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Mariner Mars 1971 Attitude Control Subsystem

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PREFACE

The work described in this report was performed by the Guidance and Control Division of the Jet Propulsion Laboratory.

This report is an edited version of a series of engineering memorandums written between August 1970 and July 1971 on the Mariner Mars 1971 attitude control subsystem. The original intent was to provide a document on the Mariner spacecraft roughly parallel to that written by Will Turk in 1964 for the Ranger spacecraft. The report was to have been a comprehensive description of the Mariner spacecraft up through and including Mariner 1971.

This objective turned out to be much too ambitious and was abandoned quite soon in the preparation cycle in favor of a report that concentrated on the Mariner Mars 1971 design for the attitude control subsystem. The report contains a reasonably complete physical and functional description of the subsystem, including extensive coverage of the attitude control logic. Preflight design and design goals are emphasized, but references are made to reports that include postflight analysis and test results.

It is hoped that this report will serve as a useful reference for the Mariner spacecraft attitude control subsystem, in much the same manner that the Will Turk report has served for the Ranger spacecraft.

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Sun sensors: L. F. Schmidt.

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ABSTRACT

The Mariner Mars 1971 attitude control subsystem (ACS) is a three-axis stabilized system that evolved from the Ranger and early Mariner designs. It is comprised of a Sun sensor set, a Canopus tracker, an inertial reference unit, two Cold Gas Reaction Control Assemblies, two rocket engine gimbal actuators, and an attitude control electronics unit. The subsystem has the following eight operating modes: launch, Sun acquisition, roll search, celestial cruise, all-axes inertial, roll inertial, commanded turn, and thrust vector control.

In the celestial cruise mode, the position control is held to ±0.25 deg. Commanded turn rates are ±0.18 deg/s. The attitude control logic in conjunction with command inputs from other spacecraft subsystems establishes the ACS operating mode. The logic utilizes Sun and Canopus acquisition signals generated within the ACS to perform automatic mode switching so that dependence of ground control is minimized when operating in the Sun acquisition, roll search, and celestial cruise modes. The total ACS weight is 29.8 kg (65.7 lb), and includes 2.4 kg (5.4 lb) of nitrogen gas. Total power requirements vary from 9 W for the celestial cruise mode to 54 W for the commanded turn mode.

I. INTRODUCTION

The Mariners have served as vehicles for unmanned planetary observation throughout the last decade. Missions to Venus and Mars have established the value of the spacecraft both from the standpoint of scientific data returned and from the engineering experience gained. With a solid background of experience on which to build, future flights of the Mariner (and similar spacecraft) can be expected to reveal new and vital information about the nature and origin of man's universe, and in particular the planets of his solar system.

One of the functions required for every Mariner is attitude control. Attitude control is used throughout the mission to maintain the spacecraft in a preferred orientation relative to the Sun and the star Canopus. With the spacecraft in this orientation, a reference is provided for pointing the solar panels, the high- and low-gain antennas, and the scientific instrument package. Furthermore, proper temperature control of the spacecraft is obtained. Attitude control is most critical during the powered flight portions of the mission when trajectory corrections are made to insure proper planetary arrival accuracy and, in the case orbital missions, to insure that the spacecraft is inserted into proper planetary orbit.

The purpose of this report is to provide a description of the Mariner Mars 1971 attitude control subsystem (ACS). The evolution of the Mariner design is outlined in the next two subsections of Section I. Section II provides a general description of the ACS and its operating modes, and Section III gives a description of each assembly. The functional operation of the ACS is discussed more completely in Section IV where the detailed logic is presented. Appendix A gives a brief description of ACS telemetry, and Appendix B defines the spacecraft coordinate system.

A. SPACECRAFT EVOLUTION

The Mariner spacecraft have all been designed to take maximum advantage of past designs and flight experience. In this respect all of the designs have been conservative with the objective of obtaining cost and reliability benefits. Significant design changes have been made only to meet new mission requirements and, as a result, there is strong continuity between the various Mariner designs.

The Mariners 1 and 2 spacecraft were identical (Fig. 1). Their design was changed from that of the Ranger (Fig. 2) only to the extent required to accomplish a Venus flyby instead of a lunar impact. The ACS provided three-axis stabilization using attitude information from a set of Sun sensors and an Earth sensor, rate information from three single-degree-of-freedom rate integrating gyros, and control torques from Cold Gas Reaction Control Assemblies (RCAs).

Mariner 1 was launched by an Atlas/Agena Vehicle on July 22, 1962, but was destroyed by Range Safety because of an erroneous Atlas yaw-lift maneuver. Mariner 2 was launched on August 27, 1962 and completed its mission successfully.

Mariners 3 and 4 (Fig. 3) were designed specifically for missions to Mars. Their design was based on developed technology but was formulated so that future requirements of similar missions could be met with minimum changes. The most significant ACS change required for Mariners 3 and 4 was the replacement of the Earth sensor as the primary reference for roll control by a Canopus sensor. This change was necessary since the Earth-Sun geometry for a Mars mission was unfavorable for attitude control. In particular, the small Sun-spacecraft-Earth angle was unsuitable. The Sun was retained as one reference object but the star Canopus was selected as the most reasonable second reference object. Other changes included a repositioning of the cold gas thrusters from the spacecraft bus to the solar panel tips, and the addition of solar vanes to reduce the demands placed on the RCA for attitude stabilization. The solar vanes later proved to be unnecessary and were removed from subsequent designs to reduce ACS complexity.

On November 4, 1964 Mariner 3 was launched but could not achieve the desired orbit because the shroud failed to separate from the spacecraft. A new shroud was designed and on November 28, 1964, Mariner 4 was launched for a 7-1/2-month journey to Mars. The successful mission provided the first closeup pictures of this planet and provided significant new information about the technologies essential to interplanetary flight.

The Mariner 5 spacecraft (Fig. 4) was similar to the Mariner 4. The principal ACS change was the removal of the solar vanes. The spacecraft was launched on June 14, 1967, and accomplished a successful Venus flyby.

Mariners 6 and 7 (Fig. 5) were of the same basic design as Mariner 5. The principal ACS change was replacement of the Canopus sensor by a closed-loop Canopus tracker.

Mariner 6 was launched on February 24, 1969, and Mariner 7 on March 27, 1969. Both spacecraft accomplished successful flybys of Mars.

B. MARINER MARS 1971

Mariners 8 and 9 (Fig. 6) were the first to be designed for orbital missions. The large ΔV required for the Mars orbit insertion necessitated a completely new propulsion system from that used on previous Mariners. This in turn required that the thrust vector control portion of the ACS be redesigned. The jet vane actuator system used on previous Mariners was replaced with a gimbal actuator system for this purpose. The remainder of the ACS remained essentially unchanged.

Mariner 8 was launched on May 8, 1971, by an Atlas/Centaur vehicle, but failed to achieve orbit because of a Centaur control system failure. Mariner 9 was launched on May 30, 1971, and 167 days later became the first spacecraft to orbit another planet. On October 27, 1972, Mariner 9 ended a highly successful mission after having mapped the entire surface of Mars.

II. DESCRIPTION OF THE ATTITUDE CONTROL SUBSYSTEM

In this Section, the general composition of the ACS is described and a brief description of each major operating mode is given. This is followed by a tabulation of a few of the principal subsystem performance parameters and a weight and power summary.

A. GENERAL COMPOSITION

The ACS consists of the following assemblies: a Sun sensor set (SSS), a Canopus star tracker (CST), an inertial reference unit (IRU), an attitude control electronics unit (ACE), two gimbal actuators (GAA), and two redundant reaction control assemblies. These assemblies interface with each other and with other subsystems on the spacecraft as illustrated in Fig. 7. The assemblies are described briefly below and in more detail in Section III.

1. Sun Sensor Set

The Sun sensor set consists of acquisition Sun sensors, a cruise Sun sensor, and a Sun gate.

a. Acquisition Sun Sensors. The acquisition Sun sensors consist of four identical units. One unit is mounted on each of the four solar panel tips. The acquisition Sun sensors provide pitch and yaw control signals. They have a combined 4π steradian field of view (FOV).

b. Cruise Sun Sensor. The cruise Sun sensor consists of a single unit mounted on the solar panel outrigger. The cruise Sun sensor provides pitch and yaw axis control signals. The cruise Sun sensor has a FOV that exceeds that of the Sun gate.

c. Sun Gate. The Sun gate consists of a single unit with redundant detectors mounted on the solar panel outrigger. The signal generated by the Sun gate is used to identify a Sun-acquired state.

