October eclipse without as much difficulty as expected and continued to function satisfactorily even though the equipment mounting deck temperature fell to 10 degrees F.

8.1.4 CONCLUSIONS AND RECOMMENDATIONS

The major conclusions drawn from this study of the Lunar Orbiter IRU are:

1) The IRU performed adequately for this application. There were 22,027 hours of space operation without a failure. The design would be applicable to other spacecraft now in design.

2) No life or wear out limitations were evidence for missions up to one year in length once the IRU's were committed to launch. Minor degradation in performance occurred on one roll gyro which exhibited gimbal sticking.

3) Ball bearing gyros of the quality class represented by the Sperry SYG 1000 or the Kearfott Alpha have demonstrated an MTBF of 14,650 hours with a confidence factor of 95%.

4) The non-g sensitive drift rate stability of 0.27°/hour (3σ) on the ground for 16 months was achieved without calibration. A 30 day stability of 0.1°/hour (3σ) was achieved. The laboratory measurement of g insensitive drift was duplicated quite closely in flight.

5) Consistent IRU performance (for both gyros and accelerometers) was achieved as a result of meticulous attention to a multitude of details which included:

   a) Cleanliness controls

   b) Parts and process control

   c) Skilled workmanship

   d) Careful and exact testing

   e) Test equipment control

   f) Experienced personnel

6) A comprehensive and thorough IRU test plan is mandatory for a successful program. This is the means by which a design reaches operational maturity. Performance tests which simulate the actual use of the IRU provide the only rational basis for comparison of performance on the ground and in space; e.g., measurement of drift rate in open loop (rate integrate) mode on the ground. Other types of tests, which may not be strictly performance tests, are needed to provide confidence in the quality of the IRU; e.g., tumble tests at the systems level.

7) A burn-in or minimum operating life of about 600 hours prior to launch was required to eliminate infant mortality failures in the IRU. An elapsed time meter should accompany each unit so as to have a direct measure of the operating hours.

The recommendations for "what one should do differently if one had it to do over" are:

1) An earlier examination of the real mission requirements for accuracy and environment could have saved both time and money. Requirements were relaxed only after it became evident that the IRU design was incapable of passing.

2) Gyro and accelerometer instruments for an IRU should be selected from an in-being production line producing units of the quality class required. Much of the problem associated with the SYG 1000 gyros and the 16 PIP accelerometers could be attributed to restarting at near zero on the manufacturing and test experience curve after a shutdown of production.

3) <u>Any</u> change from a previously approved article in an IRU instrument or a manufacturing process should be treated with extreme caution. Even damping fluid change can cause major yield problems.

4) Fixed drifts could be corrected in flight by providing a rate integrate mode trim capability.

8.2 SUN SENSORS

The sun sensors were one portion of the attitude control subsystem described in Section 3.0. The function of the sun sensors was to provide a reference attitude for the pitch and yaw axes.

8.2.1 SUN SENSOR DESCRIPTION

The sun sensors used on Lunar Orbiter were procured from the Ball Brothers Research Company and were derivatives of those previously used on the Orbiting Solar Observatory satellite (OSO). There were a total of five assemblies on each spacecraft, as illustrated in Figure 8-12, a main sun sensor located on the equipment mounting deck, and four remote sensors located at the four corners of the heat shield. The main sun sensor, shown in Figure 8-13, consisted of two groups of eyes, one for control of the pitch axis and the second for yaw axis control. Each group in turn was composed of a set of fine and coarse eyes, the field of view of the coarse being about 5 times as wide as the fine. The remotes were each a single eye, with two associated with each axis to provide a 4π steradian field of view. The sensing element was a silicon solar cell which may be compared to a current generator. The output current is proportional to the area of illumination and the angle of incidence of the sunlight. A properly chosen trim resistor produced a voltage across the resistor proportional to the exposure area and angle of incidence. The fine sensor utilized a changing illumination area to provide the linear high gain signal. The coarse eyes operated strictly as a cosine function of the angle of incidence and, therefore, had a lower gain.

The weight and dimensions of the components are tabulated below:

Main sun sensor assembly	W_t = .6 lb	Dimensions 1" x 4.4" Dia
Remote sun sensor (C3 coarse eye)	each .03 lb	Approx 1/2" x 1/2" Dia

The sun sensors were procured with long pig-tail leads attached, which permitted direct connection to interfacing units when installed on the spacecraft. This eliminated the need for connectors for each sensor.

8.2.2 DESIGN REQUIREMENTS

Requirements on the individual sensor were derived from the arrangement on the spacecraft and from the specific performance for pointing accuracy. The 4π steradian field of view was required to ensure sun acquisition from any initial attitude after separation of the spacecraft from the Agena. The fine sun sensor was required to be unaffected by secondary light sources (such as the bright limb of the moon) outside 20° of the null. There was considerable overlap provided by the array of sensors; e.g., coarse sensors of pitch axis and yaw axis are both illuminated for large angles off sun. Even with solar panels or high gain antenna dish arranged to shadow one set of eyes, the other set provided sufficient error signal to cause sun acquisition. The detail requirements are listed as follows:

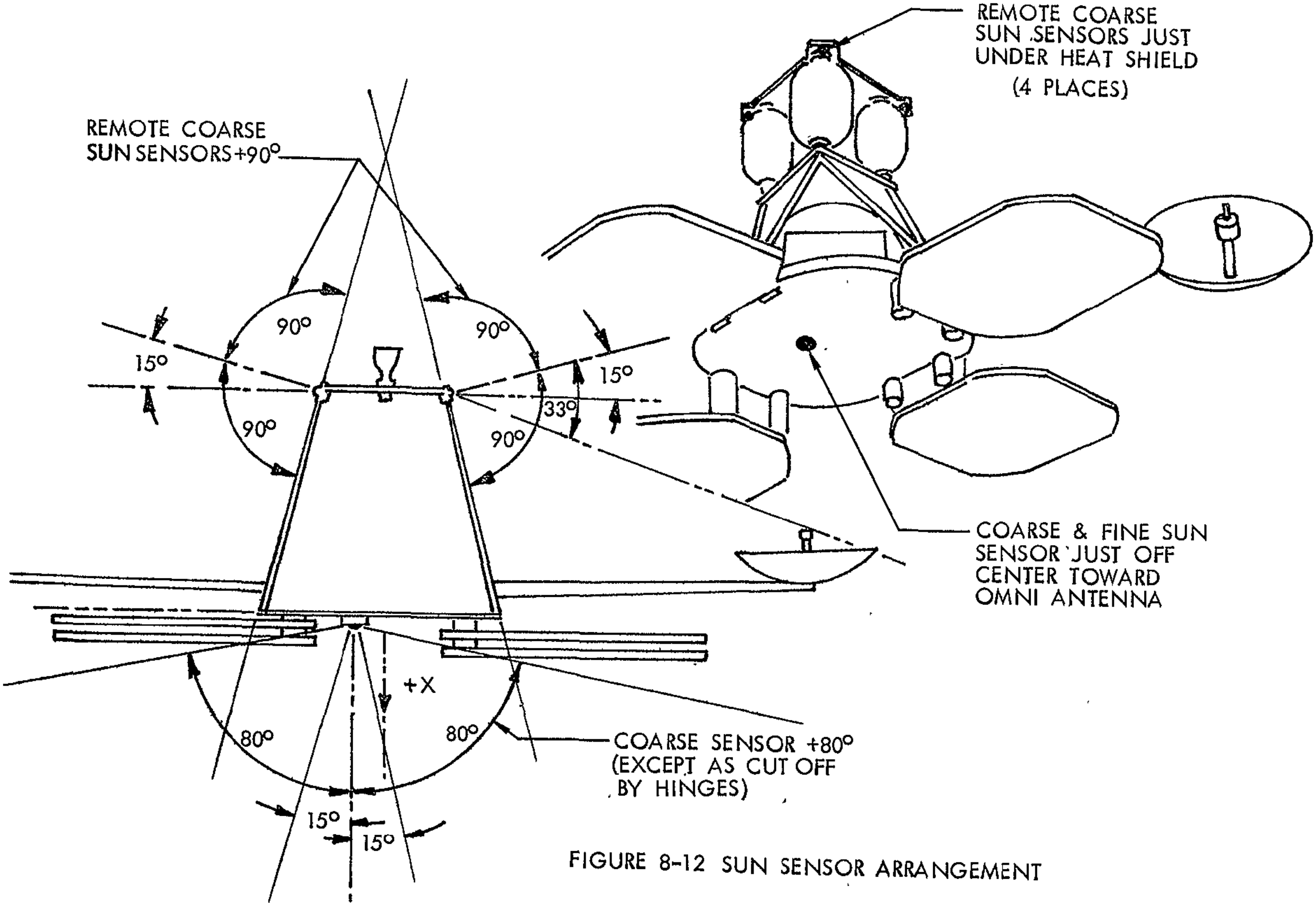


FIGURE 8-12 SUN SENSOR ARRANGEMENT

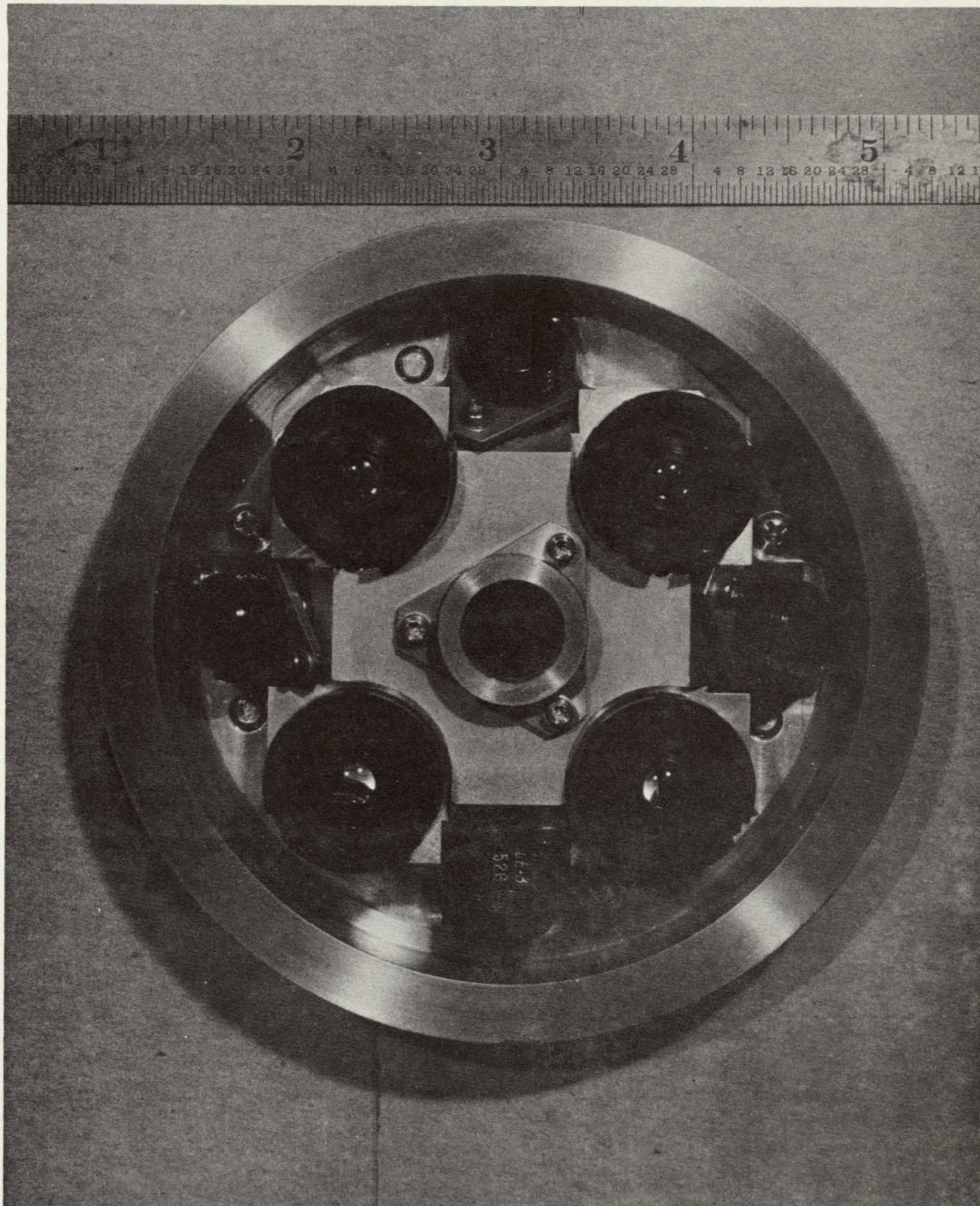


Figure 8-13: MAIN SUN SENSOR

MAIN SUN SENSOR

Field of View:

Fine eyes $\pm$ 15° Linear for $\pm$ 5°

Coarse eyes $\pm$ 80° Linear for $\pm$ 15°

Null Error: Fine sensor $\pm$ 0.017° pitch or yaw

Signal Output:

Fine eyes: within $\pm$ 0.25° of null = 12 mv/degree $\pm$ 0.6 mv/degree
within .25° to 2.5° = 12 mv/degree $\pm$ .9 mv

Coarse eyes: between $\pm$ 12° = 2 mv/degree $\pm$ 4 mv

between $\pm$ 10° $\pm$ 60° $>$20 mv

REMOTE SENSORS

Field of View: 2 Coarse eyes on spacecraft per axis $\pm$ 90°

A reference mirror was provided on the main sun sensor to provide alignment to the spacecraft. Alignment error of reference mirror to mounting surface was 18 arc seconds.

8.2.3 DEVELOPMENT & OPERATION

Two candidate approaches to the basic sun sensitive cell were available at the time of the Lunar Orbiter design. The Mariner & Ranger spacecraft had flown with cadmium sulphide photo resistive elements for sun sensors. In this method a sun shield casts a shadow, covering a portion of each cell. As the incident illumination changes angle, the area of each cell illuminated changes along with resistance. The cells were connected in a bridge circuit such that a voltage proportional to the attitude error was obtained.

The other method available was the silicon solar cell as used on OSO. The decision to use the silicon cell approach was based on the following:

1. Laboratory experience and familiarity - a silicon cell sensor had been used successfully on a Boeing air bearing attitude control simulator for over two years.

2. Signal level and power requirements - the silicon cell sensor had an output of about 10 millivolts per degree and therefore required amplification of 100:1 or 1000:1, depending on the deadzone desired. An amplifier with low noise and high gain was available from Ling Temco Vought requiring about 0.2 watts power. This was preferable to the cadmium sulphide cell which was used in a bridge circuit and required about 1 watt to produce the signal level of 10 volts/deg required for 0.2° deadzone.

3. Dynamic Response

Silicon Cell	.001 second
Cadmium Sulphide	.1 second

   In general, fast response was desired to preclude Reaction Control Jet "on" delay or "off" delay from causing a weight penalty. In the L.O. application it was not a significant factor since the sun sensor was used only as a position reference.

4. Availability - silicon cell sun sensors were available from Ball Brothers Research Company and the Bendix Corporation on a part number basis. Cadmium sulphide cells could be obtained from Nortronics Division of Norair (from Ranger experience) only on special order basis. There was no existing manufacturing source for "off-the-shelf" cadmium sulphide cells.

5. Reliability - solid state devices were to be used throughout both designs. From a part count basis the silicon cell approach was less reliable because of the added amplifier. However, an open or short circuit, or cell failure would have been passive and possibly allow a degraded mode of operation. The cadmium sulphide cell was simplest, but an open or short circuit failure could have produced a hard over signal to the attitude control.

6. Accuracy - the claimed null accuracy for the silicon cell was $.01^{\circ}$, later shown to be $.017^{\circ}$. The demonstrated accuracy of the cadmium sulphide was 0.1°. In addition, there was a physical problem of matching cells and mechanical parts which made this hard to achieve.

8.2.3.1 DEVELOPMENT PROBLEMS

The remote sensors were procured by part number as an off-the-shelf item and no development problems were encountered. The following discussion pertains to the main sun sensor.

SUN SENSOR TESTING

Because of the difficulties of simulating the true sun's illumination (geometry, intensity, spectrum, and particularly a stable centroid of illumination) the Sun Sensor was tested for performance under simulated space thermal vacuum environment as exposed to the real sun. This "real sun" was as seen at Boulder, Colorado (or Tucson, Arizona) through a quartz optical quality window in the thermal-vac chamber which was mounted on a sidereal table to track the sun. Since the sun at Boulder was approximately 80 percent of the sun as viewed from space, corrections were applied to the Sun Sensor outputs based upon the output of a "standard" eye looking through the same thermal vacuum chamber window. This standard eye had been calibrated previously by Ball Brothers Research Company against the space sun on an Aerobee shot and subsequently recovered. Optical means were provided for viewing the sun directly and at the same time measuring the error between the normal to the mounting surface of the test fixture and the sun's line of sight. Both the output and input were recorded on an X-Y recorder; the latter from a precision angle transducer mounted on the adjustable test fixture gimbal. This test was repeated for both the fine and coarse eye pairs. The peak output of the standard eye was recorded at the start and finish of the test of each axis; i.e., with the standard eye looking exactly at the sun. Typical calibration curves for the fine and coarse eyes are given in Figures 8-14 through 8-16.

The Flight Acceptance Test vibration test setup for the Sun Sensor was conventional except for the use of a light source to stimulate the eyes during vibration. This light source was a commercial photographer's light, Sylvania "Sun Gun." This light was used solely to stimulate the Sun Sensor outputs used in a "go-no-go" check for intermittents during vibration. This source was also used for incoming acceptance and spacecraft level tests at Boeing.

The major problem areas in FAT testing of the Sun Sensor were test equipment or procedure oriented.

1. "Unusual" weather for the Boulder, Colorado area reduced the available "good" sun time to such a low percentage that, late in the fall of 1965 an additional test facility was set up near Tucson, Arizona (a "Good" sun was defined as a sun with no clouds within 20 degrees of the sun line of sight).

2. It was found that great care had to be taken in cleaning and maintaining the precision mounting surfaces of the test fixture. Failure to keep these rigid requirements could lead to rejection of good Sun Sensors because of apparent misalignment between the reference mirror and the mounting plane.

3. Because of the limited space in the back of the main Sun Sensor, great care had to be taken in installing the trim resistors to make sure that they were fastened across the correct sensing eye. A peak output test as used for Incoming Acceptance Test verified this proper connection.

4. The white reflecting paint (JPL S-13) used on all surfaces to face the sun except the eye lens, was the source of two problems:

   (a) In the first few vacuum sun tests, this paint outgassed and clouded the quartz window with a projected image of the Sun Sensor (no paint deposited on the surface opposite to the eye lens which were not painted). A pre-performance test vacuum cycle to outgas the paint solved this problem.

   (b) The paint used was found to be very sensitive to proper mixing, application, part handling and the useful date life of the paint. New procedures were developed for proper care of the sensor during its entire ground test sequence to prevent paint chipping. Even with these procedures, there was a small amount of touch up necessary on the flight units.