2. Canopus Tracker

The Canopus tracker consists of a single nonredundant unit mounted on the spacecraft bus. The Canopus tracker provides a roll-axis control signal and contains the necessary logic for star identification.

3. Inertial Reference Unit

The IRU provides rate and position signals in pitch, yaw, and roll. The IRU consists of an inertial sensor subassembly (ISS) and an inertial electronics subassembly (IES).

a. Inertial Sensor Subassembly. The ISS contains three single-axis, rate integrating gyros and one pendulous rebalanced accelerometer.

b. Inertial Electronics Subassembly. The IES provides gyro and accelerometer control loops for the ISS. Position information is generated in the IES by integrating the gyro rate.

4. Reaction Control Assembly

The ACS has two identical RCAs. In normal operation, the RCAs provide torque couples about each axis. In case of a failure of either assembly, the other assembly provides bidirectional torque about each axis. Each assembly consists of a high-pressure storage vessel, a pressure reducing regulator, a low-pressure distribution system, and six jet valves.

5. Gimbal Actuator Assembly

The ACS has two identical nonredundant gimbal actuators used to rotate the propulsion subsystem rocket engine about the pitch and yaw axes. Each actuator contains a dc motor and a shaft position feedback transducer.

6. Attitude Control Electronics

The ACE consists of the RCA electronics, thrust vector control (TVC) electronics and associated control logic.

a. RCA Electronics. The RCA electronics control the operation of the jet valves and consist of a deadband circuit, minimum on-time circuit, jet valve drivers, and a derived rate circuit.

b. TVC Electronics. The TVC electronics control the operation of the gimbal actuators and consist of a preaim circuit, gimbal servo electronics and a compensator circuit.

c. Control Logic. The ACE receives command inputs from the command computer and sequencer, and from the flight command subsystem. These inputs in conjunction with an enable signal from the pyrotechnic subsystem and internally-generated logic levels provide direct control over all ACS operating modes.

B. OPERATING MODES

In all operating modes except launch and TVC, the RCA provides control torques for all three axes.

1. Launch Mode

During launch, the ACS is essentially passive, its primary function being to survive the launch environment.

2. Sun Acquisition Mode

At spacecraft separation from the launch vehicle, a signal from the pyrotechnic subsystem places the ACS in the Sun acquisition mode. In the Sun acquisition mode, the acquisition Sun sensors and the IRU rate signals cause the -Z axis of the spacecraft to be rotated into Sun alignment. When the Sunline falls within the FOV of the Sun gate, the Sun gate circuitry issues a signal that identifies a Sun-acquired condition. The acquisition Sun sensor inputs to the RCA electronics are then disabled, and the cruise Sun sensor inputs are used exclusively. Meanwhile, the roll control channel, driven by the roll rate signal from the IRU, reduces the roll rates to within the rate deadband.

3. <u>Roll Search Mode</u>

Following Sun acquisition, the ACS automatically initiates roll search when the Canopus tracker is powered. Saturated output from the Canopus tracker is summed with the IRU roll rate to initiate a roll search for Canopus. Stars other than Canopus are discriminated against on the basis of intensity and cone angle in the Canopus tracker acquisition logic. When a star satisfying the intensity gate logic is detected, the ACS terminates the roll search and locks onto the star.

4. <u>Celestial Cruise Mode</u>

In the cruise mode, the ACS maintains three-axis stabilization using the Sun and Canopus as reference objects. The derived rate circuit provides angular rate stabilization about all three axes in this mode.

5. <u>All-Axes Inertial Mode</u>

In the all-axes inertial mode, IRU rate and position error signals are used in the pitch, yaw, and roll channels.

6. <u>Roll Inertial Mode</u>

In the roll inertial mode the roll channel control signals consist of the IRU roll position and rate outputs. Pitch and yaw axis control signals are provided by the cruise Sun sensors and the IRU pitch and yaw rate outputs.

7. <u>Commanded Turn</u>

Commanded turns are performed with the ACS in the all-axes inertial mode. A turn bias is inputted to either the roll or yaw axis IRU rate integrator. The IRU output causes the spacecraft to rotate through the desired angle about the selected axis.

8. <u>Thrust Vector Control Mode</u>

The TVC mode is initiated simultaneously with engine burn. The ACS utilizes IRU rate and position outputs in all three axes for control signals.

Orbiter attitude control torques are provided in pitch and yaw by controlling the gimbal orientation of the engine, and in roll by the RCA. The accelerometer in the IRU provides velocity pulses to the command computer and sequencer (CC&S).

C. PERFORMANCE PARAMETERS

ACS position deadbands and turn rates are given in this section. Additional parameters are given later for each of the assemblies. Table 1 lists the position deadbands. These are the positions at which the jet valves are actuated in the absence of rate inputs. Turn rates are given in Table 2.

D. WEIGHT AND POWER SUMMARY

Table 3 summarizes the weights of the various ACS assemblies.

The power subsystem supplies the ACS with 2.4-KHz, 50-Vrms transformer isolated power, 30-Vdc regulated power, and 3-phase, 400-Hz, 26-Vrms phase-to-phase power. The Canopus tracker and inertial reference unit each has its own power conditioning equipment. The ACE contains the remaining power conditioning equipment. Table 4 presents typical average power requirements for the major operating modes. Transient conditions are not considered (power demands existing for less than about 1 s).

The Canopus tracker demand is relatively constant throughout the mission, but Sun shutter operation requires an additional 6 W if it is activated.

The IRU power depends upon whether one or three gyros are being operated. Average power for one gyro is on the order of 2.7 W with a power factor of 0.56 (inductive).[1] The gyros operate from a 400-Hz, 3-phase inverter. During the commanded turn mode the gyro electronics requirements increase from 13 to 17 W to provide gyro torquing current. The accelerometer is powered in conjunction with the pitch and yaw gyro, and hence is on when all three gyros are on. Approximately 2 W is required for the accelerometer and associated electronics.

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[1] Power factors for all other equipment are between 0.95 and 1.0.

The ACS uses 2 gimbal actuators. Stall power is 50 W each (for about 10 ms at power turn on). Typical slewing power is 10 W. Slewing is required for about 1 s at initial turn on so that the gimbals can attain their bias position. Typical limit cycle operation requires 2 to 3 W per actuator.

The ACE power requirements vary considerably as a function of mode of operation. Power required for the Sun sensors (about 1/4 W) and the RCAs is included as a part of the ACE power. The jet valves require about 2.4 W each and normally two valves fire together as a torque couple. However, when the spacecraft reaches a limit cycle state the valves pulse only every 10 min or so for 20 ms. Hence power demand during the limit cycle operation is negligible. During spacecraft rate reduction, however, as many as 6 valves may fire simultaneously. During motor burn 2 valves (in the roll axis) may be on as much as 50% of the time for 15 min.

III. DESCRIPTION OF ACS ASSEMBLIES

In this Section, a description of each of the ACS assemblies is given. A physical and functional description for each is followed by a listing of some of the more important performance parameters.

A. SUN SENSOR SET

The Sun sensor set consists of a number of photoconductive cells and associated mounting surfaces. The SSS provides the sensing function required for pitch and yaw attitude control of the spacecraft. Three types of sensors are used; acquisition sensors, cruise sensors, and Sun gate.

When the spacecraft is operating in the acquisition mode the acquisition and cruise sensors operate together to bring the spacecraft into the desired orientation relative to the Sun. The Sun gate provides an output that is converted into a logical signal to mark the completion of Sun acquisition. The acquisition sensors are then disabled and the cruise sensors supply the information required to maintain the proper spacecraft-Sun orientation throughout the cruise portion of the mission.

1. Physical Description

The Sun sensors all use cadmium sulfide detectors that operate as light sensitive resistors. A total of 12 acquisition detectors are employed, 6 each for pitch and yaw axis control. They are mounted so that regardless of spacecraft attitude the Sun will illuminate at least one of them. Four cruise detectors are used, two each for pitch and yaw axis control. They are mounted with a more restricted field of view and the Sun illuminates them only when the spacecraft has been brought into approximately the correct attitude. The Sun gate has the narrowest field of view and is fully illuminated only when the roll axis of the spacecraft has been brought within approximately five degrees of the Sun line.

The mechanical configuration of the sensors is shown in Fig. 8. The four cruise detectors are all mounted on the same block, whereas the acquisition detectors are mounted in groups of three. The Sun gate is mounted as a redundant pair. Figure 9 illustrates the manner in which the detectors are mounted to reduce their susceptibility to failure during launch vibration. All sensors were built by Honeywell, Inc., to JPL specifications.

The total weight of the Sun sensors is 0.45 kg (1 lb). They utilize dc power converted from 50-Vrms, 2.4-kHz power and require about 0.25 W.

2. Functional Description

The cruise sensor uses the balanced bridge detector principle (see Fig. 10). Two cadmium sulfide photoconductors are paired and precisely located beneath bars that partially shadow their sensitive elements from the Sun. A detector fully illuminated at 90720 lm/m^2 (8400 ft-cd) (2/3 solar constant) has a resistance of approximately 750 Ω. When fully shadowed, the resistance exceeds 20 MΩ. Direct current excitation is applied to the detector pair and produces a zero sensor output voltage when the Sun is directly on the zero reference line. Moving the sensor off the reference line results in a differential action of the shadows that unbalances the illuminated areas of the detectors and results in an output voltage.

The functional diagram for the acquisition mode is illustrated for one axis in Fig. 11. The acquisition sensors are disabled whenever the spacecraft is aligned within approximately 5 deg of the Sun. This action prevents degradation of the cruise sensor output by the acquisition sensors, which could introduce null errors from their less precise outputs.

The Sun gate uses two detectors identical to those used for the acquisition and cruise sensors. They are not connected in a bridge circuit, however, and function as variable resistors only. The detectors are connected in parallel to increase Sun gate reliability. The Sun gate has a conical field of view that is centered on the spacecraft roll axis. As the spacecraft moves to acquire the Sun, the Sun gate detectors begin to be illuminated. The detector resistance decreases sharply until about 5-deg offset. At this point a level detector circuit in the ACE generates a logic signal indicating that the Sun has been acquired.

3. Performance Parameters

The nominal input-output characteristics of the combined cruise and acquisition sensors as measured at the input to the ACE switching amplifier is shown in Fig. 12. Figure 13 shows the input-output characteristics of the cruise sensors only. Both figures are for a single axis only, pitch and yaw axes being identical. The nominal input-output characteristic of the sun gate is given in Fig. 14.

The cruise sensors are the only ones having critical design parameters, since it is the cruise sensors alone that determine performance capabilities in the celestial cruise mode. Table 5 lists some of the more significant parameters.

B. CANOPUS TRACKER

The Canopus tracker provides position information for the roll attitude control loop. It consists of an image dissector tube with a photocathode surface and associated optics and electronics.

The Canopus tracker performs two primary functions. It makes an identification decision on each star that enters the field of view during roll search, and provides to the attitude control electronics a signal proportional to the roll error angle when a star is identified as Canopus. The identification process consists of measuring the star intensity and comparing it with a previously calibrated value. Error angle information is obtained by repetitively scanning a slit field of view across the image field and then measuring the modulation phase of any star signal that appears.

1. Physical Description

A photograph of the complete Canopus Tracker Assembly (including the outer baffle assembly) is shown in Fig. 15. The heart of the assembly is the sensor, consisting of an image dissector and associated optics (see Fig. 16). The image dissector or scanning photomultiplier tube is an electro-optical-mechanical package consisting of a cesium-antimony photocathode, an electrostatic electron optical image section, a Schlesinger electrostatic deflection cone, and a conventional electron multiplier, all combined into one vacuum tube.

The sensor is protected from exposure to the Sun by a Sun shutter that is activated by a Sun detector mounted on the Canopus tracker (see Fig. 17). Baffling is provided to minimize effects of stray light from Earth and Mars, which would have a detrimental effect on tracking capability.

The Canopus Tracker Assembly weighs 4. 2 kg (9. 3 lb) and operates on 50 V rms, 2. 4 kHz power. Nominal power requirements are 5 W with an additional 6 W required to activate the Sun shutter. Since Sun shutter activation is a rare event, the average power required by the tracker is about 5 W. The exterior dimensions for the package containing the image dissector, optics, electronics, and inner baffle are 29. 2 × 12. 7 × 10. 9 cm (11. 5 × 5. 0 × 4. 3 in.). The exterior dimensions for the outer baffle are 31. 8 × 18. 3 × 13. 5 cm (12. 5 × 7. 2 × 5. 3 in.). The Canopus tracker was built by Honeywell, Inc., to JPL specifications.

2. Functional Description

Three different fields of view are defined for the Canopus tracker in Figs. 18 and 19. A simplified block diagram for the Canopus tracker is given in Fig. 20. The basic operation of the tracker is described as follows:

The optical field of view is defined by the Tracker Lens Assembly and image dissector. The image of any bright object within the optical field of view is formed by the objective lens on the image dissector tube photocathode. The emitted photoelectrons are accelerated and imaged by the focus potentials onto a conducting grounded aperture plate that separates the focusing section from the electron multiplier section. A slit in the aperature plate allows a certain portion of these electrons through to the electron multiplier. The portion of the optical field of view that is imaged through the slit to the electron multiplier is called the instantaneous field of view. The size of the instantaneous field of view is defined by the electron aperture dimensions and the tracker focal length.

A saw-toothed voltage, supplied by a sweep generator, is applied to the electron roll angle deflection plates in the tube imaging section and causes the instantaneous field of view to be scanned over a portion of the optical field of view. The resulting field of view is a function of the scan amplitude and is called the scanned field of view. It can be biased to any position between the clock limits of the optical field of view and stepped to any one of five discrete positions between the cone limits of the optical field of view. If a star is within the scanned field of view, the scanning action of the instantaneous field of view modulates the resulting electron beam, which is then amplified in the electron multiplier section.

The output of the sensor is conditioned and amplified by an ac preamplifier. The preamplifier output is used to generate a star intensity signal and a roll error signal.

The intensity signal is utilized for two purposes; for star identification and for gain control in the image dissector. The first stages of the intensity loops modulate the preamplifier output into two channels 90-deg separate in phase. The two channels are then peak detected and combined in a summing circuit. This produces a dc output proportional to the radiant source intensity that is relatively free of background noise. The last stage of the intensity circuits is a high-voltage supply, the output of which is inversely proportional to the preamplifier output. This signal is input to the logic circuit for star identification and is also fed back to the image dissector to provide automatic gain control.

When the logic determines that Canopus has been acquired, the demodulator is enabled and the star is automatically tracked, that is, the star image is automatically centered in the scanned field of view. This is accomplished by feeding back a dc signal to the roll deflection plates that is proportional to the angular displacement of the star from the null plane of the tracker. This signal is obtained by first demodulating and filtering the preamplifier output, integrating it to obtain a dc roll error signal, and then amplifying it for input to the roll deflection plates of the image dissector. The integrator output is also supplied to the ACE as the roll error signal.

The cone angle generator and bias circuits are used for field of view control and are discussed below. The sweep generator, in addition to generating the scanned field of view, provides timing and clock signals to various circuits. The power supplies provide all the ac and dc voltages that the tracker requires.

3. Field of View Bias Control

The Canopus tracker has two degrees of freedom: cone and clock. The cone angle of the scanned field of view can be changed in discrete increments only. The clock angle of the scanned field of view is controlled by three separate fixed biases in addition to the variable tracking bias, which can be applied by the demodulator. Two of the fixed biases, the flyback bias and the roll override bias are hard biases in that if they are applied they will predominate over both the demodulator tracking bias and the search bias. The search bias is a soft bias, in that it can be overridden by either of the hard biases or by the tracking bias.