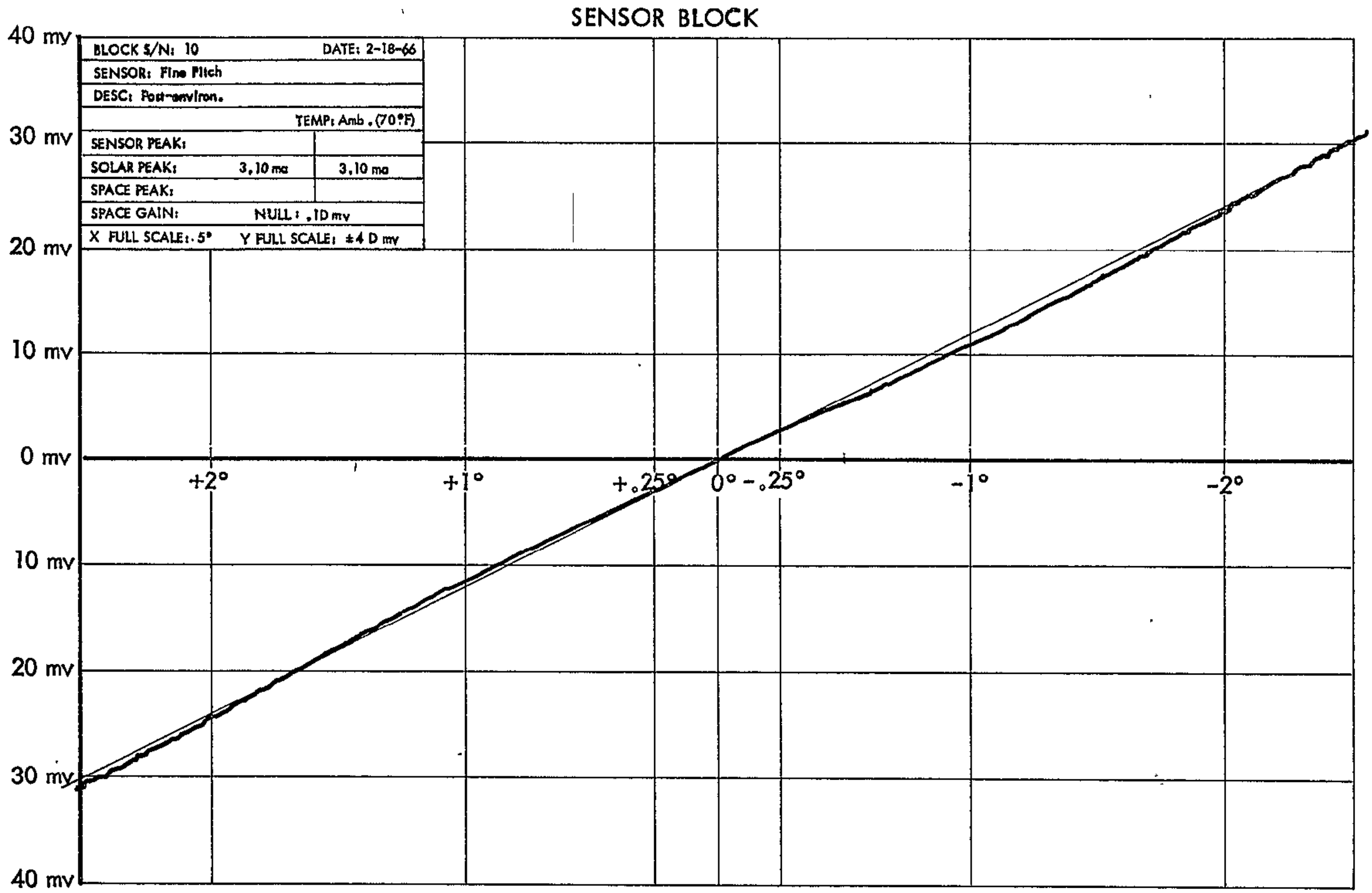


FIGURE 8-14 FINE SUN SENSOR SMALL ANGLE CALIBRATION

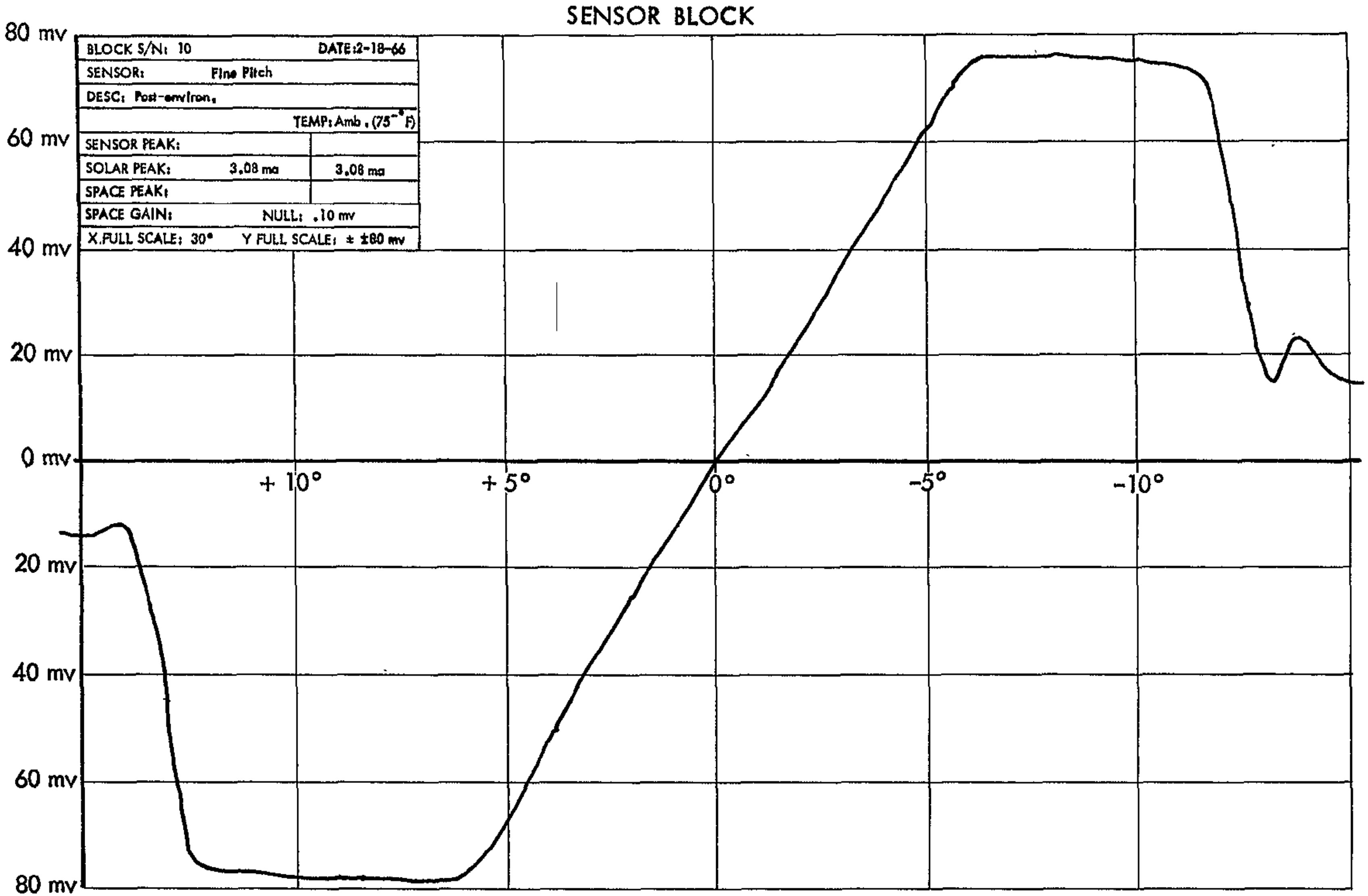


FIGURE 8-15 FINE SUN SENSOR WIDE ANGLE CALIBRATION

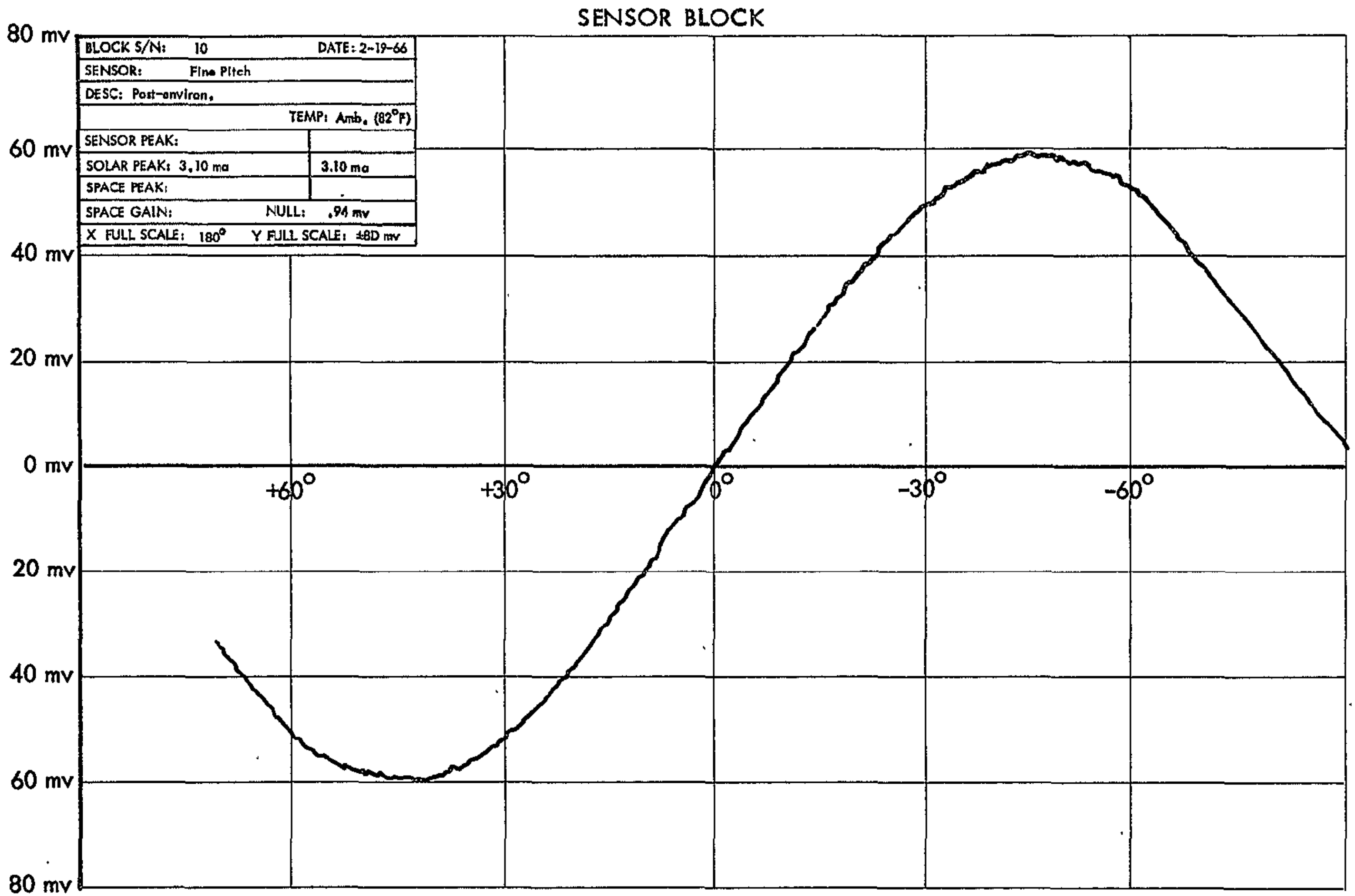


FIGURE 8-16 COARSE SUN SENSOR CALIBRATION

8.2.3.2 MISSION PERFORMANCE

The sun sensors provided an accurate celestial reference for a variety of non-nominal situations. Approximately 60% of the time was spent off the sunline and the capability of having fine, coarse, or both sun sensors in the loop proved extremely useful.

The initial sun acquisition took place automatically on all missions within the maximum allowable time of 20 minutes from separation.

Figure 8-17 shows a coarse pitch sun sensor calibration curve obtained during a thermal pitch maneuver away from the sun of Mission II. Figure 8-18 is a yaw coarse sun sensor calibration curve based on yaw updates during the readout phase of Mission II.

The coarse eyes proved to be very useful during photo readout on all missions. Nitrogen was conserved since the sun did not have to be reacquired every orbit. Pitch attitude during most of readout was estimated from the initial pitch maneuver, knowledge of pitch drift rate, and solar array current. Yaw attitude, however, was continuously monitored via the coarse sun sensor. Attenuation of the yaw output due to pitch was approximately the same for all missions. At a pitch angle of 30^{o}, yaw was observed to be degraded an average of 0.8. The predicted attenuation for this case is cos 30^{o} or 0.866. Also, on all missions, moonlight was seen to affect the coarse eye output during certain portions of an orbit, but with no bad effect on the mission.

8.2.4 CONCLUSIONS AND RECOMMENDATIONS

1. There were no flight problems with the sun sensors.

2. The aches and pains of obtaining flight qualified units according to schedule were considerable. A good calibrated artificial sun source would have saved calendar time as well as money. Substitution of a sun simulator for a Boulder, Colorado, or a Tucson, Arizona sun is recommended for future programs.

3. Thermal control coatings used on the sun sensor caused trouble because of apparent misalignment between the reference mirror and the mounting plane. Great care had to be exercised to maintain clean mounting surfaces.

4. Because of the limited space in the back of the main Sun Sensor, great care had to be taken in installing the trim resistors to make sure that they were fastened across the correct sensing eye.

5. The use of pig-tail leads on each sensor, to provide direct connection to interfacing units, eliminated 5 connectors. This method of connecting the sun sensors is recommended to reduce the total spacecraft connector count.

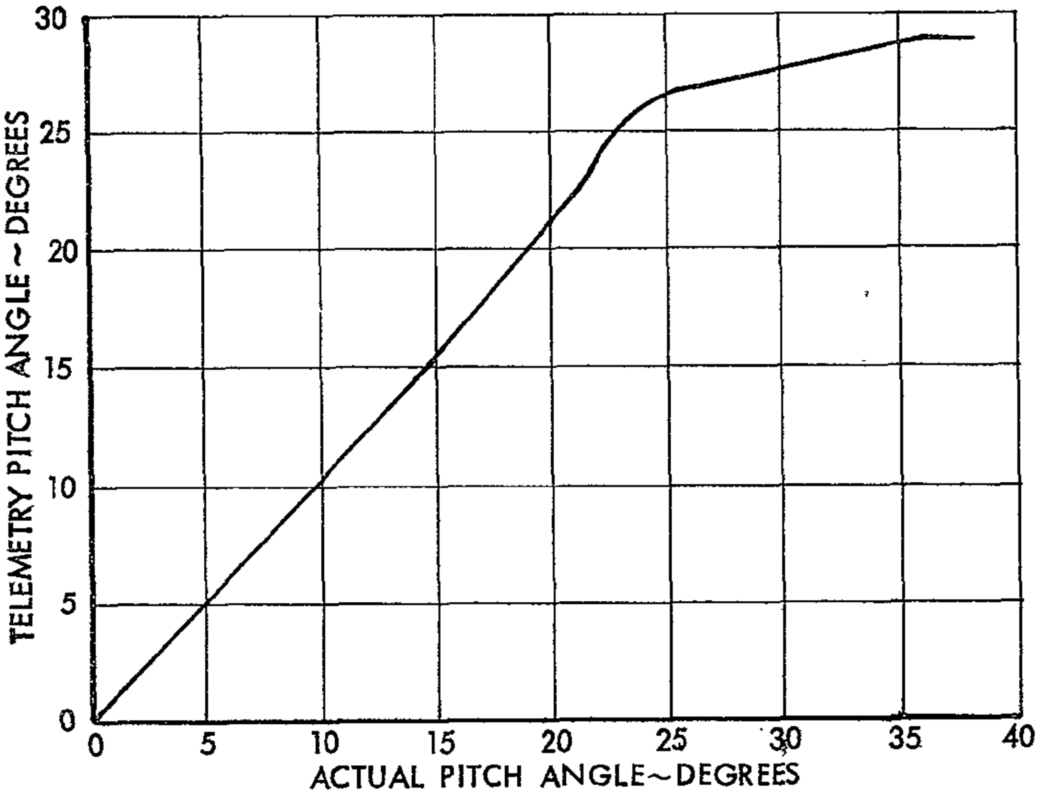


FIGURE 8-17 INFLIGHT COARSE SUN SENSOR CALIBRATION

NOTE:
MISSION II CURVES

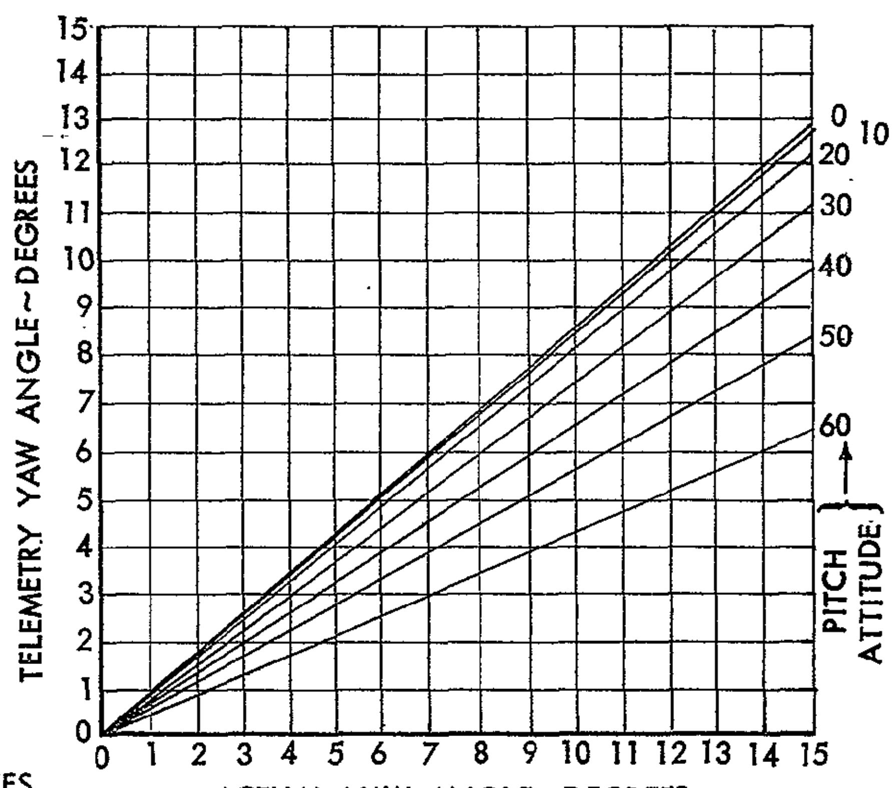


FIGURE 8-18 INFLIGHT COARSE SUN SENSOR CALIBRATION

Recommendations for "doing differently if one had it to do over" are:

1. The wide angle sun sensor array should be calibrated over its full range prior to flight. A linear wide angle telemetry output would have been useful for the unanticipated "off-sun" mode of operation employed on Lunar Orbiter.

8.3 CANOPUS STAR TRACKER

The Canopus star tracker (CST) used on the Lunar Orbiter program was developed by ITT, Federal Laboratories. The CST was a part of the Attitude Control Subsystem described in Section 3.0. It's primary functions were to provide: (1) a roll reference signal to the attitude control subsystem, (2) a star map signal which was proportional to the light flux entering the tracker for identification of Canopus via telemetry on the ground, and (3) a Canopus recognition signal to the programmer for mode switching of the attitude control roll reference.

8.3.1 STAR TRACKER DESCRIPTION

The primary element of the tracker was an image dissector phototube. A schematic is shown in Figure 8-19. A star was imaged by the lens upon the image dissector photocathode and an electron beam image was formed by focusing electrodes. Behind the photocathode was a narrow slot aperture which allowed only star image signals that passed through the aperture to enter the photo multiplier dynode chain to provide outputs. Deflection coils placed around the tube allowed electron beam images from the photocathode to be scanned across the aperture if appropriate currents were applied to the coils.

Through this means of electronically scanning the electron image of a star, phototube output was modulated to provide error signals indicating star position. The narrow slot aperture was aligned with the vehicle roll axis. The tracker electronics generated a 14 Hz triangular wave form to sweep the full field of view in "search" mode. When sufficient output was obtained from the photo multiplier to trigger the "track" mode threshold, the scan frequency was changed to 800 Hz and the scan amplitude reduced to $\pm 1^{\circ}$.

Star position information was obtained and used in an internal feedback tracking loop to average the electron star image about the center of the aperture. The actual magnitude of the DC deflection voltage required indicated star position with respect to the roll axis and this magnitude was provided as the output roll error voltage to the ACS.

The electron beam which passed through the aperture was amplified in the dynode chain. The resulting current was proportional to the brightness of the star which was imaged. This video output provided the star map voltage. A recognition signal was generated based on the star map voltage above 1.2 volts and below 4.4 volts. These levels corresponded approximately to 1/3 times to 3 times Canopus.

8.3.2 DESIGN REQUIREMENTS

The star tracker performance requirements (paraphrased from the latest revision of the specification) were:

1) To track the star Canopus and provide a roll reference signal for attitude control. The roll error scale factor was 1 volt per degree and was required to be linear to $\pm$ 2.5 deg and with an output over 2.5 volts to $\pm 4.1^{\circ}$. The tolerance allowed was $\pm$ 5%.

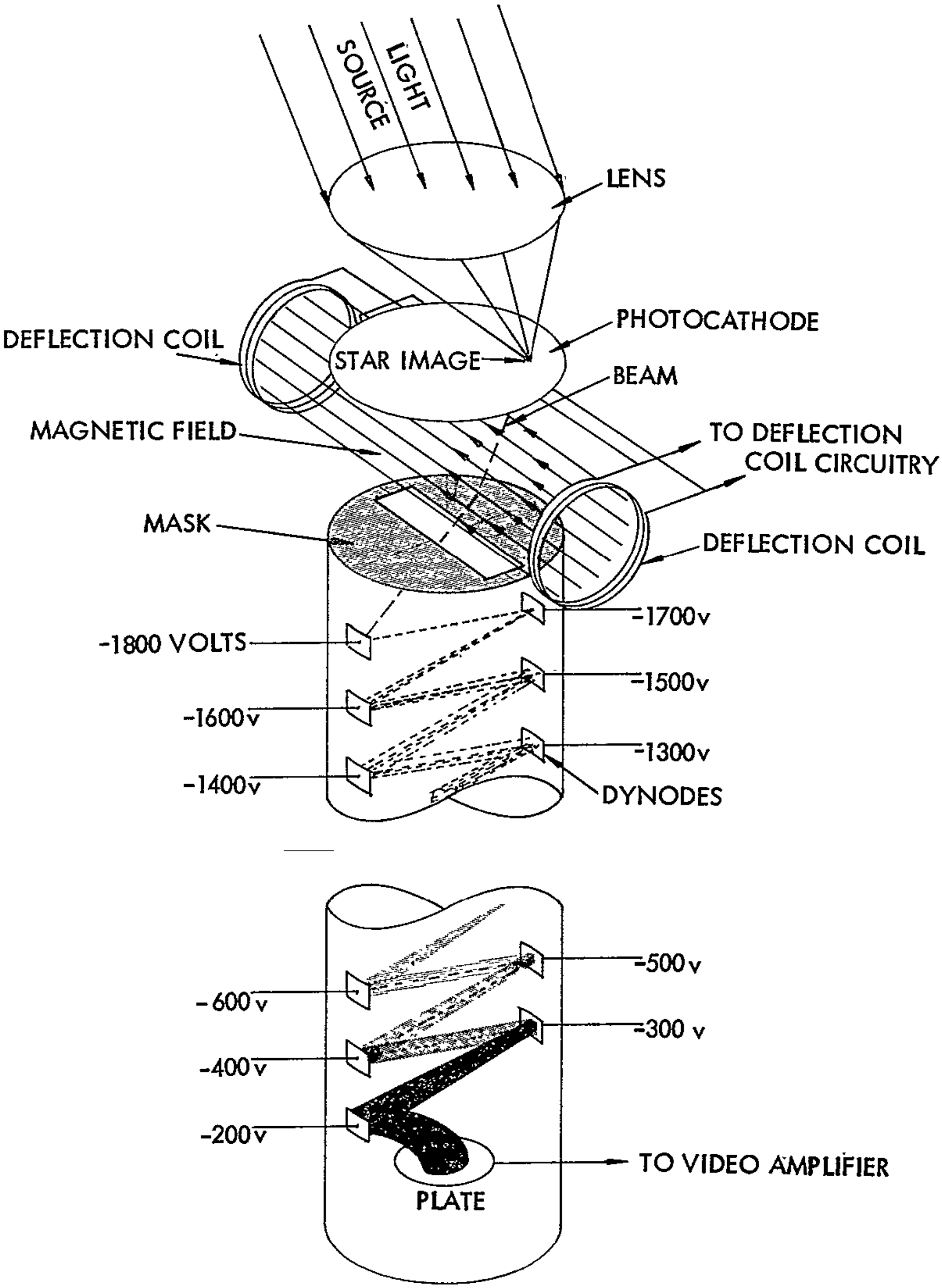


FIGURE 8-19 SCHEMATIC DIAGRAM OF IMAGE DISSECTOR TUBE AND LENS

2) To provide an analog star map signal to the telemetry subsystem which was linearly proportional to the light flux from the star being tracked. Operation after 6 hours in space and throughout the rest of the mission was required.