The motion of the instantaneous field of view within the scanned field of view is controlled by the sweep generator mentioned before. In the discussion that follows, it should be recalled that the motion of the instantaneous field of view defines the scanned field of view, and that the former always lies within the latter.

a. Cone Angle Control. Canopus is located about 15 deg from the south ecliptic pole. As the Sun-pointing spacecraft orbits about the Sun, the star field appears to rotate about the ecliptic pole and the Sun-spacecraft-Canopus angle (cone angle) varies sinusoidally with a peak-to-peak excursion of 30 deg. To accommodate this apparent motion of Canopus, the tracker is electronically gimballed so that the scanned field of view can be biased to any of five discrete cone positions within the optical field of view. Figure 21 shows the cone angle motion of Canopus as it occurred for the Mariner 9 mission.

b. Tracking Bias. In the absence of either of the two hard biases, the demodulator tracking bias controls the position of the scanned field of view (in clock). The tracking bias is applied when a star is being tracked and causes the star image to be centered in the scanned field of view.

c. Search Bias. During roll search for Canopus, the motion of the spacecraft is such that the star field appears to move from the plus clock to the minus clock edge of the optical field of view. During this time, a search bias is applied at the input of the Canopus tracker integrator that causes the scanned field of view to be positioned against the plus clock edge of the optical field of view.

Figure 22 shows the motion of the instantaneous field of view within the optical field of view during a typical roll search and acquisition sequence. The time varying position of Canopus shown in this figure is the result of spacecraft motion, this motion being controlled by the combined effects of the Canopus tracker roll error signal, the gyro rate signal, and the RCA torques.

d. Roll Override Bias. If a star is acquired that for some reason is not Canopus, or if for some reason ground control personnel decide to break lock on any star being tracked, then a roll override bias can be applied to the input of the integrator. This will cause the scanned field of view to be positioned against the plus clock edge of the optical field of view and the spacecraft will automatically begin a roll search for a new star.

e. Flyback Bias. Under certain conditions it may be possible to acquire Canopus without going through a roll search. If, for example, Canopus is momentarily lost, it may be possible to reacquire by sweeping the scanned field of view across the optical field of view. This is accomplished by applying a flyback bias to the input of the integrator. This causes the scanned field of view to be positioned against the minus clock edge of the optical field of view. Removal of the flyback bias and application of the search bias will then cause the scanned field of view to sweep towards the plus clock edge of the optical field of view. The flyback and sweep is automatically performed whenever Canopus is lost.

Figure 23 illustrates the motion of the instantaneous field of view during a typical flyback and sweep. Here it is assumed that Canopus track was lost because of a bright particle traveling through the optical field of view.

4. Star Identification

a. Star Intensity Gate Settings. Canopus has an illumination intensity about one-half that of the only brighter star, Sirius. Similarly, the next two dimmer stars, Vega and Rigel, are about one-half the intensity of Canopus. This illumination intensity separation is the basis for identifying the star in flight. Two star intensity gates are used for this purpose: a lower and an upper gate. When a star having an intensity between these gate settings is encountered, it is identified as Canopus. Once the spacecraft has acquired the Sun, the brightest star in the field of view of the Canopus tracker is Canopus itself. Hence, the primary purpose of the high-gate setting is to prevent the tracker from locking onto a planet. The low gate is utilized to discriminate against stars dimmer than Canopus.

The Canopus tracker utilizes an adaptive low gate having three possible level settings, progressively lower, numbered 1, 2, and 3. The adaptive gate feature was first implemented on Mariners 6 and 7. The intent is to provide an automatic safeguard against rejecting Canopus if for some reason the intensity signal generated by the tracker electronics falls below the nominal (number 1) low gate.[2]

The low-gate levels are as follows: level 1, 0.7 times Canopus, level 2, 0.35 times Canopus; level 3, 0.2 times Canopus.

A fixed high gate of 3.5 times Canopus is used. However, once Canopus has been acquired, the high gate is automatically removed. The purpose of removing the high gate after acquisition is to avoid a problem experienced on the Mariner 4 mission. On this mission, there were several instances when bright particles entered the field of view of the Canopus tracker after Canopus acquisition had been established. This resulted in violation of the high gate, and subsequent loss of Canopus acquisition. The Mariner 9 ACS is mechanized to track bright particles until they disappear from the field of view[3] and then to perform a flyback and sweep to reacquire Canopus.

[2] The adaptive low-gate levels 2 and 3 were not required on the Mariner 6, 7, or 9 missions.

[3] Normally, bright particles have high velocities and remain in the field of view only momentarily.

b. Canopus Acquired Signal. A Canopus acquired signal is generated in the tracker for use in the ACE. This signal is normally generated only when a star falls within the intensity gate settings. As mentioned before, however, once Canopus is acquired the high gate is effectively removed and the Canopus acquired signal will remain on so long as the low-gate setting is not violated. If the star intensity signal drops below the low gate, the Canopus acquired signal will remain on for 7 s, during which time a flyback and sweep is performed. If Canopus is not reacquired during this 7-s interval, the Canopus acquired signal will be turned off and the ACS will go into a roll search.

c. Typical Roll Search Pattern. Figure 24 illustrates the celestial geometry during a typical roll search for Canopus.[4] Figure 25 is the corresponding intensity plot generated by the Canopus tracker. The high and low gates marked on this plot are those of Mariner 4. Subsequent to the Mariner 4 mission the stray-light baffling of the tracker was improved, the high-gate setting was reduced to 3.5 times Canopus, and an adaptive low-gate implemented as described previously.

5. Performance Parameters

A typical output characteristic for the Canopus tracker as a function of the roll error angle is presented in Fig. 26. The roll error signal is supplied continuously to the ACE. It should be recalled, however, that due to the limited field of view of the tracker the output is a result of a star signal only when the roll error is less than approximately 5 deg. For roll errors larger than this Canopus lies outside the field of view and the output is provided by the fixed search bias discussed previously. The output error is linear between ±3.5 deg roll offset.

Table 6 lists some of the more important performance parameters for the Canopus tracker roll error signal. Characteristics of the Canopus acquired signal are given in Table 7.

[4] Data for Figs. 24 and 25 were taken from the Mariner 4 mission.

C. INERTIAL REFERENCE UNIT

The inertial reference unit (IRU) consists of three single-degree-of-freedom gyroscopes, one single-axis accelerometer, and associated electronics. The purpose of the IRU is to provide angular rate and position information when the celestial references are not acquired, and linear velocity data during motor burns.

1. Physical Description

The IRU is housed in two subassemblies: the inertial sensors subassembly (see Figs. 27, 28, and 29) and the inertial electronics subassembly (see Fig. 30). The sensors consist of three single-axis floated rate integrating gyroscopes, and one single-axis pulse captured pendulous accelerometer. Temperature control of the inertial instruments is used only for ground testing and they operate in spacecraft ambient temperature (typically 286 to 311 K (55° to 100°F)) during flight and all spacecraft level testing. No heater power is required for any of the inertial components.

During certain modes of operation gyro rate information is integrated to obtain angular position change. An electronic integrator is used for this purpose. This circuitry is sensitive to power supply variations, electrical pickup, and local temperature variations, and requires electrostatic shielding and several selected resistance values to reduce drifts and offsets.

Power supplied to the IRU consists of 27-V rms, 400-Hz, 3-phase power for the gyro spin motors and 50-V rms, 2.4-kHz power for other equipment. The 2.4-kHz input is conditioned prior to use. Power required for each gyro is nominally 2.7 W at a 0.56 power factor (lagging). Power required for the electronics is nominally 13 W with an additional 4 W required during commanded turns for torquing the gyros.

Each subassembly weighs approximately 2.3 kg (5 lb). Exterior dimensions are 9.4 × 15.2 × 16.8 cm (3.7 × 6.0 × 6.6 in.) for the inertial sensors subassembly and 5.3 × 16.8 × 35.6 cm (2.1 × 6.6 × 14.0 in.) for the inertial electronics subassembly.

The inertial sensors were built by Kearfott, a Division of Singer-General Precision, Inc. The subassemblies were manufactured and tested by General Electric to JPL specifications.

2. Functional Description

A simplified block diagram for the IRU is given in Fig. 31. Angular rate inputs are sensed by the gyros. The gyro pickoff outputs are amplified, demodulated, servo-response compensated, and returned to their respective torquers through a series resistor to ground (see Fig. 32). The voltage across the precision resistor is proportional to torquer current and hence input rate. Angular position information is obtained by integrating this rate signal.

Linear acceleration inputs along the roll axis during motor burn are sensed by the accelerometer. The accelerometer pickoff output is amplified, demodulated and servo compensated (see Fig. 33). The output is then compared to a reference voltage by a comparator circuit. If the output exceeds the reference voltage, a digital gate is enabled that allows a square wave pulse of precision width to drive a digital switch. The switch output is a pulse of predetermined width and amplitude and corresponds to a known velocity increment. The number of pulses represents the change in spacecraft velocity due to a motor burn. The actual pulse counting is accomplished in the central computer and sequencer (CC&S). A tight analog loop is utilized during launch to reduce the probability of accelerometer damage in the launch environment.

The IRU operates in four principal modes: rate only, all-axes inertial, roll inertial, and commanded turn. Logic inputs to the IRU from the ACE control the switches within the IRU, and determine which of the above modes is to be utilized.

The rate only mode is utilized during Sun and Canopus acquisition. Rate information from all three gyros is supplied to the ACE during these times for damping of spacecraft rates.

The all-axes inertial mode is utilized just prior to the commanded turn mode, and during motor burn. During these times the integrators are enabled and rate plus angular position is supplied to the ACE attitude control electronics to maintain a constant spacecraft attitude in all three axes.

The commanded turn mode is identical to the inertial mode except that a bias is added to the input of either the roll or yaw integrator. This causes the spacecraft to rotate at a prescribed constant rate about the selected axis for as long as the bias is applied.

The roll inertial mode is used when position information from the Canopus tracker is not available or when it is unreliable. For this mode the roll gyro only is turned on and the roll axis operates in the inertial mode described above.

3. Performance Parameters

a. Gyro Parameters. The dynamic characteristics of the gyro loops may be approximated by a linear second order model with a natural frequency (ω) of 88 rad/s and damping factor (ζ) on the order of 0. 35. Figure 34 shows a typical measured response of a single-axis gyro loop to a step input compared with an ideal linear second order response.

The actual gyro response is nonlinear as a result of pivot and jewel clearance effects. This nonlinearity results in the effective damping factor mentioned above. A linear analysis that neglects the nonlinearity and computes the damping factor from the average value of viscous damping results in the unrealistic low estimate of the damping factor (on the order of 0. 04).

Maximum linear slew rates for the gyro loops before electrical saturation are 1. 6 deg/s in roll and 3. 2 deg/s in pitch and yaw.

During the commanded turn mode a precision bias input to the roll integrator results in a nominal 650- deg/h turn rate. A precisely calibrated turn rate as a function of gyro temperature is required since the desired turn angles are obtained by timing the commanded turn about the roll and yaw axis.

The turn rates are calibrated prior to launch at four gyro temperatures: 303, 311, 319, and 328 K (85, 100, 115, and 130° F.) In flight, bay temperature measurements are then made to determine the appropriate calibration of the inertial instruments. Typically the turn rates vary by 4 deg/h over the above temperature range.

The total uncertainty for the change in spacecraft angular position as a result of IRU errors is given in Table 8 for the commanded turn and inertial modes. Two figures are given for each mode: one is the nominal day-to-day uncertainty seen during ground testing, the other is an estimate of the uncertainty during the mission itself.

b. Accelerometer Parameters. Accelerometer parameters are given in Table 9.

The accelerometer, like the gyros, requires that the instrument temperature be taken into account to arrive at the correct calibration value. Typical scale factor variations over the expected temperature range of 292 to 317 K (65 to 110°F) are 0.2%. In addition, scale factor changes result as a function of torquer aging and acceleration magnitude. Both of these effects are predictable and are taken into account when computing velocity change.

D. REACTION CONTROL ASSEMBLY

The Reaction Control Assemblies (RCAs), also called the cold gas system, provide the actuating torques required for spacecraft attitude control. The system is comprised of two identical assemblies, each having an independent gas supply. The required actuating torque is produced by exhausting controlled amounts of nitrogen gas through jet valves mounted on the solar panel tips. Each valve in an assembly has a companion in the other assembly that provides a pure torque couple.

1. Physical Description

Each of the two identical assemblies consists of a high-pressure storage vessel, a pressure reducing regulator, a low-pressure distribution system, and six jet valves. One of the assemblies is shown mounted on a handling fixture in Fig. 35.

The high-pressure vessel has an outer diameter of 23.4 cm (9.2 in.) and holds 1.2 kg (2.7 lb) of gas at 18.9×10^6 N/m^2 (2740 psig) and 38°C (100°F). The spherical titanium vessel is comprised of two hemispheres joined by a uniweld technique.

The pressure regulator is a single-stage unit that regulates to $104 \times 10^3 \pm 8.3 \times 10^3$ N/m^2 (15 ± 1.2 psig) over an inlet range of 0.69×10^6 to 24.2×10^6 N/m^2 (100 to 3500 psig). The control valve is a 0.157-cm- (0.062-in.-) diam precision tungsten carbide ball that is activated by direct linkage to the sensing diaphragm.

Six jet valves are used for each assembly. The valves are mounted on manifolds to form either a four-valve unit or a two-valve unit. A four-valve unit with test nozzles in place is shown in Fig. 36. The flight nozzle orifices are sized to provide precise thrust levels and are not mounted to the valve until immediately prior to launch (Fig. 37). The steady-state thrust levels per valve are 0.033 N (7.4 mlb) for pitch and yaw and 0.074 N (16.6 mlb) for roll.

The valves are actuated by the ACE that supplies a nominal command excitation of 26 Vdc. Power required to actuate a valve is approximately 2.5 W. No other power is required by the Reaction Control Assembly. Total weight of each assembly is 11.8 kg (26 lb) plus an additional 2.7 kg (6 lb) of gas.

2. Functional Description

The functional operation of the RCA follows. Gas stored in the high-pressure vessels is reduced in pressure by the regulators and supplied to the jet valves via a low-pressure distribution system. The valves are actuated in pairs by the ACE, which outputs electrical pulses of 20-ms duration.

The frequency of gas pulses is dependent on the ACS mode of operation. After spacecraft separation, the jet valves are on continuously until the initial spacecraft rates have been reduced to the desired levels. Thereafter, except for commanded rate changes and during motor burn, the jet valves are pulsed only enough to maintain the desired spacecraft rate. To maintain a constant nonzero rate as in roll search, or a near-zero rate as in celestial cruise, the jets for any one axis need be pulsed only once every 10 to 20 min. During motor burns the pitch and yaw jet valves are not used since control torques are provided by the gimballed thrust motor. The roll axis valves, however, may be on as much as 50% of the time to overcome swirl torques produced by the motor.

Loss of control torques would be catastrophic since spacecraft attitude control is essential to mission success. For this reason the gas system is designed to operate even in the event of complete failure of one assembly. One worst-case failure mode would be a jet valve that stuck open. If this were to happen, one of the high-pressure vessels would be completely depleted, but the other would retain 2/3 of its gas supply. This amount of gas would be enough to successfully complete the mission. Also the roll control torque of one assembly would be adequate to counteract swirl torques during motor burn.

3. Contamination and Leakage Control

The most significant problem associated with the successful operation of the RCA is contamination control. Contamination is of concern because of the requirement to maintain exceptionally low leakage rates. The primary source of gas leakage is imperfect seating of the jet valve ball (see Fig. 38). Any foreign particles that come into contact with the ball seat have the potential of damaging the seat and increasing leakage. Worse yet, if the particle were large enough and became lodged between the ball and the seat, the valve would remain open and result in the loss of all gas in the assembly associated with that valve.

To reduce the probability of damage to the ball seat, and the attendant problem of increased gas leakage, several precautions are taken. First, the system is designed to minimize the potential for contamination; second, precautions are taken during assembly; finally, a fill manifold is used to prevent contaminated gas from being placed into the system.

One example of the design precautions used is the exclusive use of welded joints, since contamination is more difficult to control with mechanical joints. Assembly of individual components of the system is accomplished wherever possible in a 100 Class clean room, whereas final assembly and welding is done in a 10, 000 Class clean room. Extreme caution is used in filling the system since contamination is virtually impossible to remove once assembly has been completed.

The fill manifold used contains a filter that removes particles down to 1.26×10^{-8} m (0.5-μin.) diam. It is removed from the system prior to flight. No filters of any kind are flown with the system. The principal reason for not flying filters is to avoid the possibility of the filter itself acting as a particle generator when exposed to launch vibrations.