3) To provide a Canopus recognition signal whenever a star of the proper brightness fell within a specified ratio of Canopus. The purpose was to automatically discriminate against other stars. The recognition gate levels were first specified to be .5 to 1.5 X Canopus. Later this was opened up to .3 to 3 X Canopus. It was also tailored to assure a margin for the star map voltage outside the recognition levels.

4) To provide a roll field of view of $\pm$ 4.1 degrees. The roll field of view was to allow operation in a $\pm$ 2 degree deadzone before, during, and immediately after occultation without losing the star due to gyro drift while in the rate integrate mode.

5) To provide a yaw field of view sufficient to accommodate Canopus, for a 30-day photo mission plus the launch period associated with any given month. The yaw field of at least $\pm$ 8 was provided.

6) To provide a mechanical adjustment in the yaw plane (pre-launch) in 2 degree increments to provide a total yaw field coverage of 33.5 degrees.

7) To provide a bright object sensor and shutter to protect the photo tube from illumination greater than .03 foot candle. The shutter closure time was less than .5 seconds.

8) To provide dynamic response as follows:

   a) acquisition of a star after entering field of view $\tau \leq .5$ sec.

   b) roll error signal time constant $\tau = .14 \pm .05$ seconds.

9) Signal to noise ratio was required to be less than 24 to 1 at .1 degree roll error. The noise output was required to be less than 15 arc seconds at null. At maximum roll error the noise was required to be less than 100 millivolts RMS. The necessity to specify the noise at maximum roll error was not recognized in the original specification.

10) Tracking of Canopus to within 30 degrees of the bright limb of the moon was required.

11) The weight was to be less than 7.5 lb. The original goal was 5.5 lb.

12) The power was to be less than 3.0 watts at 21 volts or 5.0 watts at 31 volts. The latter allowance was made to accommodate the design as it evolved.

13) An alignment prism was required to permit measurement of the tracker alignment to within .05 degrees.

14) A light shield intended to prevent light from the omni antenna from impinging on the lens of the tracker was required. The stated requirement was to exclude illumination from light sources at angles greater

than 24.66° from the optical axis in the roll direction.

To telemeter the CST information to the ground for star map interpretation and Canopus identification with a telemetry frame repetition rate of 23.9 seconds imposed a requirement that the CST map and roll error signal be subcommutated at 8 times the frame rate, or approximately one data point every 3 seconds. Roll rate during star map maneuvers was normally $1/2^{\circ}$ per second. This combination assured at least 5 data points while crossing the $\pm 4.1^{\circ}$ required roll field of view.

8.3.3 DEVELOPMENT AND OPERATION

At the time the design decisions were being made concerning the Lunar Orbiter Canopus star tracker the state-of-the-art was as follows:

a) No spacecraft had actually used Canopus as an attitude reference.

b) The Mariner IV Canopus tracker was under development at JPL. The Barnes Engineering Company was involved in manufacturing the design. Development problems included:

   1) Non-uniformity of the cathode of the CBS image dissector tube. Since discrimination of Canopus was based on brightness this affected the recognition gates.

   2) Manufacturing yield of the fiber optics used to adapt the star image in the focal plane of the catadioptric lens to the spherical photo cathode surface. Breakage or change occurred due to temperature or vibration exposure.

   3) Tolerances on the analog electronics photo tube and uncertainty about the spectral calibration of the test source as compared to Canopus (as seen in space) caused lack of confidence in the design.

c) The Surveyor Canopus tracker was under development by Santa Barbara Research Company. It used a motor-driven mechanism to scan the photo tube. The mechanical complexity and low predicted reliability ruled it out as a contender for the one year life mission desired for Lunar Orbiter.

d) The long range earth sensor that had been used successfully on Mariner II mission to Venus was developed by JPL and the Nortronics Division of Norair Inc. It used a vibrating reed to provide mechanical scanning of a photo tube. The sensitivity with a wider field of view required in this application was not adequate and extensive modification would have been required.

e) Narrow field of view mechanically gimbaled star trackers were available from various missile programs. Typically they were used to align inertial guidance platforms and were not space qualified.

f) A narrow field of view star tracker was under development for the Orbiting Astronomical Observatory program at the International Telephone and Telegraph Federal Laboratory. A wide yaw field of view was required for Lunar Orbiter to accommodate the angular excursion of the line of sight to Canopus relative to the normal to the ecliptic plane.

In the following discussions the development will be related to discrete time intervals between the major program events.

8.3.3.1 PROPOSAL SUBMITTAL TO PRELIMINARY DESIGN REVIEW

The Barnes/JPL tracker was initially proposed for use on Lunar Orbiter. Reports of development problems indicated that de-bugging the tracker was a major task for JPL.

It was desired to procure the star tracker from an expert subcontractor, to firm fixed price, and to have high confidence in the operation of the end product. A formal competitive vendor selection was made based on the requirements listed in 8.3.2.

The ITT Federal Laboratories were selected to develop a new tracker. Early constraints on the design were:

1) The ability to increase the optical field of view to 32 degrees in yaw and employ electronic gimballing within this field for search and track was required. This option was considered necessary to provide for tracker operation for the one year extended mission. When it was established firmly with NASA LRC that a Canopus referenced orientation was not necessary for the extended mission, this requirement was reduced to that required for 30 days of operation at the worst time of year.

2) Use of the CBS electrostatic deflection image dissector tube. Magnetic deflection coils were used in the other ITTFL designs. An early desire was to keep the spacecraft magnetically clean. It was thought that the magnetic deflection technique could cause problems for a potential magnetometer experiment. This experiment requirement went away. As a consequence, when ITTFL ran into delivery problems with the CBS tubes because of inability to meet rigid cathode uniformity tests, a switch was made to their own tube.

8.3.3.2 CRITICAL DESIGN REVIEW TO FIRST FLIGHT

Many development problems with the Canopus Star Tracker became evident during the comprehensive test program.

The formal portion of the test program included: (1) Flight Acceptance Tests, (2) a Qualification Test, (3) a Reliability Demonstration Test, and (4) an Incoming Acceptance Test. These tests and some of the significant results are presented in the following paragraphs.

Canopus Star Tracker Testing

The parameters of the CST which were checked before and after each Flight Acceptance Test (FAT) are given below:

a) Roll error versus Roll angle over the entire Roll/Yaw fields of view ($\pm$ 4.1 degrees and $\pm$ 8.20 degrees, respectively).

b) Star Map Telemetry output versus star intensity (expressed as times Canopus) over the entire Roll/Yaw FOV.

c) Canopus Recognition output (attitude control and telemetry) versus Star Intensity (× Canopus).

d) Roll Error Signal to Noise Ratio. The requirement was that this ratio must be greater than 24:1; all units had ratios greater than 30:1.

e) Roll Error Noise at Max Deflection. In all cases, the noise level was less than the 100 MV RMS requirement. Typically the noise was less than 70 mV RMS.

f) Sun Shutter Response. It was required to close within 0.5 seconds at a level of illumination equal to or greater than one × Moon (as seen by the CST at the 46 KM perilune). "Good" units typically had response times less than .3 seconds. The problems in setting the proper level of illumination to trigger the shutter, and in fabricating the shutter mechanisms are discussed below.

g) Roll Error Frequency Response. The requirement was not stringent; the time constant was to be less than 0.16 seconds.

The FAT Vibration tests were similar to those described for the IRU, Section 8.1. Problems with Sun Shutter operation and Roll Error null alignment resulted in a reduction in the required sine wave vibration levels. These reductions (to the same levels as for the IRU) were based on the results of a special test of a dynamic model of the spacecraft, which demonstrated that the actual vibration environment was less than originally estimated and specified. The random level requirements were not changed.

The CST was subjected to periodic solar illumination to check Sun Shutter operation during thermal vacuum tests and to continuous star illumination (at 1 × Canopus) so that Roll Error, Star Map and Canopus Recognition outputs could be monitored for intermittents.

The Qualification tests were similar to FAT except that the vibration represented the predicted 3σ mission values rather than the nominal values. No change was made in the random levels. Also electromagnetic interference (EMI) and humidity tests were added.

The Reliability Demonstration test, originally specified as two 30 day mission simulations performed on a CST which had first passed FAT, was reduced to one launch vibration simulation, and one 30 day thermal vacuum simulation with CST "events" equivalent to those in the original 60 days or thermal vacuum squeezed into the 30 days.

The performance tests run before and after the vibration,and before and after thermal vacuum were the same as for FAT. The deviations in Roll Error "Gain" (Roll Error vs Roll Angle), Roll Error Null (output at zero input), and Star Map Telemetry versus time were monitored.

Incoming Acceptance tests were run at Boeing to check the integrity of each CST after FAT at ITT and before installation on the spacecraft. In these tests, the CST was illuminated using an ITT-furnished Star Tracker Test Set (STTS). The tracker with test set attached is shown in Figure 8-20. The parameters checked in IAT were the same as the basic parameters checked in FAT.

Component Test Problem Areas

High Voltage Arcing - Although the first tracker passed its FAT test with adequate margin, its electronics were inadvertently destroyed by arcing in the high voltage power supply in a subsequent test when it was turned on, by error, at a pressure 10^{-5} mm. of mercury as the thermal vacuum chamber was being evacuated. This danger had been recognized and turn-on in the L.O. mission was delayed for six hours after launch. Operator error allowed the warnings in the test procedure to be missed.

Two other trackers were subsequently damaged by arcing even when procedures to delay turn on until a pressure of 10^{-5} mm. of Hg was reached. Slow bleed off of gases from the potting compound around the tube caused the problem. The gas was trapped in pockets formed by a mylar film which was wrapped around the trackers for reflective insulation. Final solution to the problem was reached when the CST was vacuum cycled after potting, the reflective film was loosely wrapped, the tracker cover vents were increased and the soak time at 10^{-5} mm. of Hg in the thermal vacuum chamber was increased to three hours before turn on.

Sun Shutter Sticking - The operation of the sun shutter was found to be erratic following FAT vibration of the second tracker. It would sometimes fail to open after illumination of the Bright Object Sensor (BOS) had closed it to protect the image dissector tube. Insufficient clearance between the shutter and its housing, inadequate stiffness of the solenoid shaft and of the shutter blade, and excessive amplification of tracker vibration inputs at the shutter location were the causes of this problem. These were not completely identified and solutions implemented until the Qualification Test tracker was modified. Solutions included structural stiffening of the shutter blade and of the entire tracker frame with epoxied stiffeners, addition of a butyl rubber damping pad between the tracker structure and spacecraft structure, and a change in the damping coefficient of the silicone rubber used to pot the image dissector tube to the magnetic shield which, in turn, was potted to the tracker structure.

Tracker Alignment Shifts - Along with the sun shutter sticking problem, it was determined that the vibration (and thermal cycling) of the tracker was causing the Roll Error null plane to shift relative to the front face and

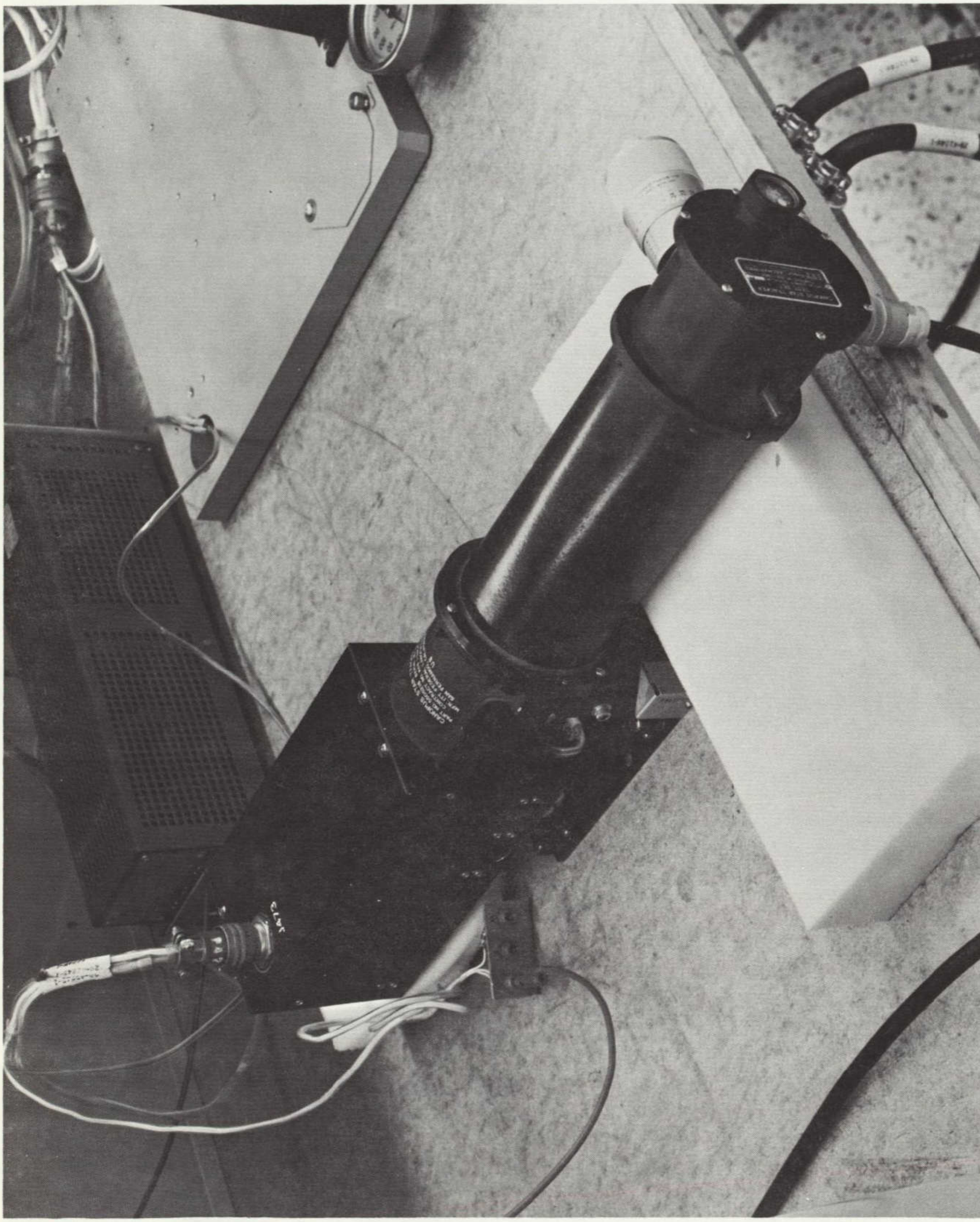


Figure 8-20: CANOPUS STAR TRACKER WITH TEST SET ATTACHED

reference prism. The problem was identified as motion between the image dissector tube and the support structure caused by excessive vibration strain and thermal distortion. Magnetic fields induced in the tracker structure by the large DC field surrounding the vibration shaker also caused null shifts.

Solution to this and the sun shutter sticking problem was realized by the same actions which included reduction of vibration amplification of the tracker structure through use of stiffeners and dampers as described above, and reduction of thermal distortion and image dissector tube vibration amplification relative to the structure by changing the potting compound from RTV-106 to RTV-11. Vibration levels at the tube were reduced from over 100 g's (in response to a 15 g tracker input) to less than 30 g's. It is of interest, and a tribute to the integrity of the ITT tube and electronics, that there were no failures of these elements during the diagnostic vibration tests where the extreme amplification levels were encountered. Thermal distortions of the Roll Error Null were reduced from over 0.3 degrees to less than 0.1 degree by changing the potting compound and by temperature cyling it before final alignment. Magnetically induced shifts were minimized by degaussing the tracker periodically and replacing all "hard" magnetic materials in the tracker structure with "soft" or non-magnetic materials.

Star Map Gain - In spite of increasing "burn-in" time to 100 hours of exposure for each image dissector tube to illumination of intensity 5 times Canopus, the tubes continued to show degradation of video output with time when exposed to illumination as low as 1 × Canopus. This degradation was as much as 2.5:1 in the RDT unit during both 15-day cycles even though it was retrimmed after the first cycle. The degraded tubes also showed a recovery, or healing, tendency when not illuminated and when turned off. The mechanism was and is not understood and no solution was or has been effected other than frequent recalibration. The flight experiences with this phenomenon are discussed later in the section on flight data.

Simulated Star Source (SSS) Calibration - The process of calibrating the SSS to represent Canopus as seen by the S-11 image dissector tube consisted of:

a) Measuring the intensity and spectrum of the GE quartzline lamp as seen through the color filter.

b) Analytically applying a correction factor for the optics of the SSS and tracker and for the S-11 photocathode in order to calculate the necessary size of the aperture.

c) Analytically applying a very small correction factor to rating of the neutral density filters to account for the measured deviations from the 20 nominal logrithmic steps from 5 × Canopus to 0.1 × Canopus.

As the result of comparisons between the output of a tracker in response to illumination by the two ITT-produced SSS's (one at ITT and one at Boeing) and to the JPL SSS it was determined that there were errors in the calibration calculations for the SSS's at ITT and Boeing which resulted in them producing an output 30 percent too high. These errors were discovered and the flight trackers recalibrated before the second L.O. flight. The fact that the recalibrated results were right and the correlation of ground test and flight test Star Map telemetry outputs are discussed in the section of the L.O. mission results.

Star Tracker Test Set (STTS) Calibration - Although the original design requirements for the STTS called for interchangeability between all trackers and STTS's, it was discovered that the spectral responses were not well enough matched nor stable enough to serve as a secondary standard for checking Canopus intensity during various tests. In an attempt to circumvent some of these difficulties the approach taken was to "marry" each STTS to a tracker by calibrating each of the five star sources of the STTS against the SSS at ITT, using the particular tracker as a transfer standard. The STTS was then delivered to Boeing with the tracker and Boeing used them as a matched set throughout IAT and spacecraft testing. It was found that the intensity of the STTS illumination varied with time in an unpredictable manner and although the STTS proved to be a reliable stimulus for Roll Angle, it could not be used as a reliable stimulus for star intensity even though the intensity control was calibrated against the SSS filters. In the end, the STTS was used in a "go-no-go" test of star map output (which was what it was originally designed to do), and the flight trackers were returned to ITT for star map calibration just before the spacecraft were placed on the booster.

8.3.3.3 MISSION PERFORMANCE

Mission I

During the initial phases of Mission I the star tracker failed to operate as expected because of excessive light flux into the optics. This "glint" problem occurred while attempting to produce a star map that could be used to determine a precise attitude baseline for a midcourse maneuver. The initial star map is shown in Figure 8-21. Also shown in the same figure is the computer predicted a priori map. After numerous commanded maneuvers in the attempt to determine the source for the apparent interference, a decision was made to use the Moon as a celestial reference. The high gain antenna pattern "map" was used to confirm the roll orientation. The outcome of the 37.8 meter per second midcourse was successful which placed the spacecraft on such a precise trajectory that a second midcourse was not needed to achieve the injection point for transfer to the initial ellipse three days later. Subsequent operation of the star tracker in the shadow of the moon was considered normal except for the low map signal voltage sensitivity. Earlier test and development work with the tracker had indicated that prolonged exposure of the tracker to bright light with the tracker turned on caused degradation of star map gain. Since the tracker was on continuously for 60 hours during which time it was exposed for 2 3/4 hours to the moon in addition to locking on stray light the remaining time, it was felt that this was probably responsible for the low sensitivity.

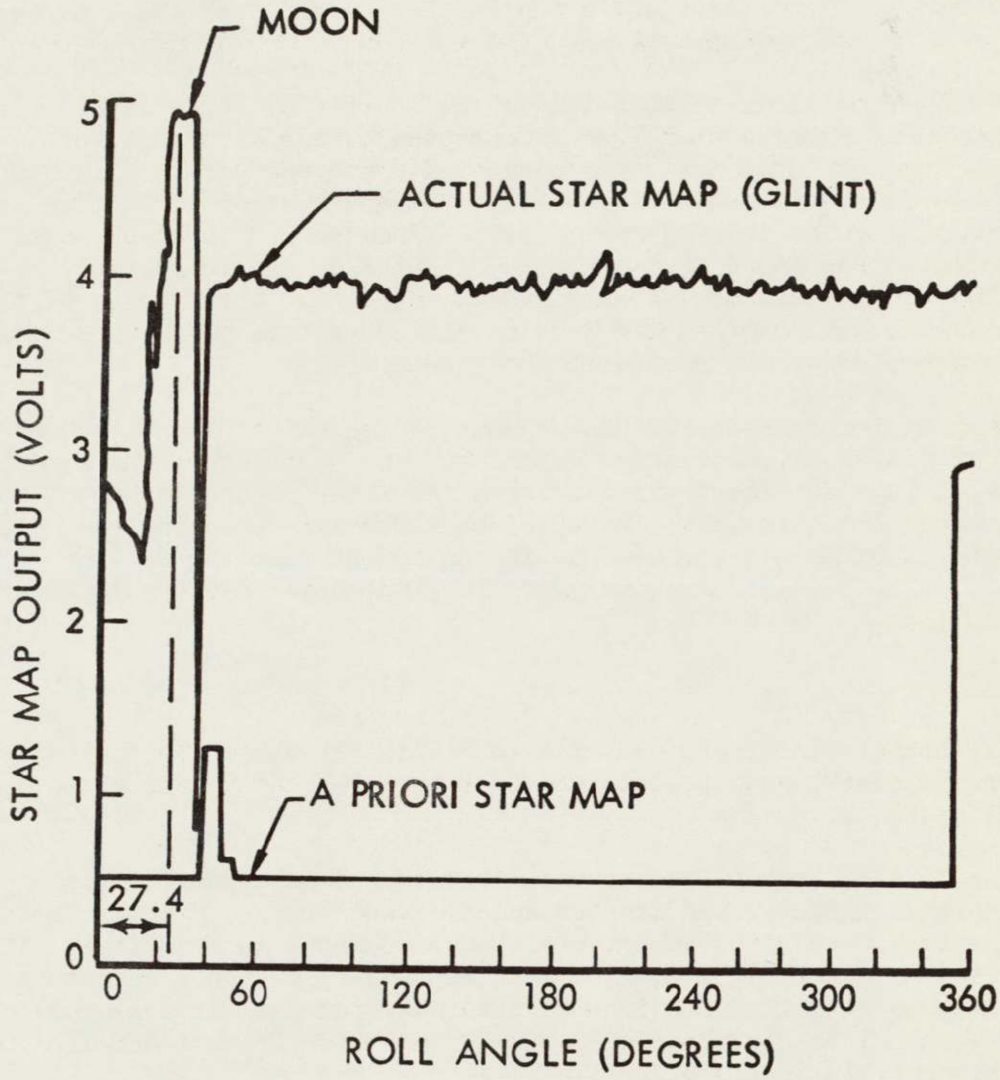


FIGURE 8-21 CANOPUS STAR TRACKER STAR MAP
L. O. I GLINT PROBLEM

A Space Glint Test (see Figure 8-22) showed clearly that the omni antenna was a major contributor to the stray light which impinged upon the Canopus tracker. An interesting facet of the glint problem was that the tracker invariably locked onto light reflected from the baffles on one or the other side, resulting in either plus or minus roll saturation. Negative roll saturation was generally associated with a higher map signal output, probably because the antenna reflected directly onto the baffles in the negative side of the tracker while the baffles on the positive side were lighted by indirect reflections.

A number of ground tests to investigate the sources of stray light interference with the Canopus Star Tracker observed during Mission I were initiated. One of these tests was conducted using Spacecraft No. 1, shown in Figure 8-23, showed that the Omni antenna gave a stray light flux sufficiently high for the tracker to switch from search to track mode. In addition, it was found that the stray light flux due to multiple reflections from the equipment mounting deck was high. The purpose of the tests was to obtain quantitative data on the corrective measures proposed to be incorporated on the second flight spacecraft.

As a result of these tests, it was recommended to paint black the Omni antenna, the edges of solar panel #2 and #4, and portions of the backside of the solar panels. These fixes lowered the interference to 23% of the danger level. The tests also indicated that further reduction could be achieved by light shields mounted to the Equipment Mounting Deck or the top of the solar panels. A summary of the glint data is given in Figure 8-24.

Mission II

Spacecraft changes incorporated prior to Mission II were: 1) Omni antenna painted gloss black, and 2) edges and backside of solar panels 2 and 4 painted flat black.

On Mission II the Canopus Tracker was first turned on approximately 6 hours into translunar flight. The tracker acquired and tracked Canopus immediately, indicating the spacecraft was oriented toward Canopus at that time. This was verified by making a 360 degree star map and by a high gain antenna signal strength map. The roll error from Canopus was then used in a closed loop mode until the midcourse maneuver. The Canopus Star Tracker appeared to be working properly in full sunlight.

The first star map matched the a-priori map fairly well except for the appearance of a broad (approximately 50 degree) object which had a peak map signal strength of about 4 volts. At this time, the earth had a clock angle (defined as the angle about the sun line from Canopus) of 130 degrees and a cone angle (defined as the angle away from the sun line) of 140 degrees. It was determined that the earth, which was 30 degrees out of the field of view in cone angle, was the source of the stray light. The apparent lag in cone angle of the peak signal strength was consistent with the location of the low gain antenna.

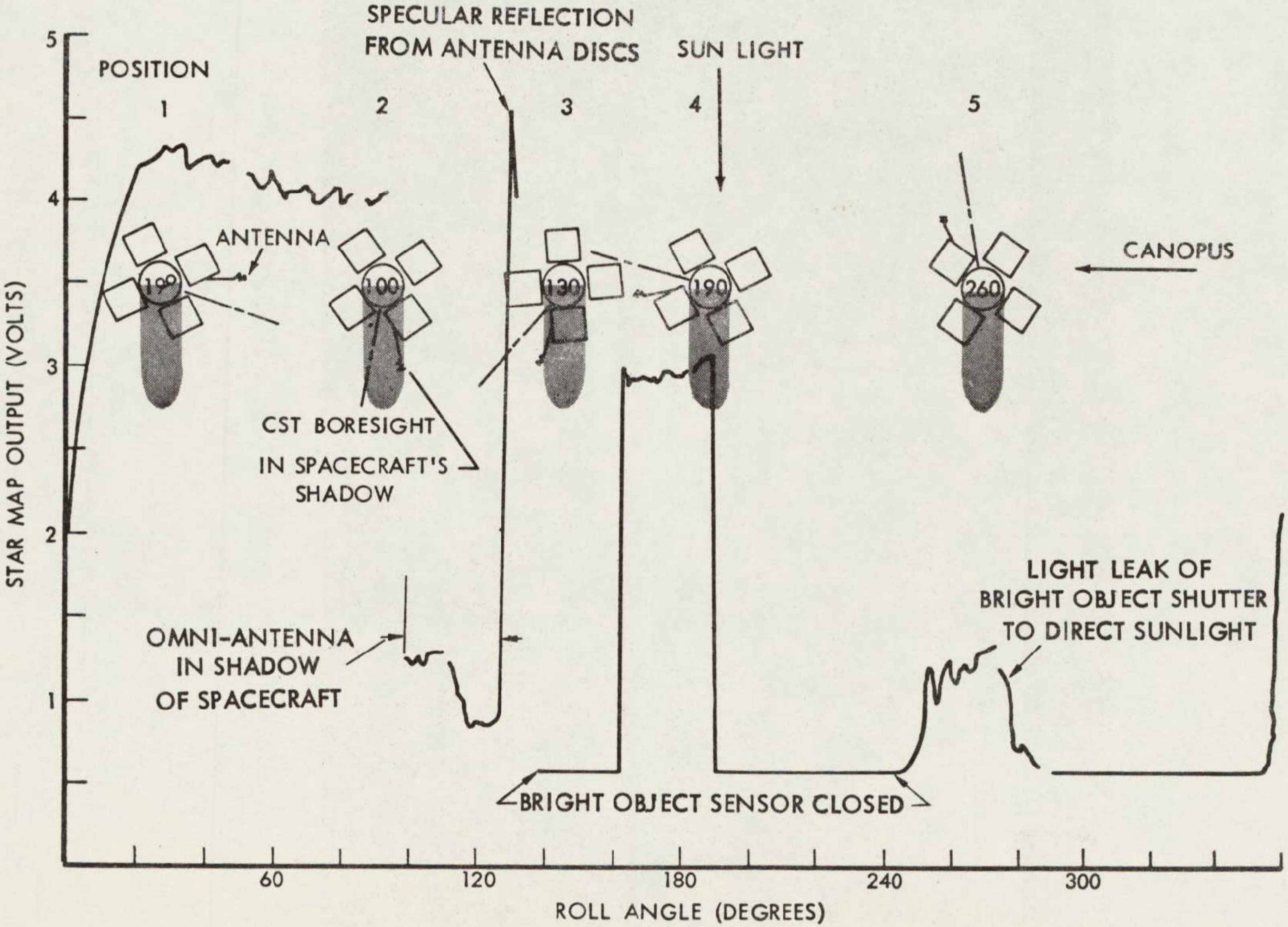


FIGURE 8-22 CANOPUS STAR TRACKER L. O. I SPACE GLINT TEST

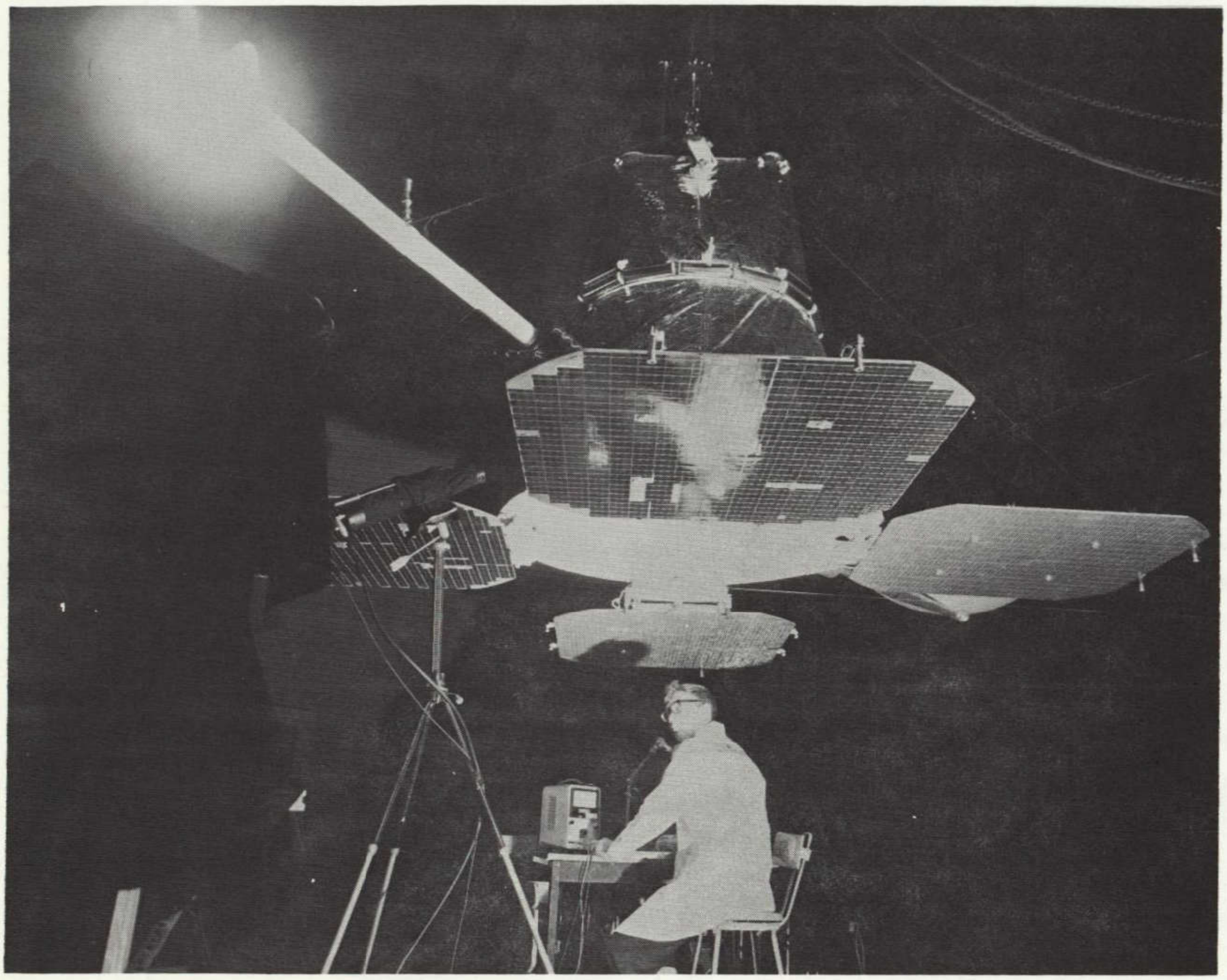


Figure 8-23: GROUND GLINT TEST IN PROGRESS

SPACECRAFT #1 - NO MODIFICATIONS INTERFERENCE LEVEL	INITIAL*
1. OMNI ANTENNA & BOOM, PAINTED WHITE	1.09
2. SOLAR PANEL #2, EDGE NOT PAINTED	0.068
3. SOLAR PANEL #4, EDGE NOT PAINTED	0.021
4. HI GAIN ANTENNA (MULTIPLE REFLECTIONS)	0.035
5. E M D (MULTIPLE REFLECTION) SOLAR PANEL TOP FLAT BLACK, EDGE NOT PAINTED	0.40
TOTAL	1.614

SPACECRAFT #1 MODIFIED TO REDUCE INTERFERENCE	FINAL*
1. OMNI ANTENNA BOOM, GLOSS BLACK, DISCS FLAT BLACK	0.027
2. SOLAR PANELS #2 & #4, EDGES TAPED BLACK, BELOW NOISE LEVEL	---
3. HI GAIN ANTENNA, BELOW NOISE LEVEL	---
4. E M D (MULTIPLE REFLECTIONS SOLAR PANEL, TOP FLAT BLACK, EDGE TAPED BLACK)	0.21
TOTAL	0.237

* NUMBER GIVEN AS RATIO OF LIGHT FLUX TO CAUSE A STAR TRACKER TO SWITCH FROM SEARCH TO TRACK

RESULTS OF GROUND GLINT TESTS

FIGURE 8-24

Just prior to midcourse, the propellant squib valves were fired. Immediately thereafter, Canopus track was lost and the attitude control system went into inertial hold in roll. This resulted in a delay of the midcourse maneuver while a second star map was made.

The second star map, shown in Figure 8-25 matched the a-priori map fairly well but minor glint from the earth was again evident. In this case, however, the maximum signal due to earth glint was down to 1.1 volts due to the greater earth range. After the second map, the Canopus Tracker continued to track Canopus but it was not used in a closed loop mode to control attitude. Instead the roll axis was controlled by the gyro and maneuver commands were transmitted to the vehicle as required to maintain Canopus in the field of view and to zero the roll error in preparation for Midcourse, injection, and transfer maneuvers.

The injection maneuver occurred during sunset. Immediately after sunrise, Canopus was lost due to reflections of moon light. A subsequent experiment established that Canopus could not be tracked reliably in lunar orbit and that continued exposure to moon light would degrade the Canopus Tracker map signal voltage. As a result, it was concluded that the Tracker should be operated only during sunset.

After transfer to the final orbit, during photography and readout, the method of roll control adopted was:

(a) Turn Tracker on after sunset.

(b) Acquire Canopus in closed loop mode to update roll attitude.

(c) Turn Tracker off before sunrise.

The method worked well with reasonable nitrogen consumption rates and minimal operational problems.

Mission III

Results of the first star map on Mission III were uncertain because of data loss during the maneuver, however, a second 360 degree roll, shown in Figure 8-26 was successful in establishing a roll reference. The spacecraft was rolled +125 degrees to Canopus and a tracker off-on cycle was performed to observe and track the star.

Canopus was tracked throughout the translunar flight but without the error signal being used by the attitude control in a closed loop mode. The tracker lost Canopus six times during this period, once when the squibs were fired, four times with no apparent spacecraft disturbance, and once just prior to injection when the moon albedo caused a pronounced glint problem. Each time track was regained by commanding the tracker off-on cycle.

Star map signal was initially 3.7 volts, decayed to 2.6 volts through translunar and recovered to 3.25 volts by the end of the mission. Following injection the tracker was operated only in the dark.

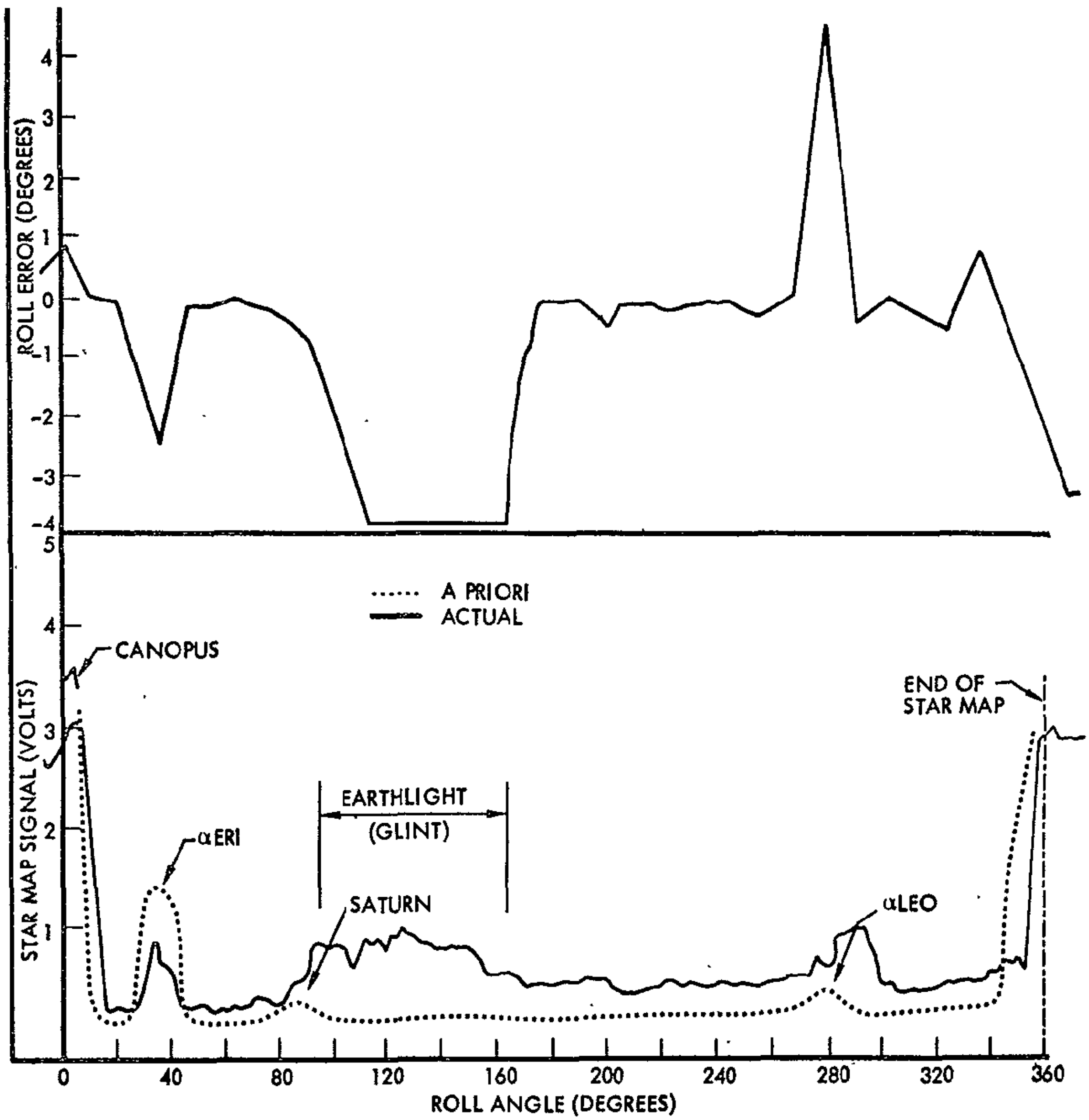


FIGURE 8-25 STAR MAP – L. O. II

TELEMETRY DATA STAR MAP TABULATION

T/M NO.	SIDL NO., NAME	CLOCK ANGLE APRIORI	CLOCK ANGLE OBSERVED	CANOPUS FLUX RATIO	MAP VOLTAGE APRIORI	MAP VOLTAGE ACTUAL
1	13, BE CRU, MIMOSA	311	310	0.16	0.7	1.22
2	8, AL CRU, ACRUX	316	314	0.38	1.4	1.30
3	74, TH CAR	326	326	0.04	*	1.02
4	61, IO CAR	335	337	0.06	*	1.06
4	28, BE CAR, MIAPLACIDUS	335	337	0.09	*	1.06
5	2, AL CAR, CANOPUS	0	REFERENCE	1.0	3.9	3.5
6	601, EARTH	96	40-120	3.1×10^7	*	1.38

* NO DATA

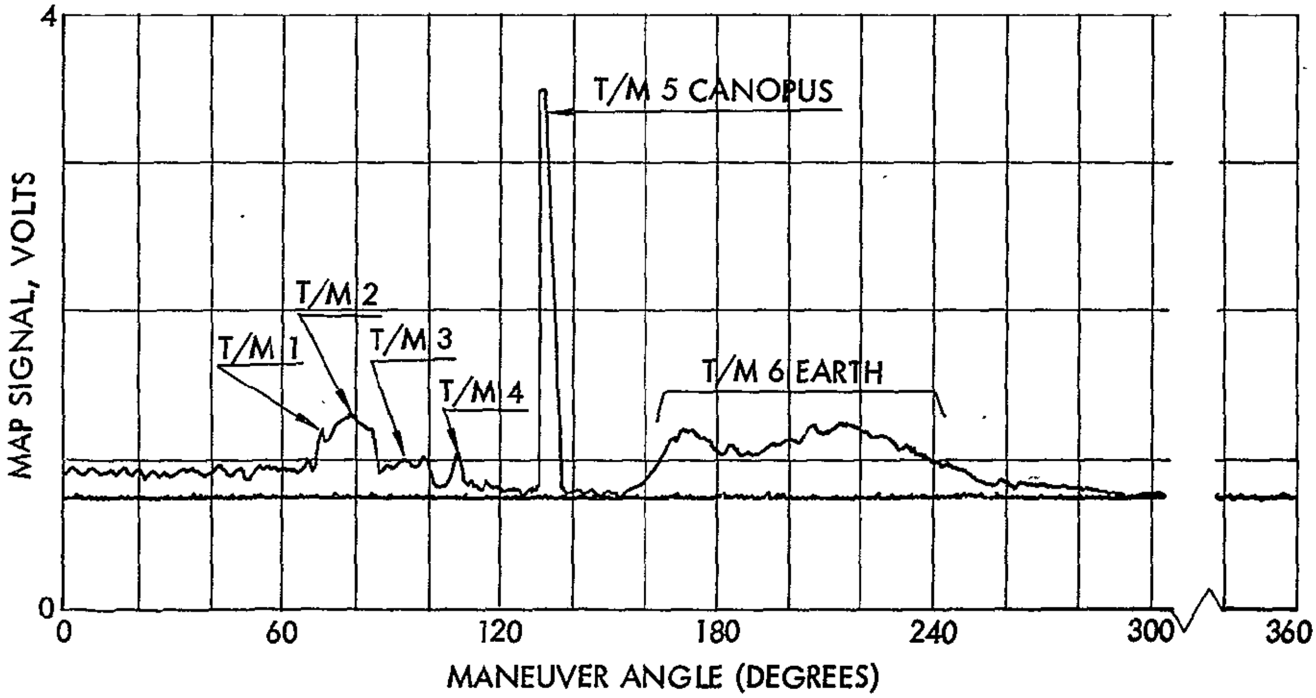


FIGURE 8-26 MISSION III TELEMETRY DATA STAR MAP

During sunset of the fourth orbit, Canopus was acquired in the "closed loop" mode. For the remainder of the photo mission the operational procedure was to acquire Canopus during sunset and operate the roll axis in the closed loop mode. Star maps were performed three more times during Mission III. Results of these maps agreed well with the a priori maps.