Nominal leakage rates for each assembly are on the order of 16 to 20 cm^3/h, and are caused almost entirely by valve leakage. Leakage rates of each thrust valve are typically between 2 and 3 cm^3/h. To maintain these low rates, valve seat damage must be prevented and for this reason the diameter of foreign particles in the system is specified to be less than 2×10^{-6} cm (5 μin.) metallic and 10×10^{-6} cm (25 μin.) nonmetallic.

4. Gas Consumption

Table 10 presents a preflight estimate of the gas requirements for attitude control. The estimates are the result of computer simulations that take into account a number of variables affecting gas consumption. Estimates are given for the 12-h orbit mission flown by Mariner 9. Two separate calculations were made. The first calculation is based on the assumption that both assemblies were operating properly (full-gas system); the second assumes that one of the assemblies failed during launch (half-gas system).

Two of the principal sources of gas consumption are leakage and scan platform stepping. A leakage rate of 3 cm^3/h was assumed per valve. Motion of the scan platform during orbit is required to keep the science instruments pointing toward the planet. This motion introduces transients that require additional cold gas torquing.

Gas consumption during cruise limit cycling is the next largest source. Estimates were based on the presence of a one-sided disturbance torque of 50×10^{-7} N-m (50 dyne-cm) about pitch and yaw, and 1×10^{-6} N-m (10 dyne-cm) about roll.

For the discrete events, the following assumptions were made: initial spacecraft rates of 0.2 deg/s about each axis[5]; a total motor burn time of

[5] These rates are typical for the Centaur launch vehicle.

20 min with swirl torques of 0.011 N-m (0.1 in.-lb)[6]; spacecraft rates during Sun acquisition, and roll search, of 3.5 mrad/s; commanded turn rates of 3.14 mrad/s.

To assure successful completion of a mission with a worst case failure (one valve stuck open immediately following launch) the gas system was sized to have three times the gas required for a half gas system. This resulted in a requirement for 2.4 kg (5.3 lb) of gas for the Mariner 9 mission. Thus the 2.5-kg (5.4-lb) capacity of the system was adequate to sustain a worst case failure.

5. Performance Parameters

The input/output characteristics for a typical thrust valve are shown in Fig. 39. The command excitation is supplied by the ACE. The valve opens between 6 and 12 ms after voltage is applied to the valve. This is the time it takes to establish a current in the coil sufficient to actuate the valve. When the valve opens, the chamber pressure builds up to its steady state value — typically 1.7×10^3 to 6.9×10^3 N/m^2 (0.25 to 2 psi) below the inlet pressure of 10×10^4 N/m^2 (15 psi) depending primarily on the nozzle diameter. Nozzle thrust is directly proportional to chamber pressure. When the command excitation is removed, there is a closing delay due to mechanical inertia, inductance in the valve, and the effects of a diode in the ACE. The diode is used to limit spiking of the input voltage and thereby protect the ACE from excessive transients. Closing time is typically 10 ms with the diode in place, and 1.5 to 2 ms without the diode. Once the valve closes, there is a residual amount of gas in the chamber that gradually bleeds off reducing the thrust to zero.[7]

The torque levels selected for the gas system influence the operation of the ACS in all of its modes of operation and must be chosen so as to give desirable performance characteristics in the most critical modes. The minimum level of torque in the roll axis is required to be large enough to control swirl torques during motor burn with only one of the two assemblies

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[6] The roll nozzles are sized for a 0.23 N-m (2 in.-lb) worst case swirl torque.

[7] The above parameters are based on preflight test data.

operating. The torque levels in the pitch and yaw axes are selected to give satisfactory commanded turn performance, good limit cycle behavior in the celestial cruise mode, and adequate control over disturbance torques. Torques are not increased unnecessarily, however, because increased torques tend to increase gas consumption, particularly in the celestial cruise mode where torque requirements are at a minimum.

For the pitch and yaw axes, the torque requirements are not specified directly, but rather the torque is selected to give the desired angular acceleration. The torque (T) and angular acceleration (α) are related quite simply by

$$T = I\alpha$$

where I is the spacecraft moment of inertia about the axis under consideration.[8] The torque is determined by the product of the thrust produced by the valves for each axis and their respective moment arms. Placing the jet valves on the solar panel tips maximizes the moment arms and therefore minimizes the thrust required to produce a specified torque or acceleration. This, in turn, minimizes the gas consumption required for control torques.

Once the required thrust for each valve has been determined, the nozzle throat diameters are sized to produce this thrust. A computer program that gives the valve thrust as a function of nozzle throat diameter, ball travel, and other parameters has been developed for this purpose. Representative curves are shown in Fig. 40. As long as ball travel is adequate, thrust is determined by the nozzle throat diameter. This is the most desirable situation since ball travel cannot be modified once the jet valve has been built. For larger throat diameters, however, it is possible for flow rates to begin to approach sonic levels near the ball seat area, and with this condition, ball travel begins to influence thrust levels as shown in Fig. 40.

[8]This equation, while not generally valid for a freely rotating spacecraft, is adequate for purposes of this section.

The requirements to control 0.23-N-m (2-in.-lb) swirl torque with one assembly led to the selection of nozzle throat diameters of 0.84 mm (0.033 in.) with 74-mN (16.6-mlb) thrust. The pitch and yaw axes valves were sized to give 0.45 mrad/s^2 angular acceleration, and nozzles with 0.51-mm (0.020-in.) throat diameters and 33-mN (7.4-mlb) thrust were selected for this purpose. Table 11 summarizes these and other parameters. It can be seen from the table that the spacecraft moments of inertia are substantially affected by the orbit insertion burn and, as a result, so are the angular accelerations produced by the RCA. The effective moment arm for the roll axis is less than the other two axes because the roll gas nozzles are inclined at an angle of 21 deg from a line perpendicular to the solar panels to eliminate gas plume impingement. The effective thrusting time per pulse is the time an ideal valve would fire at full thrust to produce the same impulse produced by the actual jet valves. The tailoff (Fig. 39) for the roll valves is shorter than for the pitch and yaw valves because the larger nozzle diameters 0.84 mm vs 0.51-mm (0.033 vs 0.020 in.) allow the chamber pressure to bleed off faster. As a result, the effective thrusting time for the roll valves is slightly less than for the pitch and yaw valves.

E. GIMBAL ACTUATORS

The Mariner spacecraft uses a 1335-N (300-lbf) gimballed bipropellant rocket engine to perform the motor burns required for trajectory corrections, orbit insertion, and orbit trims. Actuation for the motor gimbals is provided by two identical linear electromechanical gimbal actuators contained in the ACS.

1. Physical Description

The actuator has no gears and performs its mechanical function by transforming the rotary motion of its drive motor to linear motion through the action of a recirculating ball lead screw assembly. Exterior and cutaway views of the actuator are shown in Figs. 41 and 42. It consists of three major parts contained in a pressurized, O-ring-sealed housing. The three parts are a linear motion transducer, a recirculating ball lead screw assembly, and a direct-current drive motor.

The motion transducer is a linear variable differential transformer excited by sine waves at 6000 Hz, 17 V rms. The demodulated transducer readout is 1.26 Vdc/cm (3.2 Vdc/in.) of linear movement. Average power required for the transducer readout is 0.2 W.

The ball screw assembly consists of a screw and mating nut, each having a specially formed concave helicoid ball groove. The nut, perfectly mated over the screw, contains the balls filling one or more circuits, which serve as the engagement medium between the nut and screw. Three ball circuits are used, each containing 35 balls of 0.16-cm (1/16-in.) diameter. The pitch of the screw is 0.25 cm (1/10 in.) per turn, with a ball circuit diameter of 1.6 cm (5/8 in.).

Transfer inserts spaced symmetrically around the nut circumference form a crossover path guiding the balls from the end of the turn to the start of the same turn. This arrangement provides a continuous recirculation or closed circuit of balls, which prevents the balls in a particular circuit from entering any other race within the nut.

The load-carrying capacity of the ball screw assembly is 2448 N (550 lbf) per circuit. With three circuits, this provides a static capacity of 7343 N (1650 lbf). The mechanical efficiency of the ball lead screw assembly is 94%. A high-chromium, hardenable stainless steel is used throughout.