Mission IV

The Canopus Tracker was first turned on approximately seven hours into translunar flight. It was locked on positive glint at that time and remained there throughout the first star map maneuver. The star map voltage was consistently high, and averaged about 3 volts. Two hours later the tracker was turned on in a negative glint condition and a second +360 degree star map maneuver was performed with similar results. Two blips in the star map voltage were noted at 75 degrees indicating that the tracker had viewed a celestial object. The spacecraft was rolled +84 degrees to view this object and a tracker off-on cycle was performed and the star was tracked successfully. The third star map and the high gain antenna signal strength map verified that the tracker was tracking Canopus.

Canopus was tracked without acquisition for one hour prior to injection. For the remainder of the photo mission the operational procedure was to turn the tracker on without acquisition once or twice an orbit to obtain a roll reference and update roll attitude prior to a photo sequence. On those occasions when the tracker was locked on glint, track was usually regained by performing an off-on cycle.

The orbit for Mission IV was nearly Polar with a period of approximately 12 hours. As a consequence the spacecraft operated in full sunlight for long periods and large illuminated areas of the moon regularly came within view of the tracker.

The Bright Object Sensor was closed for approximately 59 hours during the latter part of the mission.

The star map signal, was initially 2.5 volts, decayed to 2.1 volts and recovered to approximately 2.4 volts by the end of the mission.

As a consequence of the Canopus Star Tracker anomalies observed during Mission IV additional tests were conducted (similar to those conducted after Mission I) to define possible glint sources on the spacecraft. These tests indicated that the power resistor dissipation panel on the omni antenna was a bright point source of glint in the Star Tracker baffle field of view. In addition, the stray light flux reflected from the equipment mounting deck to the backside of number 4 solar panel (painted flat black) was high.

It was therefore recommended that the resistor panel be shielded, that the number 4 solar panel be painted gloss black, that the solar panel stowing lugs be painted flat black, and that all the unpainted screws, etc., on the resistor panel be painted black.

Diagnostic analyses of the star tracker internal design were also initiated at Boeing because it was evident from flight experience that the changes to the spacecraft were ineffective in solving the problem. Design data were obtained for the lens and sunshade baffles. Lab tests were conducted of the electronic scanning. The analyses showed that the baffle edges were effectively inside the imaged field of view and that the electronically scanned field was $\pm$ 11.2°. This information was not available soon enough to effect any changes in the star tracker for spacecraft V.

Mission V

The Canopus tracker was first turned on approximately seven hours into translunar flight. Initially it was in the track mode indicating a roll error of minus 3.2 degree and a map signal of 1.12 to 1.18 volts. This star was later identified as Acrux (a Crucis). The star was tracked for approximately two minutes at which time the roll error went to positive saturation due to "glint". From this point on the story was essentially a repeat of Mission IV, but with different details.

The next unexpected event occurred approximately 9 hours prior to the end of the primary photo mission of L.O. V. The deadzone was opened and drift tests in pitch, roll, and yaw were started. On three occasions thereafter, the tracker was turned on and the bright object shutter was closed. This data clearly showed that the tracker was extremely sensitive to glint as a function of yaw attitude. Each time the spacecraft was yawed toward the Sun .75 degrees the bright object shutter closed. This corresponds to the tracker centerline moving from 84 degrees to the sunline to an angle of 83.25 degrees.

The star map signal was initially 2.4 volts, decayed to 2.1 volts and "healed" to approximately 2.9 volts by the end of the mission.

The Bright Object Sensor and Canopus Presence Signal Gates worked as expected on all missions. A summary of "on" time in space and number of ON-OFF cycles for each tracker is given in Figure 8-27. A summary of "glint" problems is presented in Figure 8-28.

8.3.4 CONCLUSIONS AND RECOMMENDATIONS

Proper operation of an orbiter spacecraft with a wide field of view optics on a star tracker required considerably more attention to details of the design and the configuration than was expended on the Lunar Orbiter Canopus tracker. The following points are important and need be considered more seriously:

1) Broad sources of diffuse light, such as a nearby planet, are potential glint problems that must be analyzed carefully.

2) Location of spacecraft elements which may be illuminated and be within the field of view of the baffle, required careful analysis.

3) Sun shade design must be such that edges do not appear in the imaged field of view.

MISSION	ON-OFF CYCLES	TOTAL HOURS ON TIME IN SPACE	PRELAUNCH MECHANICAL SETTING *
I	133	68.0	83.0^{o}
II	252	162.5	98.2^{o}
III	299	118.3	98.2^{o}
IV	145	39.8	81.5^{o}
V	229	80.75	84.0^{o}

* SUNLINE TO TRACKER CENTERLINE

FIGURE 8-27 TRACKER ON-OFF HISTORY

LO I, IV, & V - BORESIGHT TOWARD SUN

- o STAR MAP VOLTAGE INITIALLY HIGH - DEGRADED WHEN CST WAS ON - RECOVERED IN TIME
- o TURN CST OFF/ON WHILE LOOKING AT CANOPUS TO ACQUIRE
- o USED HIGH GAIN ANTENNA AS ATTITUDE CHECK AND BACKUP
- o USED MOON FOR LUNAR ORBIT INJECTION ROLL REFERENCE ON L.O. I
- o TURNED CST OFF DURING SUNLIT ORBIT PERIODS, GYRO IN RATE INTEGRATE MODE
- o UPDATED GYRO ROLL ATTITUDE WITH CANOPUS TRACK DURING SUN OCCULTATION
- o PERFORMED EXPERIMENT TO SHADE OMNI ANTENNA, AND IDENTIFY GLINT SOURCE ON L.O. I
- o SUNLIGHT CLOSED BRIGHT OBJECT SENSOR ON L.O. V AT SMALL ANGLE TOWARD SUN

LO II & III - BORESIGHT AWAY FROM SUN

- o GLINT REDUCED FROM L.O. I
- o INITIAL STAR MAP WITHIN 10% OF GROUND TEST
- o SUNLIGHT DEGRADATION TESTED BY TURNING CST ON ONLY DURING SUN OCCULTATION
- o ROLL ERROR CHECKED WITHIN TELEMETRY ERROR FOR 360° MANEUVERS

FIGURE 8-28

SUMMARY OF STAR TRACKER GLINT & IMAGE DISSECTOR TUBE DEGRADATION

4) A criteria to minimize stray light effects would be to prevent all second bounce light from entering the tracker lens. The Lunar Orbiter baffle design excluded only first bounce light at best.

5) Electronic sweep of the field of view should be limited to prevent "seeing" baffles or other mechanical edges.

6) Mechanical adjustment of tracker boresight requires careful scrutiny to assure that the sunlight does not enter the sun shade or that illuminated surfaces are not brought into the optical field of view.

7) The Star Tracker Test Set was found to be the cause of many of the problems associated with testing of the Canopus Star Tracker. The angular error measurements were accurate, but the intensity levels for calibration of the star map signal could only be used for "go" or "no-go" testing.

8) Star Tracker image dissector tubes are subject to output degradation (and healing) in the long term presence (or absence) of illumination at star intensities. Physical phenomenon and solution are not clearly understood.

Recommendations for "doing differently if one had it to do over" are to develop the technology which would allow a better analytical and hardware evaluation of the following:

1) Star simulation for calibration of the sensor.

2) "Bright object" simulation of nearby planets.

3) Spacecraft level glint simulation.

4) Dark room glint suppression.

5) Star Tracker component evaluation under realistic operating conditions.

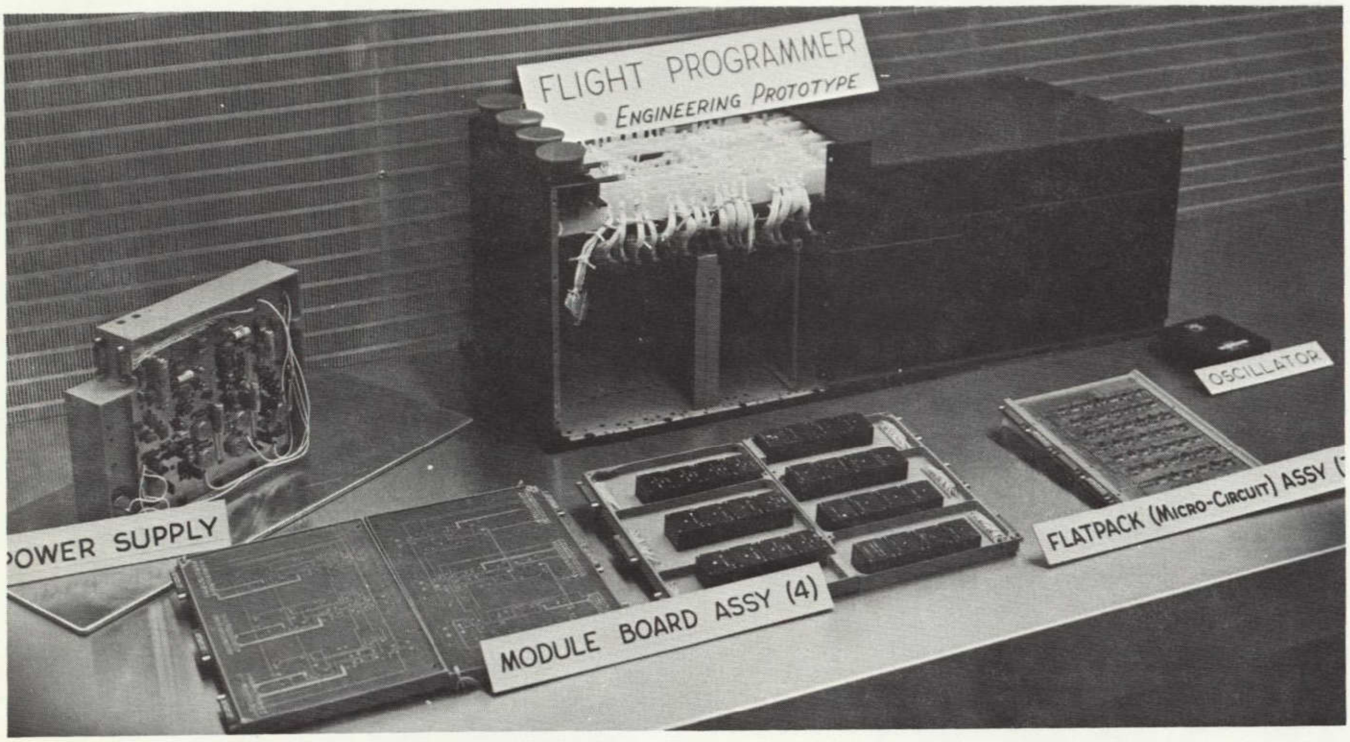


Figure 8-29: FLIGHT ELECTRONICS CONTROL ASSEMBLY

8.4 FLIGHT ELECTRONICS CONTROL ASSEMBLY

The Flight Electronics Control Assembly, shown in Figure 8-29, interfaced with each spacecraft system to control all timing and sequencing of spacecraft events. All elements of the G&C system were interlinked by this unit and functionally it was a part of each of the four G&C subsystems discussed in Sections 3.0 through 6.0. The weight of the assembly was 18.13 lbs. The average power used in sun (at 32 volts) was 42 watts; the power used at night (21 volts) was 32 watts. This assembly consisted of two distinct sections, the Programmer and the Closed Loop Electronics (CLE). The Programmer was a low-speed digital data processor, with memory capacity large enough to provide 16 hours of control from stored commands and a 29.1-hour clock. The Closed Loop Electronics, which was basically all analog in operation, performed the task of attitude control. It accepted error signals from the gyros and sensors to control the reaction thrusters and thrust vector control actuator in response to mode commands from the Programmer. These two sections, Programmer and CLE, will be discussed individually in the following paragraphs.

8.4.1 PROGRAMMER DESCRIPTION

The Programmer portion of the Flight Electronics Control Assembly was a serial data processor, operating at a bit rate of 2.4 KHz. The programmer contained a 21 bit, 128 word memory. The unit could execute either real time or stored program commands. Commands transmitted from the ground could provide for program updating, revising, or initiating the program routine as the mission progressed. The programmer is shown in the functional block diagram of Figure 8-30.

The programmer has a hard line input function which controlled two modes of the system for ground checkout. One mode prohibited execution of the stored program, but allowed memory loading and verification. Discrete real-time-commands could override this control. The second mode allowed normal execution of stored commands, real-time-commands, memory loading and 11 spacecraft functional interrupts.

The Programmer is described by discussing the functions of the individual blocks shown in Figure 8-30.

Signal Conditioner - provide input and output circuit isolation, proper external voltage levels, noise suppression, and minimized susceptibility to radio frequency interference.

Command Register - This 20 bit register held stored program commands from memory or real-time-commands from the command decoder and provided complemented and uncomplemented, parallel outputs to the output matrix decoder. The register also held time, velocity or attitude constants for comparison with integrated values during event timing and maneuver sequences.

Mode Control Register - The mode control register accepted command information and nine signal interrupts from spacecraft telecommunications, two occultation signals from the attitude reference subsystem, and one from the photographic subsystem. It set up the flight programmer for:

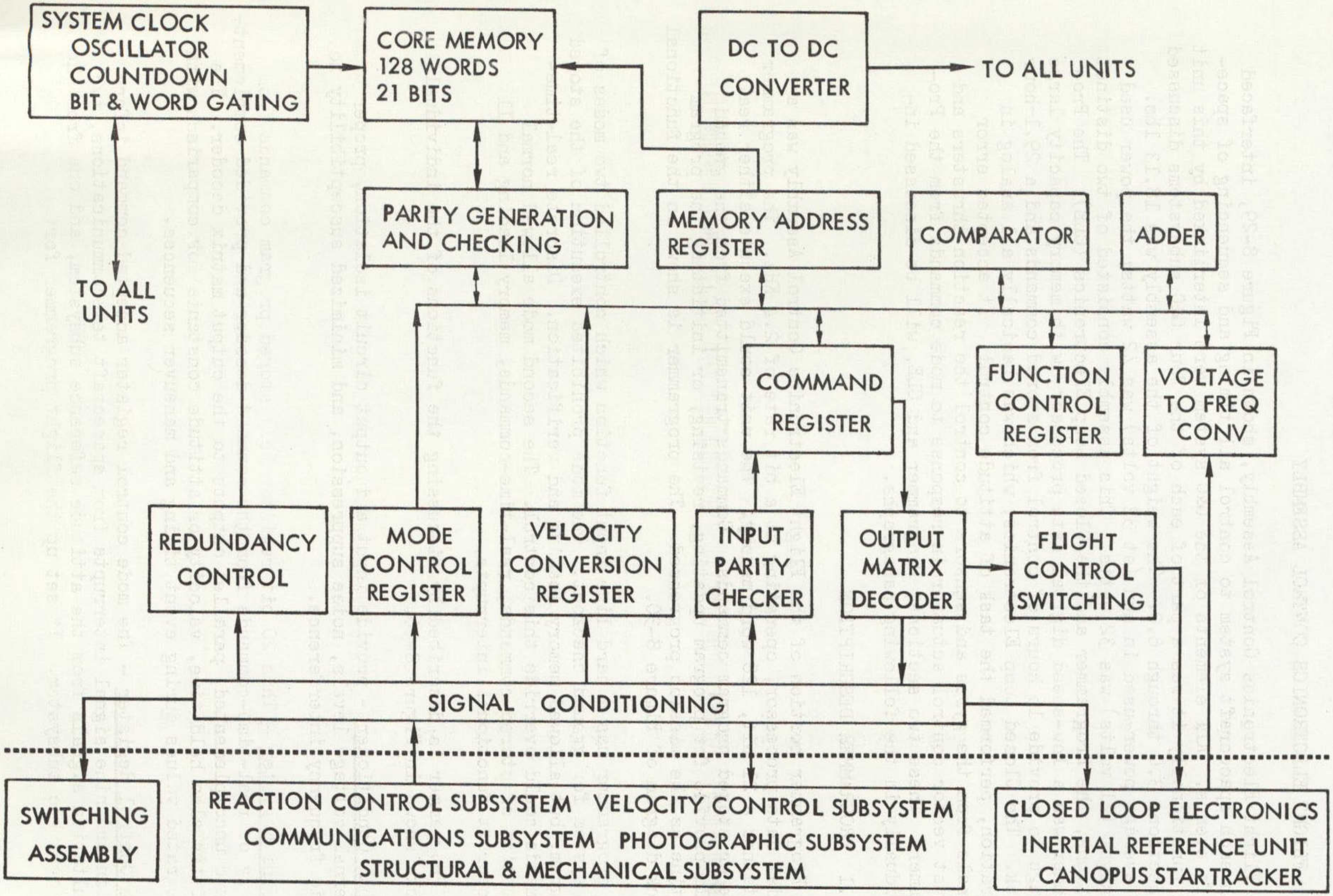


FIGURE 8-30 PROGRAMMER BLOCK DIAGRAM

1) Real time or stored-program operation;
2) Interrupt commands (stop stored program and execute new command word);
3) Information requests (telemetry and spacecraft time).

Input Parity Checker - The input parity checker checked the command words transmitted from the ground through the communications command decoder for even parity.

Function Control Register - The function control register held the present five-bit operation code and functional mode of the programmer and combined them with timing to direct the information flow. The function control register also controlled changes in the programmer operating mode.

Adder - This unit was a half-adder and had logic circuitry which added one bit at a time to a value stored in the memory. The adder was used to add time, position and velocity increments, and to increment addresses.

Velocity Conversion Register - The velocity conversion register detected the difference between two input frequencies. The sum of the two frequencies was constant; the difference rate was variable and proportional to the acceleration measured by the accelerometer. The difference frequency was scaled into 0.1 foot-per-second increments for midcourse and orbit insertion velocity changes.

Memory Address Register - Seven address bits, held in this register, controlled the addressing of words read in or out of the memory. Addressing of special-purpose memory words, as commanded by the function control register, overrided, but did not destroy, the contents of the memory address register flip-flops. This technique allowed all indexing and address-modification-registers to be contained in the memory instead of external flip-flop-registers.

Memory - The memory consisted of a magnetic-core plane with associated discrete part read-and-write circuitry. The memory contained 128-word storage which could be randomly selected. Read in/readout of the 21-bit word was serial, controlled by an internal counter. The basic bit cycle was read/modify/write. The memory accepted data from the command register and adder. Eight specific word locations were employed as spacecraft time-accumulation-register magnitude-integrating-register, next instruction-address-register, store program word-address-register, Sun and Canopus occultation time recording registers, interrupted instruction and address-register, and telemetry word-transfer. The use of these special word locations reduced the number of active circuit registers. This resulted in a reduction of power consumption and weight.

Voltage-to-Frequency Converter - The voltage-to-frequency converter accepted an analog rate voltage from the gyros in the inertial reference unit, which it converted to equivalent digital pulses. The digital pulses were integrated during a maneuver and compared to a constant value corresponding to the desired maneuver. Equality terminated the maneuver command.

Parity Generation and Checking - This unit provided buffering between the memory and the programmer logic. The parity generator provided an even parity bit in bit-position 21 for words transferred into memory. An even parity check was performed on the words transferred out of memory.

Comparator - The comparator provided serial comparisons between a constant value from the command register and an incremented value from memory. Twenty-bit time words, and 15-bit attitude and velocity magnitude words were compared. When comparison was reached, the comparator provided a logic signal which caused the programmer to proceed to the next instruction.

System Clock - There were two crystal oscillators in the programmer (for redundancy). Each oscillator drove a countdown register from which all programmer frequencies and timing were derived. The crystal oscillator was assembled with discrete parts and provided a basic 614.4 KHz output frequency. The oscillator provided an extremely stable, jitter-free clock frequency with an accuracy of 1 part in 10^6. The countdown register generated 0 to plus 4.5 v.d.c. square-wave trains of the following frequencies: 614.4 KHz; 307.2 KHz; 153.6 KHz; 76.8 KHz; 38.4 KHz; 19.2 KHz; 4.8 KHz; 800 Hz; 100 Hz; 50 Hz; and 10 Hz.

Redundancy Control - The redundancy control provided switching, in event of failure, between the two countdown registers. Control was exercised by a real-time command from the command decoder.

Output Matrix Decoder - Decoding matrices and necessary logic elements provide unique commands and signal formats to all required spacecraft subsystems. Pulse commands, on-off commands, timing bursts, and gated sequences were formed in this unit and were sent to the signal conditioning unit for output.

High level signals and their drivers were electrically and physically isolated from low-level signals and devices. High-level signal circuits were enclosed in the Switching Assembly which is discussed briefly in paragraph 8.4.3.2.