The actuator motor is of the reversible dc type, which was chosen in preference to an ac type because, for the same power input, it is possible to develop a greater torque in a smaller package. An eight-pole configuration is used for the motor with a permanent-magnet stator placed outside, around the armature. The commutator is placed axially. The motor has a top speed of 700 rev/min with a linear torque speed characteristic to a stall torque of 0.49 Nm (70 oz-in.). The top speed will drive the actuator at a rate of 3.0 cm/s (1.2 in./s) under no-load conditions. The stroke is nominally 2.0 cm (0.8 in.), fully retracted to fully extended.

The actuator housing is filled with a mixture by volume of 90% nitrogen and 10% helium at 3.45×10^4 N/m^2 (5 psig). The gas pressurization of the actuator allows it to be lubricated by conventional lubricant and also protects the brushes and bearings. The helium in the gas mixture allows the use of a helium mass spectrometer to measure leak rate.

The actuator weighs 1.25 kg (2.75 lb). It operates on 0 to 30 V dc and draws 1.5 A at stall. Overall exterior dimensions are (approximately) 19.1 cm (7.5 in.) long with the actuator arm retracted and 9.9 cm (3.5 in.) in diameter at the widest point.

JPL designed and assembled the actuators. Actuator components were supplied by a number of different vendors.

2. Functional Description

Command excitation is provided to the single control winding of each actuator motor by the ACE in the form of a 6-kHz pulse width modulated wave. In response to the command inputs, the gimbal actuators extend and retract a few tenths of an inch, rotating the engine in its gimbal system (see Fig. 43). The maximum angular excursion about either gimbal axis is ±9.0 deg. This engine rotation capability allows the thrust vector to be aligned through the spacecraft center of mass, thereby maintaining attitude stability in pitch and yaw. Stability about the roll axis is maintained with the RCA as described before. The output of the linear motion transducer is fed back to the ACE as one input required to perform the TVC function.

3. Performance Parameters

The relationship between actuator force and linear speed is shown in Fig. 44. Table 12 summarizes some of the other important performance parameters.

F. ATTITUDE CONTROL ELECTRONICS

The attitude control electronics assembly (ACE) contains the electronic circuits required to control RCA torquing and gimbal actuator position, all logic for the ACS with the exception of that contained in the Canopus tracker, input buffering modules for most commanded inputs to the ACS, electrical interfaces with other spacecraft subsystems, power conditioning equipment, and a number of other minor circuits.

As illustrated before, the ACE is the main interface between the various assemblies of the ACS and between the ACS and other spacecraft subsystems. As a result, it is difficult to discuss the functional operation of the ACE without discussing the functional operation of the ACS as a whole. Subsection II-B was devoted to discussions of the ACS operating modes, and discussions in this section will be restricted primarily to the RCA electronics, and the TVC electronics, with only brief mention of other components of the ACE.

1. Physical Description

The ACE circuitry consists of linear and digital circuits, power supplies, and signal and power switching. All circuitry is packaged within one assembly (Fig. 45).

The linear circuits, power supplies, and power switching relays utilize components and integrated circuits that are packaged in transistor cans. These circuits are packaged in encapsulated welded cordwood modules that are interconnected by a nine layer board. The multilayer board utilizes plated through holes.

The digital and signal switching circuitry utilizes integrated circuits that are packaged in flat packs. These circuits are packaged in stick modules that use both welding and soldering processes.

The completed assembly weighs 3.7 kg (8.2 lb) and contains 1073 electronic parts (including 100 integrated circuits and 84 transistors). Its dimensions are 36.103 × 6.985 × 16.674 cm (14.135 × 2.750 × 6.600 in.).

2. RCA Electronics

The RCA electronics supply the command inputs required to actuate the jet valves. Three nearly identical circuits are used, one each for the pitch, yaw, and roll channels. Each circuit consists of a dc summing preamplifier, a switching amplifier, and a derived rate feedback circuit. The pitch and yaw channels are identical and also include a mixing and switching circuit for the Sun sensor inputs.

A functional diagram for the pitch (yaw) axis is illustrated in Fig. 46. Inputs to the scaling resistors of the preamplifier consist of gyro rate, gyro position, cruise and acquisition Sun sensor position, and a feedback signal from the derived rate circuit. Switches in the mixing and switching circuit determine which of the Sun sensor inputs are sent to the preamplifier. The status of the inertial reference unit determines whether or not the gyro inputs are available. The derived rate feedback signal is also switched.

The input scaling resistors of the preamplifier are individually selected for each spacecraft to give the desired system response characteristics. They determine the position deadband of the switching amplifier, the system damping in the gyro on modes, and the spacecraft turning rates in the acquisition modes.

The preamplifier output is sent to the switching amplifier (also called the driver). The switching amplifier has an input threshold set by zener diodes. If the input threshold is exceeded, the amplifier turns on. The circuit is designed so that 5 μA at the input to the preamplifier will cause this threshold to be exceeded. At the same time the amplifier turns on, supplying driving current for the thrust valves, a unijunction oscillator turns on, which after a 20-ms interval, turns off the amplifier. Thus once the switching amplifier threshold is exceeded it turns on and remains on for a minimum on-time of 20 ms and then turns off. If the input still exceeds the threshold, the amplifier turns on again immediately, otherwise it remains off until such time as the threshold is exceeded.

The switching amplifier has four outputs, two each for the clockwise and counterclockwise thrust valves. The valves are operated as a couple unless one of the valves fails. To increase the electronic reliability of the circuitry, independent drive transistors are used for each valve, so that if one valve fails (electrically) the other transistor drive is not affected. The switching amplifier also provides an output to the derived rate circuit coinciding with each on-time.

The derived rate circuit is enabled only when the gyro inputs are off. The circuit is a lag network with a zener diode clamp to limit the feedback voltage. Its purpose is to provide damping to the ACS during the celestial cruise mode.

3. TVC Electronics

The TVC electronics provide command inputs to the gimbal actuators. Inputs to the electronics from the inertial reference unit (IRU) and gimbal actuator motion transducer are used to obtain the desired command signals. Positioning of the motor gimbals is required during the launch and TVC modes. During the launch mode the IRU inputs are effectively disabled, and a fixed bias input is used to hold the gimbals in a constant position. Operation of the electronics in the TVC mode is of primary interest and is discussed in the following paragraphs.

Two nearly identical circuits are used, one each for the two gimbal actuators. A functional block diagram for one axis is shown in Fig. 47. The principal components are an input summing preamplifier (1), a path guidance loop, a pulse width modulated actuator driver, a demodulator for the linear motion transducer (LMT) of the linear actuator (LA), and a 6-kHz power supply.

Inputs to the scaling resistors of summing preamplifier (1) consist of pitch and yaw gyro rate and position, and a feedback from the path guidance. The gimbal actuator coordinate frame is not coincident with the pitch, yaw, and roll coordinate frame, hence the requirement for both pitch and yaw inputs for each channel. In effect, summing preamplifier (1) provides a coordinate transformation for the gyro inputs by weighting the inputs in the desired manner.

The path guidance circuit is a simple lag network utilizing an operational amplifier. Its purpose is to compensate for the spacecraft pointing error generated by the initial offset of the thrust vector relative to the spacecraft center of mass.

The 6-kHz power supply consists of an astable multivibrator clamped with zener diodes, a band-pass filter, and a power amplifier. It supplies a square wave signal to the input of the pulse width modulated actuator driver, and sine wave inputs to the motion transducer and demodulator.

The output of the linear motion transducer is a sine wave, with magnitude proportional to the actuator extension or retraction from null. The phase of the output is in phase with the excitation for extensions from null and 180-deg out of phase for retraction from null. This signal is converted to dc by the LMT demodulator, which performs full-wave rectification and switches the sign of the rectified signal depending upon the phase of the LMT output. The demodulator output is then smoothed by a filter and sent back to the summing junction, through a compensator, as the position feedback.