Flight Control Switching - Commands from the output matrix decoder, and Sun and Canopus presence signals were combined to provide mode commands for the attitude control subsystem in roll, pitch, and yaw axes, as well as turning the IRU on or off, switching the gyros between rate or rate integrate modes, and turning the accelerometer "on" or "off" (actually this latter only closed or opened the rebalance loop). Accelerometer excitation and temperature control were always "On" whenever IRU power was on.

DC to DC Converter - Converted spacecraft voltage (+ 32 to + 21 vdc) into plus 4.5 volts, plus 6 volts, plus 15 volts, and minus 15 volts for the integrated logic circuits, memory and signal conditioning. The converter provided voltage to the switching assembly and + 15 and - 15 volt signal reference voltage to the thrust vector actuator.

Command-Word Format - The ground-transmitted command word consisted of 22 information-bits. The order of transmitting the information word began with the least significant bit. The ground-transmitted word format was received by the command decoder, verified and transferred to the programmer on receipt of an execute tone. The 22nd bit was a parity bit which was checked as the command was transferred into the programmer. The first bit was used to identify either a stored-program mode or real time mode of operation and was separated out as the word was transferred from the command decoder and used to set up the proper execution of the word. Word format is illustrated in Figure 8-31.

Instruction Sets - Command word instructions were separated into two sets: those instruction words that control internal functions of the programmer; and those instruction words that control other spacecraft subsystem functions.

Internal Instructions - The internal operation performed was designated in the command word by bits 2 through 6. The 9 internal codes were:

1) Telemeter Memory (TEM). This command was used to verify specific stored commands.

2) Wait Time (WAT). This instruction was used as a delay between stored commands.

3) Compare Time (COT). Instruction was used to initiate an event at a desired spacecraft time. The address was indexed each time the command was executed.

4) Execute Magnitude Minus (EMM). A magnitude command with a minus sign to control the maneuver direction.

5) Execute Magnitude Plus (EMP). Same as EMM except for sign.

6) Store Program Address (SPA). Command designated the address where the next stored program word transmitted from ground was to be stored. The address was indexed each time a stored program word was received.

7) Terminate (TER). A real time command used primarily as a backup command to stop a comparison of time or magnitude.

8) Jump (JMP). The stored program under execution was interrupted and the programmer jumped to the address designated.

9) Jump Modified (JPM). Same as jump except the address was indexed and there was no limit to the number of repeated executions.

External Programmer Commands - Seven operation codes provided magnitude commands, and six operation codes provided discrete functions in independent order. All spacecraft commands could be issued as real time or stored-program commands. Description of the external codes follows:

COMMAND WORD FORMAT

22	21. 7	6 . . . 2	1

BIT NUMBER (1-22)	FUNCTION AND STATE	
1	TRUE: REAL TIME COMMAND	FALSE: STORED PROGRAM COMMAND
2 THRU 6	OPERATION CODE	
6	TRUE: S/C FUNCTIONS	FALSE: PROGRAMMER FUNCTIONS
7 THRU 21	MANGITUDE OR FUNCTION TO BE CONTROLLED	
22	EVEN PARITY-GENERATED ON THE GROUND	

TIME WORD FORMAT

22	21. 7	6 . . . 2	1

BIT NUMBER (1-22)	FUNCTION AND STATE
1	FALSE: STORED PROGRAM COMMAND
2 THRU 21	TIME: 0 to 29.1 HOURS THIS VALUE WILL BE IN SPACECRAFT TIME FOR THE EXECUTION OF SOME COMMAND OR FUNCTION.
22	EVEN PARITY-GENERATED ON THE GROUND

FIGURE 8-31 WORD FORMAT

1) Velocity - Low Thrust (VEL) - The fifteen bits of binary magnitude were equivalent to the desired velocity change in 0.1 foot-per-second increments. The command caused thrust engine burning and integration of the resultant acceleration which was compared to the fifteen-bit constant. When equality was achieved, the VEL command was terminated and the program advanced. In the stored program mode, the command was valid only when addressed by an internal EMP command.

2) Pitch Plus (PIP) - The fifteen-bit binary magnitude was equivalent to 1/100th of a degree of angular movement about the pitch axis. Integration-of-gyro rate was compared to the constant and equality terminated the command. In the real-time mode, the command was used directly and caused a plus rotation. In the stored-program mode, the command was valid only when addressed by an internal EMP or EMM command.

3) Pitch Minus (PIM) - Same as PIP except for sign.

4) Roll Plus (ROP) - Same as PIP except movement is about the roll axis.

5) Roll Minus (ROM) - Same as ROP except for sign.

6) Yaw Plus (YAP) - Same as PIP except movement was about yaw axis.

7) Yaw Minus (YAM) - Same as YAP except for sign.

8) Discrete Commands (COM, CON, ACF, ACS, DHS, DAE, DEP, CAP) - The eight operation codes were combined with their respective fifteen-bit function codes to produce up to 15 unique functional commands per operation code. The function commands were fifty-millisecond pulses to the required spacecraft subsystem or to flip-flops for control of on-off commands. The total number of discrete commands provided was as follows:

Photographic Subsystem	23 Commands
Communication Subsystem	12 Commands
Structure and Mechanical Subsystems	9 Commands
Attitude Control	27 Commands

Programming - Maximum efficiency of storage space was obtained by combining individual discrete functions into subroutines that were addressed from a master sequence and reusing as many subroutines as possible. Discrete command words were formatted to provide a maximum number of independent subsystem functions that could occur simultaneously. A program flow diagram is shown in Figure 8-32. Each subroutine returned the program to a compare time which will control the time when the next event was executed. The mission profile command decision block was a series of jump commands which directed the program to the various subroutines.

The programmer clock was implemented with a 20-bit word that was incremented every 0.1 second, and provided a 29.1-hour recycle period. The compare time (COT) command is only capable of equality comparison for a specific time within the recycle period.

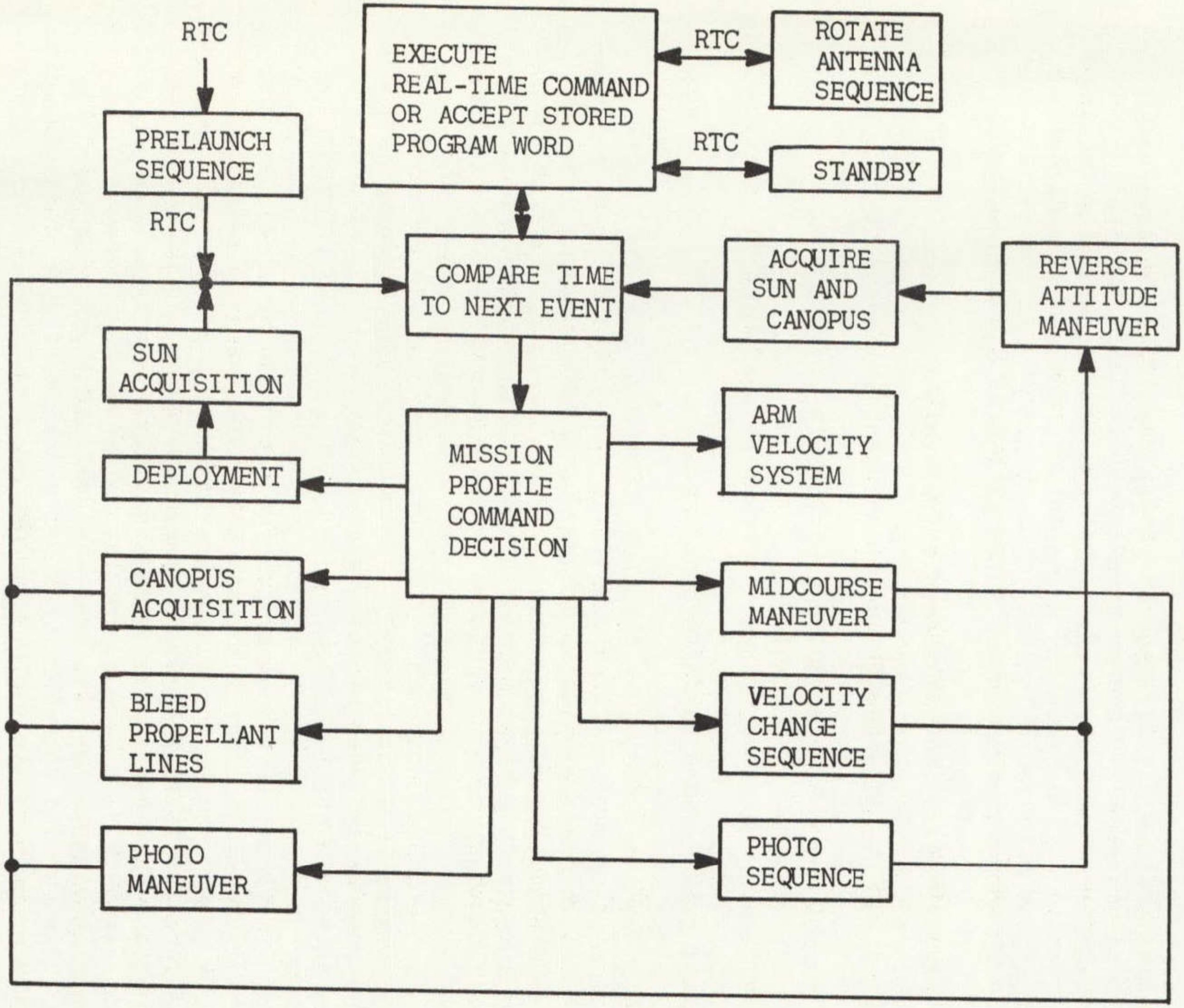


FIGURE 8-32: LUNAR ORBITER PROGRAMMING FLOW DIAGRAM

8.4.2 PROGRAMMER REQUIREMENTS

The programmer requirements evolved progressively with more detailed definition of mission and subsystem requirements. As subsystem changes occurred, it was expected that the programmer could accommodate them. In this sense, there was no firm specification on the total requirements until after the critical design reviews of all subsystems and of the total spacecraft. The environmental requirements were the same as for other equipment on the equipment mounting deck. The reliability allocation was .949.

The requirements for the programmer (as it was finally built) are listed below:

1) Command Capability, Sequencing, and Priority

The programmer shall decode, time, properly sequence, and execute the necessary stored program, real time commands, and signals to the following spacecraft subsystems:

1. Attitude Control Subsystem
2. Velocity Control Subsystem
3. Reaction Control Subsystem
4. Communications Subsystem
5. Photographic Subsystem
6. Switching Assembly
7. Power Subsystem

The programmer shall receive, condition, and utilize the necessary commands, power, and signals to accomplish the foregoing. Command execution sequence and reprogramming shall be as specified in the spacecraft mission event sequence and time line analysis. Real time commands shall take priority over stored program commands.

In addition, the programmer shall provide sufficient storage to control the spacecraft for three (3) lunar orbits.

2) Interrupt Capability

The programmer shall provide the necessary circuitry to permit interrupting of the stored program under execution by a real time command, the execution of the real time command, and the restoration, if feasible, of the stored program that was interrupted.

3) Program Verification

The programmer shall provide the capability to telemeter information from various programmer points such that basic program execution and operation of attitude control system can be monitored.

4) Time Reset Capability

The programmer shall provide the capability to reset the spacecraft time to any value upon real time command. In addition, spacecraft time shall be reset to zero by external hard line control.

5) Time of Command Execution

The programmer shall provide the capability of time dependent execution of commands to an accuracy of plus or minus ($\pm$) eight-tenths (0.8) of a second over an eight (8) hour time interval. The accuracy shall be based on the component's recording of time.

6) Time Generation and Recording

The programmer shall provide the capability of generating and recording time for a twenty-four (24) hour period with a resolution of one-tenth (0.1) of a second and a stability of one-tenth (0.1) of a second over an eight (8) hour interval. This capability to be repetitive with no loss in resolution or stability.

7) Execute Attitude Maneuvers

The programmer shall provide the capability at designated times of executing commanded three (3) axis attitude maneuvers and deriving spacecraft position from angular rate information, either as a function of stored program commands or as a function of real time commands. In addition, the programmer shall provide for removing the spacecraft attitude control from control by spatial references and restoring the spacecraft attitude control to control by spatial references.

8) Execute Velocity Maneuvers

The programmer shall provide the capability at designated times of executing commanded velocity maneuvers and deriving spacecraft velocity from linear acceleration information either as a function of stored program commands or as a function of real time commands.

9) Spatial Reference Acquisition

The programmer shall provide the capability at designated times of executing commanded maneuvers and accepting pertinent information as to acquisition of spatial references, to acquire spatial references either as a function of stored program commands or as a function of real time commands.

10) Computation

The programmer shall provide the capability to add and compare numerical (binary) magnitudes, to detect one (1) part out of eight hundred and sixty-four thousand (864,000), to integrate angular rate expressed as d.c. voltage, and to integrate linear acceleration expressed as the difference between two (2) pulse trains.

8.4.3 PROGRAMMER DEVELOPMENT AND OPERATION

Design objectives were defined in the programmer functional requirements specification. It was anticipated that many hundreds of photographic sequences would have to be transmitted to the spacecraft and, therefore, it would be desirable to minimize the number of commands required to perform those sequences. This resulted in the decision to build a system which would require transmission of only the sequence variables. It was also recognized that all the functions required to be performed by the programmer could not be established firmly and therefore a flexible design must be developed which could handle changes in producing command and command sequencing. Additionally, the flight programmer must have sufficient flexibility to be able to accommodate changes in the mission. Reliability was to be given high priority in both design and component selection.

8.4.3.1 PROGRAMMER TRADE STUDIES

Prior to actual hardware design, trade studies were pursued to establish the best design approach. The most important trades are described in the following paragraphs.

One of the initial studies traded capability of executing only stored program commands in the programmer versus the execution of both real time and stored program commands. The trade study considered incorporation of real time decoding matrix circuitry in the command decoder subsystem, or incorporation of these same decoding functions in the programmer. Since the programmer had to provide essentially all of the same commands (in storage) to control the spacecraft, the same decoding circuitry could be used for both functions. This resulted in a reduction of total required hardware.

A second trade study was conducted to establish whether the programmer should be made redundant or only single thread. The results of the trade study showed that the single thread system would meet the reliability requirements established for the programmer and that providing a second redundant system would be expensive in both weight and power.

Improper execution within the programmer was given serious consideration since any malfunctions during critical operational periods would either be relatively catastrophic to the mission or would execute photographic sequences not previously planned. An even parity bit was included in the words stored in memory to prevent a single bit error from causing improper command execution.

Early in the design phase a trade study was made to select the type of electronic circuitry to be used for the programmer. The trade involved the selection of integrated circuits versus discrete component circuits. In 1964 integrated circuits were just beginning to be mass produced and there was a very definite question as to their reliability for use in the program. The trade study showed however, that the number of components, their size, weight, and volume which would be required to build the system with discrete components would be prohibitive. Therefore, the decision was made to use integrated circuits. This approach affected other subsystems within the spacecraft. Since the programmer used integrated circuits the decision was made to use the same components for the command decoder and photographic subsystem digital circuits.

Timing for the spacecraft was required to be controlled in 1/10 of a second increments over a 16 hour period of operation. This required a crystal oscillator frequency stability of one part in 10^6. Since the clock was vital to spacecraft control and there was some question as to the reliability of the oscillator, the decision was made to use two of them, each with its own countdown chain, with the provision to switch between them in case of failure.

A trade study was conducted in the initial design of the programmer to determine if information transferred within the programmer should be accomplished in a serial manner or in a parallel manner. Since power, weight and volume were critical parameters for the design of the spacecraft, it was evident that a minimum number of integrated circuits should be utilized in the design of the computer. Serial operation was selected because it did provide a minimal number of circuits. The selection of serial operation for the programmer required the magnetic core memory output information be transferred serially to the rest of the programmer circuits.

The size of the magnetic core memory was dictated by the number of command words required to be stored for 16 hours of mission operation. From study of the mission profile and the photographic sequences and maneuvers required, it was immediately evident that the development of a standard program where all command words are stored independently for each of the maneuver sequences would cause the number of words required by the programmer to approach 512 words. Since a memory of this size exceeded the initial estimate, it was apparent that some method must be incorporated in the programmer design to reduce the amount of magnetic core memory. The most obvious solution was to make the programmer execute repetitive sequences wherever possible in the mission programming. It was estimated that approximately 30 command words would be required to execute a single photographic sequence. Within this sequence however, only 8 words were required to be modified to accomplish a second sequence. By developing special instructions it was possible to program 7 photographic sequences (16 hours of operation) using only 120 words of memory. From this information the magnetic core memory for the flight programmer was determined to be 128 words.

The programmer was required to command and fire all spacecraft pyrotechnic devices. A trade study was conducted to determine where to put the firing circuits and what kind of circuit configuration should be used. Since the pyrotechnic circuits were required to supply 5 amperes to the pyrotechnic bridge wires, it was decided that these circuits should be separated from the flight programmer in order to eliminate any RFI problems. These pyrotechnic circuits were then put in a separate unit called the switching assembly and physically displaced from the vicinity of the programmer electronics.

8.4.3.2 Switching Assembly

The block diagram of the switching assembly is shown in Figure 8-33. Low-level signals from the programmer were converted in the switching assembly to high-current (5-10 amperes) or voltage level (28 volt) signals. The switching assembly weighed 6.3 lb and used 2.5 watts power continuously in addition to the short current pulses for the discretes. The arming switches were closed by umbilical command and their actuation monitored prior to launch. The inhibit switches provided further protection against inadvertent firing of the squibs. The squib drivers were directly connected to the squib bridgewires. This assembly provided isolation of large signal-transient and electro-magnetic fields from other equipment.

8.4.3.3 Programmer Testing

Upon completion of fabrication, the programmer units were subjected to a series of tests to demonstrate that the units complied with all specified design requirements. The three classifications of test at the unit level were (1) the Flight Acceptance Tests, (2) a Qualification Test, and (3) a Reliability Demonstration Test. The units were also required to pass similar Spacecraft Level Test Requirements.

The System Design Verification tests provided important operation and performance information about the programmer when combined into the operating subsystems. One of the most important of these tests was the Attitude Control Subsystem test on the Air Bearing Simulator. The inability to execute stored programs and the resulting changes are discussed in Section 3.0.

One major problem area and two major potential problem areas were discovered during component testing. The first potential problem area was noted during the vacuum thermal portion of the flight acceptance test on the first programmer unit. During this portion of the test, what was considered an excessive number of failed components and solder connections were noted. As this unit had passed the pre-functional test without detection of these defects, temperature tests were initiated at the board and module level to detect and correct these defects before final unit assembly. The second potential problem area was discovered later in the testing program. It was found that the static discharge from personnel wearing nylon garments when handling micrologic components was sufficient to cause component failure. This problem was alleviated by requiring all nylon garments to be replaced by cotton garments, and that all personnel "ground" themselves before handling any components.

The one major problem that was detected during testing was a micrologic module contamination problem. The problem was associated with a certain lot of modules. One hundred and thirty-four of 214 modules that were received from that particular lot indicated evidence of contamination. The contamination appeared as a brown or black amorphous material on the outside of the micrologic case surrounding the external gold plated leads at the case lead interface. The contaminant was conductive and when spread from lead to lead resulted in a short circuit. The contaminant was metallic lead which resulted from a module sealing procedure by the manufacturer. The problem was corrected by replacing all modules that were received from that particular lot.

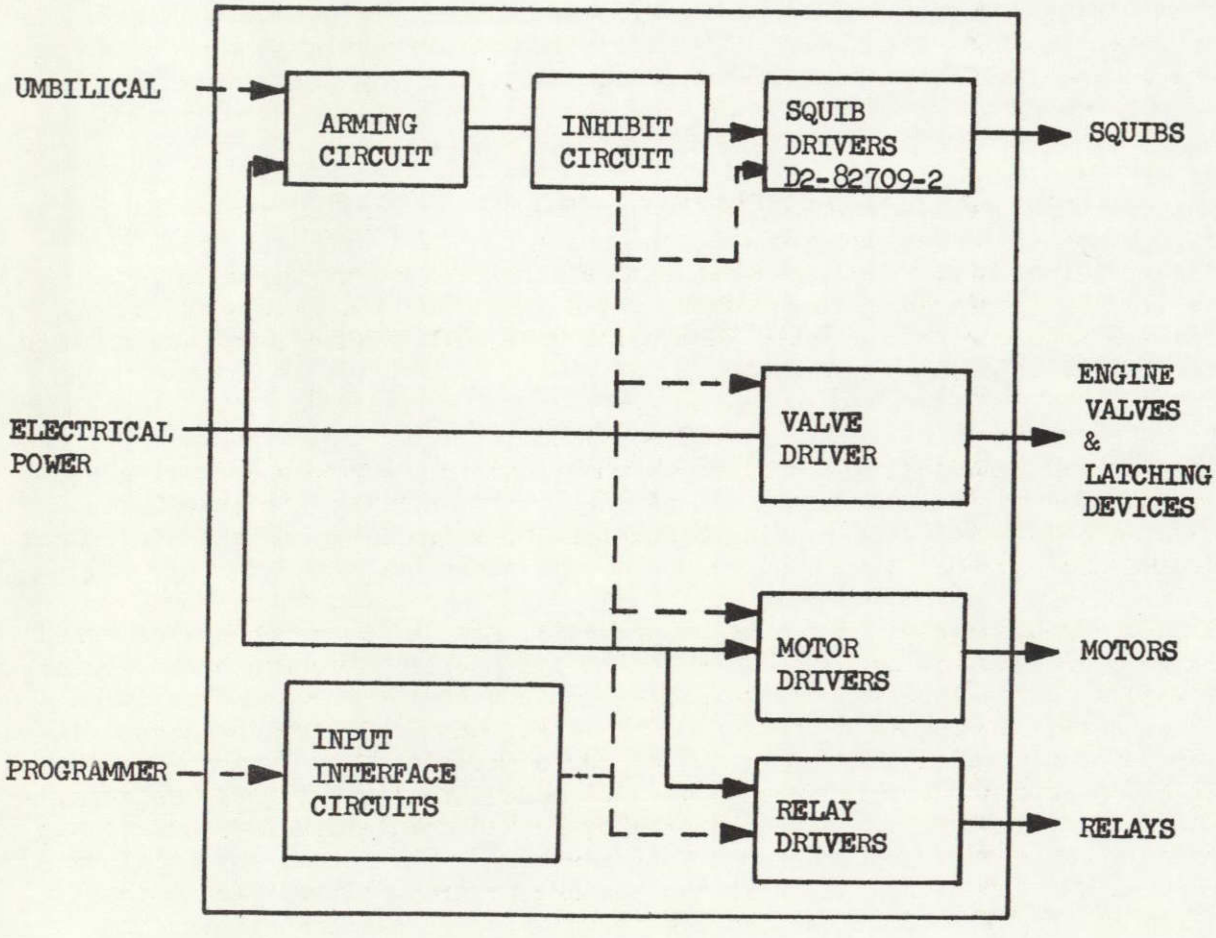


FIGURE 8-33 : SWITCHING ASSEMBLY

Of the failure reports made with respect to the programmer during testing approximately 39% of these reports were attributed to design, fabrication and workmanship deficiencies; 44% were attributed to test procedure, test equipment, and test personnel errors. The remaining 17% were attributed to part failures and causes listed as unknown. The above figures do not include part failures detected at module and board level tests.

8.4.3.4 Programmer Mission Performance

Performance of the programmer and switching assembly was satisfactory on all flights. For convenience, the pertinent data is summarized in Figure 8-34. There were no known hardware failures. There were instances of anomalous spacecraft operation that occurred. The unexpected or unusual events are discussed in the brief in the following:

Mission I - Primary Mission Phase

Command activity, although planned in minute detail before the start of the mission had to be completely revised. This resulted in occasional frantic command activity due to late definition of modified mission parameters and late parameter changes. However, all mission requirements were met, with only two wrong commands being transmitted to the spacecraft.

Mission I - Extended Mission Phase

During the extended mission activities, performance of the programmer was nominal, with the exception of a low voltage condition encountered on day 288 of 1966. This condition resulted from a series of improper maneuver commands being transmitted to the programmer and was discussed in section 7.0. The battery voltage of the spacecraft was insufficient to maintain the proper logic levels on the programmer. This resulted in improper operation. It should be pointed out, however, that the programmer was operating correctly with a spacecraft bus voltage of only 17 volts. The minimum voltage design requirements was 21 volts. The programmer was also subjected to temperatures in excess of the design or test requirements with no malfunctions.

Mission II

This was the longest of the five missions yet had the least amount of difficulty with the programmer. Although both temperature and voltage limits beyond the programmer design requirements occurred, there were no anomalies.

Mission III - Primary Mission Phase

During Mission III a programming error resulted in failure to generate a photo activation time code for the photos of site III P-1. This error resulted from a deviation from the planned photo command sequence and misinterpretation of programmer operation by the command programmer analyst. The deviation from the planned command sequence was to accommodate a real time command to change camera shutter speed.

	I	II	III	IV	V
LAUNCH DATE	8/10/66	11/6/66	2/5/67	5/4/67	8/1/67
LUNAR ORBIT TRAJECTORY DATE	8/14/66	11/10/66	2/8/67	5/8/67	8/5/67
TERMINAL DATE	10/29/66	10/11/67	10/9/67	7/17/67 101-167*	1/31/68
DAYS OF OPERATION, DAYS/HOURS	80/1920	338/8112	247/5928	97/2328	192/3608
DISPOSITION	Lunar Surface	Lunar Surface	Lunar Surface	Lunar Surface	Lunar Surface
PRIMARY MISSION LENGTH, DAYS	35	30	32	28	26
• REAL TIME COMMANDS TRANSMITTED	1,988	1,289	1,266	3,666	1,703
• STORED PROGRAM COMMANDS TRANSMITTED	2,522	2,282	2,349	3,445	2,822
• COMMANDS EXECUTED	15,000	12,000	15,000	20,000	14,000
• ORBITAL PERIODS, HRS/MIN	3h-26m	3h-28m	3h-28m	12h-1m	3h-11m
• CLOCK DRIFT (SECONDS)	+1.6	-1.2	-0.16	-0.96	+0.66
• NUMBER OF PICTURES (WIDE ANGLE AND TELEPHOTO)	422	417	327	359	425
NUMBER OF VELOCITY MANEUVERS	5	7	7	4	6
NUMBER OF LUNAR ORBITS	547	2325	1702	225	956

TOTAL SPACE OPERATION TIME OF PROGRAMMER NO FAILURES - 954 DAY/22,896 HOURS

PREDICTED UNIT RELIABILITY (MTBF) - 13,900 HOURS

*L.O IV DISAPPEARED ON 7-17-67, ORBIT DECAY TO CRASH ON SURFACE PREDICTED FOR OCT. 1967

FIGURE 8-34 SUMMARY OF LUNAR ORBITER PROGRAMMER PERFORMANCE

Mission III - Extended Mission Phase

During the extended mission phase, it appeared that the flight programmer unit was malfunctioning. On several occasions memory storage locations were found to be altered and improper command sequences to have been executed. It was determined that commands transmitted to other spacecraft concurrently in Lunar orbit were accepted as valid commands by the Mission III spacecraft receiver and were issued to the flight programmer either for storage or execution. Although each spacecraft had a separate address code, it was not adequate under conditions of weak signal strength, to exclude extraneous bits being received. The problem was alleviated by controlling transmission times with respect to spacecraft orbit locations. Also the radio frequency of the spacecraft being commanded was pulled away from the other spacecraft receiver frequency by the DSIF uplink while in lock with the spacecraft receivers.

Mission IV - Primary Mission Phase

Subsystem anomalies during the mission resulted in numerous non-standard command sequences being transmitted to the spacecraft. As a result of this increased command activity, three programming errors were made. The first error was due to the failure to update a modifiable command. This caused initiation of a wrong command sequence. The second error was the premature termination of a maneuver by the inadvertent execution of a real time command that was being held in the command decoder. The third error resulted in the late initiation of a command sequence and was caused by misinterpretation of a command sequence by the programmers. None of these errors produced serious, irrevocable conditions.

Mission V

Command activity was considerably reduced and improved during mission V with respect to earlier missions. This was due in part to further improvements in the information flow to the programmer analysts by the other mission personnel and the earlier acquisition of required mission data that was necessary for incorporation into the command sequences.

8.4.4 Programmer Conclusions and Recommendations

The programmer provided satisfactory control and sequencing of spacecraft events throughout all missions. In addition to satisfying all design requirements, it provided the flexibility to accommodate both mission anomalies and changes in mission objectives. Based on the experience with the Lunar Orbiter programmer, the following conclusions are made:

1) The selection of a simple, straightforward, single thread design with limited redundancy (redundant oscillators and countdown chain) resulted in a unit that was reliable and easy to operate. The reliability demonstrated by the units exceeded the reliability allocations of the design requirement. Being straightforward and easy to operate was of particular value to both the test program and to mission operations personnel.

2) The selection of special instructions and the use of subroutines were effective in minimizing the amount of memory required to accomplish any specific mission task.

3) Possible deleterious effects of EMI to the low level logic circuitry was avoided by the decision to place all high current or voltage switching circuits (such as squib and engine valve drivers) in a physically separate unit (the switching assembly).

Recommendations for "doing differently if one had it to do over" are:

1) Verification of programmer storage was difficult. It required one ground transmitted command for each word telemetered. Provision should have been made to allow block transfer of the entire magnetic core memory to verify the information stored there. This would have aided test significantly. The decision on the program not to incorporate this capability was made on the basis of minimizing parts count for reliability purposes.

2) The programmer was implemented with a half adder. A full adder should be implemented to allow for incrementing and decrementing of time and magnitude values for greater flexibility in command execution. Here again the decision against such inclusion was based strictly on parts count or reliability. Significant operational flexibility would have been gained by such an adder. Work-around methods on programming would have been simplified thus reducing the load on operations personnel.

3) The storage capacity of the programmer was only 128 words. As such the unit had to be remapped periodically throughout the missions placing heavy reliance on the data link. A greater storage capacity would have reduced the continuous mapping and search for storage locations during missions operations.

4) Programming of the system would have been easier if the "modified jump" command had been limited to a specific number of executions. This would have allowed more flexibility in the transfer between subroutines and also reducing the load on ground operations personnel.

5) Provide for a more general purpose instruction set to permit greater flexibility in recursive programming to minimize storage space. It would allow maximum use of programming techniques to minimize programming loads and coding errors.

6) Maneuver commands were executed through a comparison of accumulated magnitude and the commanded magnitude. A timed backup should be provided in case of breakdown of the comparison process.

7) The programmer was implemented at the time of development with state-of-the-art hardware. However, with the advance in technology, consideration should be given to newer low power devices which could reduce power requirements by a factor of 4.

8) Cordwood module interface circuits should be replaced with integrated circuits.

8.4.5 CLOSED LOOP ELECTRONICS DESCRIPTION

The Closed Loop Electronics (CLE) was located in the Flight Electronics Control Assembly. Functionally the unit was a part of the Attitude Control subsystem discussed in Section 3.0. The CLE circuit schematic is shown in Figure 8-35. The unit was basically all analog in operation but provided digital (ON-OFF) type control signals to the reaction control jets.

Specifically the CLE performed functions of amplifying, shaping and discriminating signals from the inertial reference unit, celestial sensors, and the flight programmer. It accepted signals in all three axes from the IRU rate or rate integrate modes. The unit also connected and disconnected celestial reference sensors upon command from the programmer.

The closed loop electronics circuits were designed around the power supplied by the programmer, which was ± 15.0 VDC, + 6.0 VDC and + 28.0 VDC. The unit consisted of 72 solid state circuits of 23 configurations.

8.4.6 CLE REQUIREMENTS

The CLE was required to accomplish two primary tasks.

1) The CLE shall provide for the control of three axis attitude control thrusters as a function of spatial reference signals, spacecraft dynamics, and inertial sensors. In addition, the CLE shall provide for the control of attitude control thrusters as a function of attitude commands, angular rate, and spacecraft dynamics.

2) The CLE shall provide analog displacement error signals to the pitch and yaw thrust vector control actuators as a function of inertial reference position signals and spacecraft dynamics.

These tasks were satisfied by performing the following functions.

1) Close rate and attitude control loops with inputs from inertial reference unit, sun sensors, and star tracker.

2) Provide lead/lag compensation on inertial reference unit attitude signals for reaction control and thrust vector control.

3) Amplify and limit sun sensor outputs.

4) Accept mode switch commands from the programmer.

5) Provide valve drivers for plus and minus thrusters in three axes.

6) Provide "one-shot" to thrusters for minimum impulse bit operation.

8.4.7 CLOSED LOOP ELECTRONICS DEVELOPMENT AND OPERATION

The detailed circuit parameters of the Closed Loop Electronics were dictated by the functional interface requirements of the chosen Inertial Reference Unit and the reaction control system. As such the alternatives available to the designer were relatively limited. The circuit techniques, being linear, were straight-forward and had a proven history of past usage for this type of applications. The most critical problems were: 1) achieving high reliability circuits through worst-case design methods, 2) the selection of components that would enhance operational integrity, and 3) the minimization of the power dissipation of the unit.

A primary consideration in the design of the CLE was to keep the power level as low as possible. This meant the circuits were to be designed with performance characteristics secondary to power constraints. This objective was met with a final power dissipation figure of 0.44 watts in normal operating mode and 25.5 watts with everything turned "on" including four thrusters. This was considerably less than was possible to attain with commercially available operational amplifiers. This saving was accomplished by optimizing each circuit to the task performed, and by judicially keeping power driving stages of the circuits in the "off" state during quiescent conditions. The performance capability of the sun sensor operational amplifier, which is typical of many CLE circuits, is shown in Figure 8-36.

8.4.7.1 DESIGN EVOLUTION

The evolution of the closed loop electronic unit from a set of general requirements to the final flight article was one of continuous design iteration. Starting with a set of preliminary input and output interface characteristics, representative circuits were synthesized and breadboarded. The prototype breadboards were then tested under worst case conditions in voltage variations and temperature extremes. As detail requirements evolved and became more definitive, the breadboards were modified and retested to more stringent design parameters and tolerances. The development of the CLE did not encounter any major problems. The most significant design events are discussed in the following paragraphs.

The unit originally required zero crossing detection circuits that acted on gyro position outputs and celestial sensor outputs with the objective to reduce the limit cycle deadband contribution to maneuver error. These circuits were deleted prior to the critical design review after analysis indicated the gain derived did not offset the penalties incurred in design complexity. In the design of signal switches, an investigation was conducted on the advisability of solid state switches versus relays. Relays were rejected for reliability considerations. The choice was for solid-state switches with "Field Effect Transistor" output stages.

Though not a major problem, difficulties were encountered in how best to determine the "aliveness" of the closed loop electronic unit when mated to the IRU, celestial sensors, and the reaction control system. Test points were added, along with signal conditioning circuits for telemetry monitoring functions, to permit operational checks. The objective was to be able to

CIRCUITS	REQUIREMENTS	PERFORMANCE
Operational Amplifier - (Sun Sensor)		
Voltage Gain - (Closed Loop DC)	104.8 V/V $\pm$ 2%	104.8 V/V $\pm$ 2%
Input Impedence - (Closed Loop)	Equal to Series Sensing Resistor	
Output Impedence	$50\,\Omega$	$10\,\Omega$
Frequency Response - 3 dB (Closed Loop)	10 Hz	100 Hz
Output Limits	$\pm$ 5.0 min.	$\pm$ 5.0 V
Temperature Drift	$2.5\,\mu V/^{o}C$	$1.0\,\mu V/^{o}C$
Supply Voltage	$\pm$ 15 V $\pm$ 1%	$\pm$ 15 V $\pm$ 1%
Supply Ripple & Noise ATTEN	20 dB	34 dB
Eq. Input Noise	10 μ V RMS	5 μ V RMS
Maximum Load Allowable	2.8 K	2.0 K
Voltage Gain - Open Loop	74 dB min.	86 dB
Power Dissipation	200 MW	20 MW
Undistorted Output Range	$\pm$ 4.5 V min.	$\pm$ 5.0 V

TABLE 8-36 - OPERATIONAL AMPLIFIER CHARACTERISTICS isolate a failed unit to the black box replacement level. In this sense the check points provided were adequate.

Protection of the reaction control thruster driver amplifiers required that coil induction voltage suppression be provided to limit the voltage spike which appeared across the output stage of the amplifier when the coil of the solenoid was turned off. Any voltage limiting placed across the coil increased the drop out time of the valves and thereby increased the minimum pulse of the jets. The original circuitry designed to limit the voltage spike performed the function adequately, but was subsequently modified when more realistic information on the 0.05 lb. thruster solenoid and drop-out time requirements became available. Tests to optimize the valve off delay showed that the inductive kickback suppression diode voltage must be increased to about 22 volts (see Section 4.0 for details). The power rating of the diodes was also increased to withstand higher power surges. The actual change effected put two of the diodes previously selected in series because they were on the approved part list and available from stock.

The spacecraft design and analysis progressed in parallel. This required that some provisions be made to allow selection of electronic part values quite late in the design cycle. The latest planned change which occurred was the thrust vector control compensation networks. Resistor and capacitor values were finally firmed up after the simulated closed loop thrust vector control subsystem test conducted in the fall of 1965. Again, the precise values used were selected on the basis of approved available parts which differed only slightly from those desired.

8.4.7.2 RELIABILITY

Reliability of the closed loop electronics unit was achieved through worst-case design techniques, failure mode effect and criticality analysis, and through careful selection and screening of components. Each CLE circuit was breadboarded and subjected to functional operation checks, including evaluation at temperature extremes of -30° C to $+70^{\circ}$ C prior to final packaging. Modules that failed under functional tests were dissected and analyzed to determine the failure modes. The information was fed back to the production people to prevent future problems.

Heavy emphasis was placed on component selection and part derating. 86 percent of the resistors were stressed at only one percent of the rated power and only two resistors were employed at 20 percent of rated power. All transistors, except for six, were used at less than one percent of rated power. The six were used at 20 percent of rated power, and subjected to no instantaneous voltages higher than 50 percent of rated voltage. All diodes were used at less than one percent, except for 12 that were used at 15 percent of rated power. Of the capacitors, 12 were used at less than 45 percent of rated voltage, 15 were used at less than 35 percent, 20 were used at less than 15 percent and the remaining 16 capacitors were used at less than three percent of rated voltage.

The final closed loop electronics configuration had a predicted failure rate of 2.745×10^{-6}.

8.4.7.3 PACKAGING AND MANUFACTURING

The closed loop electronic circuits were discrete component transistor-resistor circuits and were packaged in welded cordwood module configurations. The individual modules were soldered to CLE module mother boards which are plug-in type and are completely replaceable and interchangeable. Conceptually this technique is relatively straight-forward and permits the greatest packing density for an allocated volume. However, because of this need for high density, electrical and fabrication problems were encountered. Some circuit layouts were found to be susceptible to cross-talk interference between the input and output signal paths resulting in circuit oscillations. This problem was eliminated by modifying layout of sensitive components. Fabrication difficulties were the result of immaturity of manufacturing processes. The majority of failed modules resulted from poor workmanship or careless handling of parts, i.e., poor or broken welds, use of components with erroneous values, overstress components, etc. Manufacturing also encountered many unforeseen problems in implementing the electro-deposited solder process for cordwood modules and was finally required to switch to the use of a conventional KPR-resist type of etched card coated with electrolysis tin in order to meet the schedule. A corollary to this problem was the procurement of parts under Minuteman specification with tinned leads which were not compatible with L. O. manufacturing processes. This required rework of leads prior to use. Modules which tested satisfactory prior to application of potting compounds, failed tests following potting. The potting processes and compounds employed stressed the components and caused geometrical changes, throwing the potted modules out of specified electrical tolerances. Damage to module header pins was also experienced as a result of poor potting process specifications. This was alleviated by using soft silicone rubber plugs around the header pins entering into the molds allowing seepage of materials and easy removal of plugs.

8.4.7.4 MISSION PERFORMANCE

The Closed Loop Electronics performed satisfactorily on all five missions. There were only two indications of off nominal or out of tolerance operation. During the extended mission phase of L.O. IV the plus pitch jet one-shot appeared to be malfunctioning. The telemetry indication from which this was deduced was a continuous pitch jet "on" signal with no accompanying change of pitch rate or position from the edge of the deadband.

The second out of tolerance condition occurred during the extended mission of L.O. V. It appeared that the minus pitch threshold detector was triggering at -1.1 volts instead of -2.0 volts. This shifted the deadband on one side. The effect was evident in maneuver rate, especially in wide deadband, and in limit cycle operation.

The wide deadband pitch minus maneuver rate was -.26 degrees/second instead of -.05 degrees/second. It was estimated that this caused the use of an extra .25 pounds of N_2 gas over a 126 day period. The limit cycle deadband was similarly reduced on the plus pitch side by approximately 50 percent in both wide and narrow deadzone. Since neither of these anomalies jeopardized the primary mission objective and had negligible effect on the extended mission there was no failure investigation made.

8.4.8 CLOSED LOOP ELECTRONICS CONCLUSIONS AND RECOMMENDATIONS

The major conclusions of this study of the Closed Loop Electronics are:

1) The things which contributed most to the success of the CLE design were:

   (a) Use of worst case design methods.

   (b) Selection of parts to enhance reliability, e.g., no part was used at greater than 50 percent of rated voltage; most were used at one percent of rated power.

   (c) Extensive engineering development testing of breadboard circuits and prototype assemblies as the design progressed, including evaluation at extremes of temperature of -30 degrees C and +70 degrees C.

   (d) Use of proven well understood circuits.

2) The ability to make changes late in the design cycle was an important asset of the design, e.g., the final values of the TVC compensation networks were dependent on total spacecraft dynamics and not available until late 1965.

3) The use of welded cordwood modules enhanced the circuit integrity once the process was completed satisfactorily.

4) The thermal design was more than adequate for the power to be dissipated.

5) Manufacturing processes introduced difficult problems to solve, e.g., the potting of modules; the tinned leads on components were not compatible with welding.

Recommendations for doing differently if one had it to do over are:

1) Parts procurement specifications and selection should be checked for compatibility and availability before releasing actual fabrication orders.

2) The output channels of the CLE should have been tested at the component level with the expected impedance of the interfacing component, e.g., the TVC actuator jitter problems when first tested on the spacecraft.

3) Many of the operational amplifiers and other analog circuits could now be replaced by integrated circuits if weight and power limitations demanded.

8.5 THRUST VECTOR ACTUATOR

The thrust vector actuator was a new component which was developed to the specific performance, environment, and envelope requirements of the Lunar Orbiter. The actuator was one component of the thrust vector control subsystem described in Section 5.0. The function of the actuator was to position the gimbaled engine in response to error signals from the gyros and closed loop electronics.

8.5.1 ACTUATOR DESCRIPTION

The actuator was an electromechanical position servo and is illustrated in Figure 8-37. It was sealed and pressurized with nitrogen gas to 7 psia to prevent problems with bearings and motor brush operation in the space vacuum environment. The value of 7 psia was chosen so that the pressure differential effects would be approximately equal for sea level testing as for vacuum operation. The output shaft was an acme threaded jackscrew which was sealed by an accordian type metal bellows. The actuator position was maintained at each end burn position by switching off the 28 volt power to the actuator at the conclusion of each velocity maneuver.

8.5.2 DESIGN REQUIREMENTS

Specific requirements imposed on the actuator are listed in Figure 8-38. These were determined by the demands of the thrust vector control subsystem of which the actuator is a part.

A position actuator rather than a rate actuator was specified because it simplified other portions of the TVC subsystem, and because precision position feedback was easier to implement.

In order to maintain thrust alignment with the c.m., a requirement for irreversibility was set. During the motor operation, the actuator was in a controlled mode and engine vibration did not affect the actuator motion. Between burns in a space environment the load on the actuator was small and the required friction torque for irreversability was small. The worst load environment was due to vibration during the boost phase and this determined the friction level for irreversability. Following final design it was determined that the actuator would not be irreversable under the maximum (3σ) boost vibration environment and the actuator was allowed to move at a rate not to exceed .05 inch per second. An operation procedure was developed which recentered the actuator to its electrical null after boost and prior to first motor ignition to circumvent this problem. As it turned out, the actuator never moved off null during boost on any of the five flights and the recentering was not necessary. Due to the implementation of the thrust vector system in which a position actuator was employed, spacecraft center of mass offset is a significant error source. In order to reduce this error, it was necessary to provide for mechanical length adjustment on the actuator to align the thrust vector through the measured c.m. with the actuator at its electric signal null. Mechanical adjustment capability of +0.25 to -0.15 inches (2.4 degrees to -1.46 degrees of nozzle) was provided. The stroke limit was determined by calculation of the maximum travel required by the

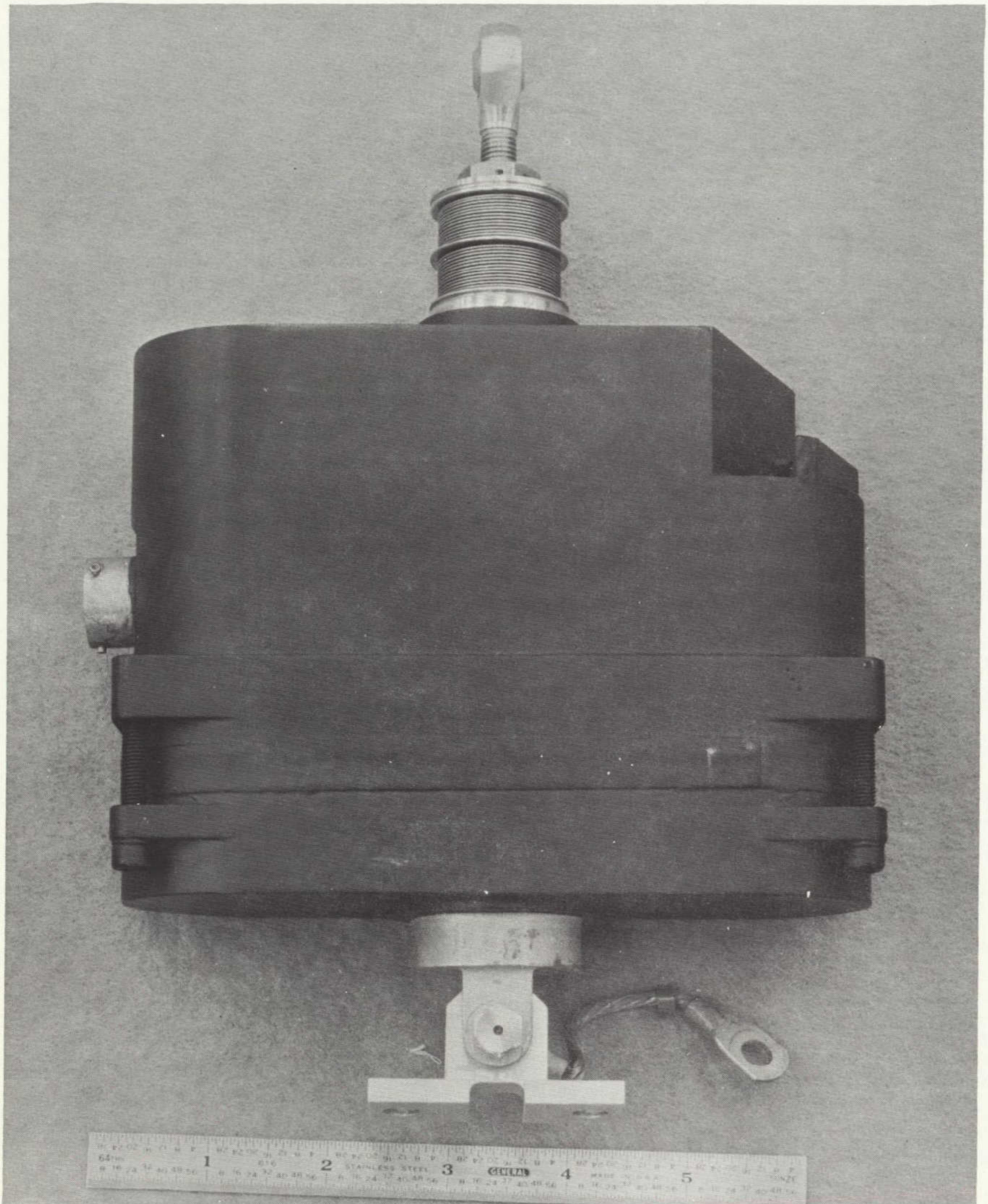


Figure 8-37: THRUST VECTOR CONTROL ACTUATOR

REQUIREMENT	EXPLANATION
1. Position Actuator	1. Predictable initial position of null and easy to check out and monitor.
2. Irreversible	2. Maintain engine prelaunch alignment through C.M. during boost. Minimize start transients by retaining engine position established during a preceding burn.
3. Mechanical Length Adjustment about nominal of 8.5 inches	3. Permit prelaunch adjustment to align engine through spacecraft center of mass with the actuator at electrical null.
4. Performance: Stroke ± 2.8 degrees Rate = 10 deg/sec Resolution = ± 0.018" (0.18 degrees) Natural Frequency = 2.5 CPS Load = 3.0 lbs. including: - Gimbal Bearing Load - Thrust Offset - Engine Hose - Heat Shield Inertia Load = 0.05 Slug ft^2 (6 inch arm)	4. To provide required system performance with at least 6 db gain margin from startburn to endburn conditions including effect of: a. Solar panel and antenna flexibility b. Shortburn requirement with: - Actuator hardover position - Maximum C.M. offset - Reaction control recovery capability without spacecraft attitude over shoot exceeding gyro gimbal limits c. Maintain thrusting vector direction within $\pm .4^\circ$ (3σ) of initial attitude.
5. Solid state servo electronics were included in actuator. Nominal scale factor = 1 volt/degree ± 5 percent.	5. The actuator performance as specified is a very intimate interface between mechanical and electronic design. Minimum development time was a factor.
6. The nominal thermal environment was 60°F $\pm 25^\circ$. In addition a thermal soakback - 175°F at rod and for one hour after engine burn.	6. An added thermal constraint was required on the actuator due to heat soakback from the engine subsequent to engine burn.
7. At least 18 day life.	7. Last planned engine burn was to obtain photo orbit within 18 days of launch.
8. Weight was 3.55 lb. Original objective was 2.2 lb.	8. Weight budget.
9. Power - 3.2 Amps maximum peak from 28 volt nominal source, 0.55 Amps nominal steady state.	9. Power budget; also internal heating.
10. Hermetically sealed with dry N_2 gas with Helium trace.	10. To withstand space environment, Helium trace for leak detection purposes.

FIGURE 8-38 THRUST VECTOR ACTUATOR DESIGN REQUIREMENTS

actuator to align the thrust with the center of mass plus adding an additional 1.5 degrees of nozzle deflection for control purposes. The additional 1.5 degrees was determined on an analog computer to give adequate control acceleration for system response.

The actuator frequency response limit was specified to be greater than the expected structural frequency. Due to the nonlinear response of the actuator caused by rate and acceleration limiting, the frequency response was defined for two amplitudes. The frequency response requirement is shown in Figure 8-39. The high amplitude response requirement has a lower frequency roll off due to rate and acceleration saturation.

The rate requirement (10 degrees/second) was set high enough to assure adequate dynamic stability margins with an engine initial misalignment with the c.m. of 3.8 degrees and stroke limits of ±2.8 degrees. The rate requirement was based on analog computer simulation results.

8.5.3 DEVELOPMENT AND OPERATION

A significant factor in the development of the actuator was the extremely compressed time scale available for development and qualification. Procurement specifications for the actuator were released early in September, 1964, following the decision to use the gimbaled engine. This was more than five months after contract go-ahead. Only 3 1/2 months were allowed to deliver the first engineering test model. The first qualified flight unit was scheduled for delivery in eight months.

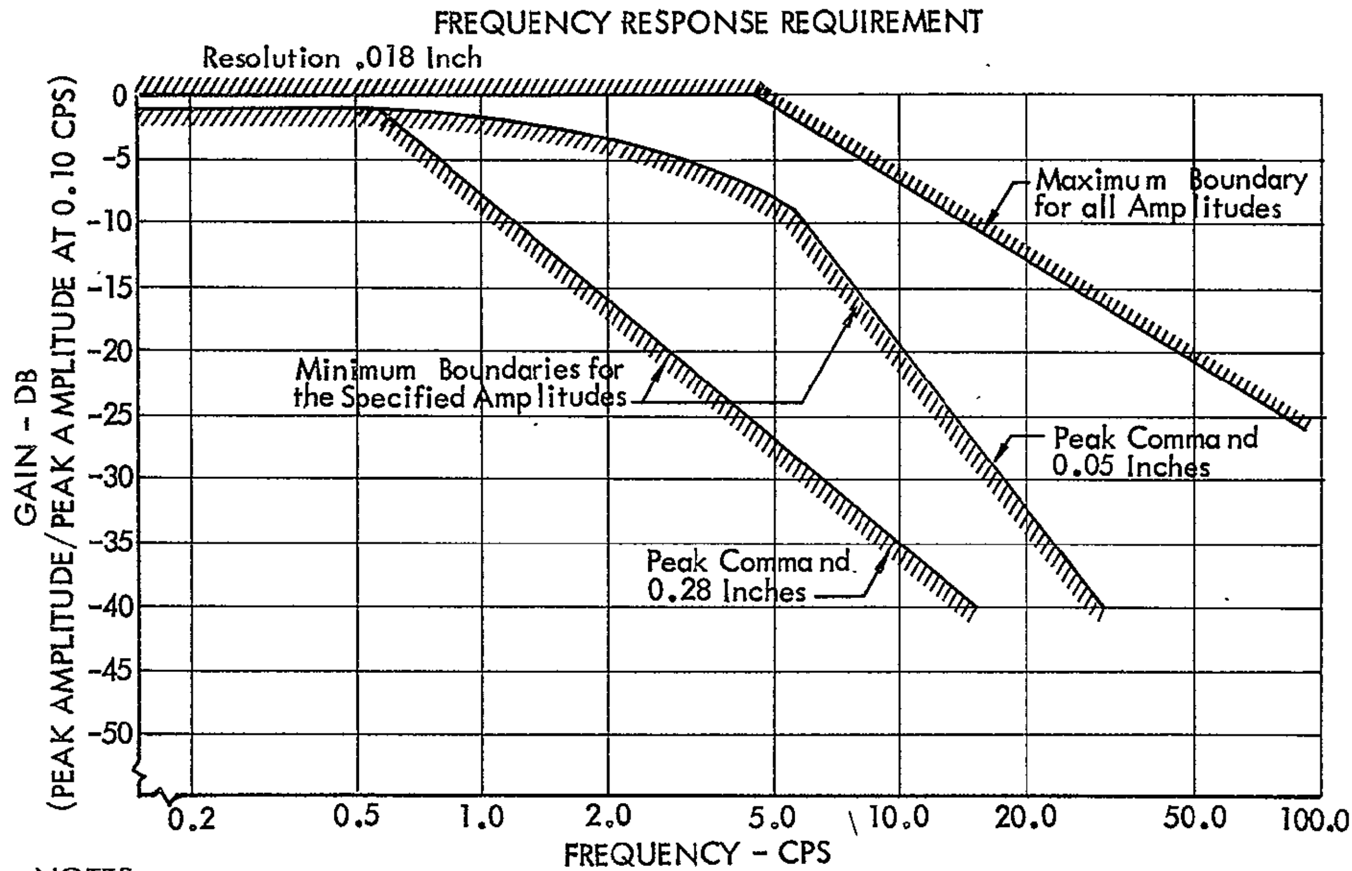
8.5.3.1 VENDOR SELECTION

There were three vendor replies to the actuator request for proposal. Two of them were similar to the chosen design. The third was a stepper motor concept. The stepper motor concept was discarded because it was felt that a stepper motor was unproven for this application.

Between the other two vendors one had neglected the effect of the hermetic seal on the actuator load. The load on the output due to pressurization was actually larger than any of the loads listed in the specification. The proposed design from Kearfott Inc. was selected. The actuator vendor had three months from go ahead to delivery of the first engineering development unit. During this time period a vendor preliminary design review and critical design review were held. Partly due to this extremely tight schedule and partly due to poor quality control on the actuator manufacture, several significant problems were uncovered during the test programs at Boeing.

8.5.3.2 DEVELOPMENT PROBLEMS

A summary of problems encountered during the developmental and early system integration testing phase and the rework or redesign required to cure the problems is shown in Figure 8-40. The first major problem was the failure of the actuator to meet the required rate limit under all design conditions which included design loads, low input voltage, and external pressure. In order to cure this problem more careful assembly and testing of the jackscrew



NOTES:
1. Under Design Load of ± 18 inch Lb.
2. Frequency Response about the Actuator Null
3. Linear Resolution - .018 inch

FIGURE 8-39 THRUST VECTOR ACTUATOR FREQUENCY RESPONSE REQUIREMENT

PROBLEM	CAUSED BY	REMEDY
1. Power transistor failure in development actuator.	1. Design of Darlington switching circuit electronics without due regard to inductive nature of load or back EMF of motor.	1. Larger transistors incorporated.
2. Low actuator rate	2. o Excessive actuator friction. o Low motor torque o Increase in design load	2. o Improve mechanical assembly procedures with maximum load tolerance. o Pre-assembly motor run-in with minimum torque tolerance. o Increase motor output.
3. Actuator motion during vibration	3. Insufficient reverse impedance	3. o Increased motion tolerance o Center actuator prior to first burn during flight operation.
4. Erratic actuator operation and high electrical power usage.	4. o Electro-magnetic interference sensitivity o Coupling of actuator and control electronics	4. o Modify actuator control electronics. o Provide filtering to decouple system. o Separate command and power ground circuits. o Increase actuator deadband (increase resolution tolerances). o Added loading resistor to C.L.E.
5. One actuator leaked.	5. Porous casting.	5. Rejected part, one time occurence.

FIGURE 8-40 THRUST VECTOR CONTROL DEVELOPMENT PROBLEMS

assembly was required. Later the motor size was increased to achieve the minimum required torque at low voltage. The electronics were changed, including larger motor drive transistors, to reduce voltage drops to the motor in order to increase the motor voltage for higher power output. One significant result of the low rate problem was a requirement to "run in", i.e., operate, the motor prior to actuator assembly. The motor "run in" was required to achieve a predictable and dependable power output for a specific power input. Motor power output and jackscrew assembly torque were measured prior to assembly to insure that required rates would be met following assembly.

Erratic operation and high power consumption due to internally and externally generated noise were a severe problem. Several modifications were required to solve this problem. Filtering was done in the compensating or drive electronics to reduce command signal noise. Filtering in the actuator was also done.

Larger motor drive transistors were used so that the higher peak power transient levels could be tolerated in these transistors without failure. A deadband built into the actuator electronics before power was applied to the motor was increased to lower the susceptibility to noise. Increasing of the deadbands reduced the resolution capability of the actuator. It was determined on the analog simulation that the wider resolution tolerance was acceptable.

During the first thrust vector subsystem level testing on the spacecraft, severe coupling was discovered between the closed loop electronics and the actuator. The problem was a result of a poor impedance match between these two components. In order to rectify the problem, a load resistor was added across the output of the closed loop electronics.

Actuators were delivered to Boeing unpainted so that the solder sealed case could be visually inspected. CAT-A-LAC black paint was applied prior to spacecraft installation but after passing incoming inspection. A helium leak detector was used to check for leakage.

Following the above modifications to the original design, there were no significant failures during the environmental, reliability demonstration testing or qualification testing. There were no failures or performance degradation on the 10 flight units. Temperature rise during operation of the actuator was not a problem because the thermal mass was great enough to withstand all the power that the motor could take.

8.5.3.3 MISSION PERFORMANCE

Performance of the actuators was satisfactory during all missions. Operational life of the actuators is shown in Section 5.0. The operational flight time for the actuators was short because the actuator was operated only during the velocity maneuvers. Of significant importance is the fact that two of the thrust vector actuators on spacecraft L.O. II operated without detectable performance degradation after exposure to the space vacuum environment for 338 days. Data is presented in Section 5.0 for each axis and includes both ground and operating time. It is of interest to note that ground test time of flight components exceeded flight operating time by three orders of magnitude for the first and second flight spacecraft. The significantly higher ground test time on the first two spacecraft is attributed to special tests conducted because of earlier component and subsystem development problems, learning how by test personnel, and the refinement of test procedures.

8.5.4 CONCLUSIONS AND RECOMMENDATIONS

The conclusions reached from the study of the thrust vector actuator design are as follows:

1) The hermetically sealed electromechanical servo actuator design was adequate for this application.

2) The mechanical design of the actuator was far superior to the electronic design.

3) The short time period for the design, fabrication, and delivery of the first article contributed to the problems which had to be solved later.

Recommendations for "doing differently if one had it to do over" are:

1) The requirement for irreversibility during the boost vibration was unnecessary. This design would have been adequate with a smaller motor if a more efficient jackscrew had been used. At most the real requirement was to hold position from one burn to the next to avoid large start up transients.

2) The decision to use the gimbaled engine should have been made four months sooner. This would have provided time enough to work out development problems.

3) The initial transient conditions set the actuator rate requirements. Less severe conditions should have been specified for this design.

4) The components should have been tested with the correct electrical loading impedances before subsystem level tests were reached. A portion of the subsystem electrical noise problems could have been avoided.

5) Electrical noise on the signal lines should have been specified to the actuator vendor. This would have forced a more thorough analysis of the electronic design.

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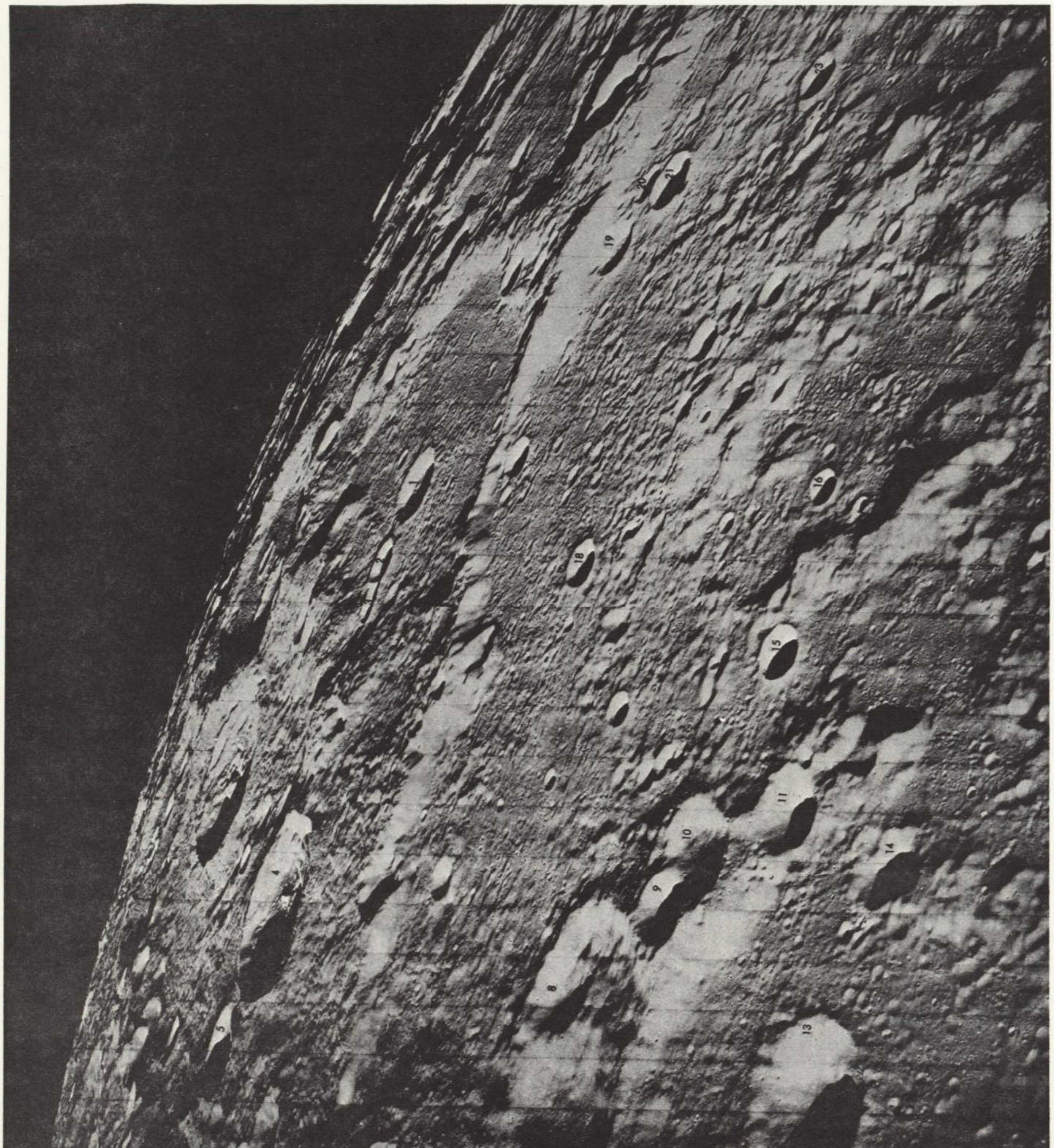

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