The pulse width modulated actuator driver consists of a summing preamplifier (2), followed by a lag circuit, two level detectors and two drivers. It operates in the following manner. At the input summing amplifier, the command and feedback signals are combined to produce an error signal. This error signal is combined with a square wave modulating signal and the resulting current is passed through a simple lag circuit. The time constant of the lag is selected to be large compared to the period of the square wave so that the output is a sawtooth wave with dc bias. This signal is sent to two power switches, one for extension and one for retraction. These switches produce 6-kHz pulse trains with the width of each pulse proportional to the error signal. The average of this signal is thus proportional to the error signal. Figure 48 illustrates this operation. Since the actuator motor cannot respond to the 6-kHz rate, it is sensitive only to its average value. One of the virtues of this type of circuit is that the motor is driven from switching transistors and thus the driver is very efficient.

As indicated in Fig. 47, the command inputs to the pulse width modulated driver consist of three biases in addition to the output of summing amplifier (1). These are all preaim biases, whose purpose is to position the motor gimbals so that the thrust vector will pass through the computed spacecraft center of mass at the start of each motor burn. One of these biases is programmed into the gimbal actuator electronics just prior to each burn by signals received from the CC&S (see Fig. 49). A 22-bit word is received by a serial register, the last 14 bits of which are retained. Seven bits of this word determine the A gimbal preaim bias and seven bits the B gimbal bias. Digital to analog converters DAC are used to convert the register contents into a dc voltage for input to the A and B axis summing points. Two other biases are used, one of which is fixed and the other of which is a switched backup bias.

During the launch mode, the gimbal actuator electronics are powered, but the outputs of summing preamplifier (1) are disconnected. For this mode the preaim biases are used to hold the gimbals in a fixed orientation to prevent possible damage from flopping in the launch environment. After launch, all power is removed until just prior to motor burn.

4. Input Signal Buffering and Logic Circuitry

The input buffering circuits receive control signals from other spacecraft subsystems and condition them for the logic circuitry. The primary purpose of the logic circuitry is to control the ACS mode of operation. Secondary functions are to select ACE telemetry outputs as a function of the operating mode, and to supply a Sun acquired logic signal to the spacecraft power subsystem. The function of the logic is documented more completely in Section IV.

5. Other Electronics

The remaining electronics consists of the power supplies, telemetry conditioning, and a number of minor circuits. The power supplies and telemetry conditioning will not be discussed here since their functional operation is relatively standard.

The minor circuits are all associated with the logic and consist of a roll search inhibit circuit, a Sun gate circuit, and a 180-s timer. The roll search inhibit provides abort control during the Canopus acquisition mode. If the negative roll thrust valve is actuated for more than approximately 30 s, the search bias input to the roll switching amplifier is removed. This is to prevent spin up of the spacecraft about the roll axis should the roll gyro signal fail.

The Sun gate circuit performs a simple level detection check on the output of the Sun gate. Depending on the output level of the Sun gate, a logical signal is generated that corresponds either to a Sun acquired or a Sun not acquired logical state.

The 180-s timer is an integral part of the logic, and acts as a clocking signal for certain logical sequences.

6. Performance Parameters

The performance parameters for the ACE are listed in Tables 13 and 14.

IV. LOGIC

A. INTRODUCTION

The logic equations will be presented in the following groups: Sun sensors, Canopus tracker, switching amplifiers, TVC, and telemetry.

Each group contains those equations directly affecting the operation of an assembly, or subassembly. Within each group (telemetry excepted) there are four major categories of equations, these are:

(1) Electrical power control.

(2) Output enabling for major assemblies and subassemblies.

(3) Enabling of specialized circuits within major assemblies and subassemblies.

(4) Generation of logic signals (or functions).

Taken together the first three categories of logic equations determine the operating mode of the ACS, and perform all switching functions within the ACS. The logic signals of category four are generated automatically within the ACS. External signals from the pyro arming switch (PAS), the central computer and sequencer (CC&S), and the flight command subsystem (FCS), are combined with these internally generated signals to provide control of the ACS operating modes.

Most of the logic equations are mechanized in the attitude control electronics (ACE). However, mechanization of the Canopus tracker equations is divided between the ACE and the Canopus tracker assemblies. The assembly in which the Canopus tracker equations are mechanized is identified explicitly in Section I, along with a summary of the logic equations and symbols used.

1. Notation: Logical Operations

The logical "or" is represented by the symbol (+). The logical "and" is represented by the symbol (•). The complement of A is denoted by $\overline{A}$. (i.e., A = A true, $\overline{A}$ = A false.) The words true, on, and high will be used interchangeably, as will the words false, off, and low.

2. Notation: Flip-Flop Terminology

The outputs of flip-flops are determined as follows: let A and $\bar{A}$ be the outputs of a flip-flop, then:

(1) A SET turns A on and turns $\bar{A}$ off.

(2) A RESET turns $\bar{A}$ on and turns A off.

B. INPUT COMMANDS

A brief description of the input commands to the ACS from the CC&S and the FCS are given in Section I. In addition, a signal from the PAS[9] is received by the ACS at spacecraft separation from the launch vehicle. This signal remains on throughout the mission.

The direct commands (DCs) (from the FCS) are input to the ACS as 100-ms pulses. Many of these pulses are input to flip-flops. The outputs of these flip-flops will be identified explicitly in the equations that follow by placing an "F" after the number of the direct command. The nomenclature will be as follows:

(1) The direct commands DC13, DC61, DC62, and DC64 set flip-flops whose outputs are DC13F, DC61F, DC62F and DC64F, respectively. DC14 is used to reset these flip-flops.

(2) The direct commands DC15, DC18, DC20, DC40, and DC63 set flip-flops whose outputs are DC15F, DC18F, DC20F, DC40F, and DC63F, respectively. DC19 is used to reset these flip-flops.

(3) DC12, DC14, DC17, DC19, and DC21 are not input to flip-flops; they remain on only for the duration of the 100-ms pulse.

(4) DC18 is the only direct command that appears in the forms DC18 and DC18F in the ACS logic equations.

When power is first applied to the ACS all the flip-flops associated with the direct commands are reset.

The duration of the commands from the CC&S is under program control; however, the commands 7A and 7B normally remain on throughout the

[9] An assembly contained in the devices subsystem of the spacecraft.

mission once they have been issued.[10] The command 7C is a periodic pulse occurring every hour with a pulse width of 6 ms. The command 7D is a single pulse with a nominal width of 100 ms.

The signals 7E and 7M1 through 7M6 are level commands from the CC&S. Each of these signals is anded with DC13F to provide a capability for aborting a spacecraft maneuver by direct command. Rather than indicate this in all the logic equations explicitly, we will write

$$A7Mi = 7Mi \cdot \overline{DC13F} \quad i = 1,\ 2,\ 3,\ 4,\ 5,\ 6$$

$$A7E = 7E \cdot \overline{DC13F}$$

When these signals are not anded with $\overline{DC13F}$, the A prefix will be dropped.

C. SUN SENSORS

1. Power Enable

The Sun sensors (Sun gate, cruise sensors, and acquisition sensors) are powered continuously throughout the mission. No logic equations are required.

2. Sun Gate Signal

The output of the Sun gate is available throughout the mission. Its output is converted into a logical signal by the ACE, which determines when the Sun is acquired. The Sun acquired signal (SG) is output from this circuit whenever the angular deviation of the Sun from the spacecraft roll axis is less than approximately 5 deg.

Since the Sun acquired signal is so important to the operation of the ACS, a backup direct command is provided. An alternate Sun acquired signal is then defined as:

$$ASG = SG + DC61F \tag{1}$$

[10] It is possible to turn 7A and 7B off during the mission by reprogramming the CC&S.

The signal ASG, in addition to its extensive use in the ACS logic equations, is supplied to the power subsystem.

3. Cruise Sensor Enable

The Cruise sensor outputs are input to the pitch and yaw switching amplifiers except during command turns, powered flight, and other portions of the mission when an attitude reference is established by gyro outputs alone. A7M2 is used to disable the cruise sensor inputs to the switching amplifier.

The logic equation for disabling the cruise sensor inputs to the pitch and yaw switching amplifiers is:

$$\text{cruise Sun sensor disable} = \text{A7M2} \tag{2}$$

4. Acquisition Sensor Enable

The acquisition Sun sensor inputs to the pitch and yaw switching amplifiers are disabled automatically whenever the cruise Sun sensors inputs are disabled. They are also disabled whenever the Sun has been acquired, a condition signaled by the presence of ASG.

The logic equation for disabling the acquisition sensor inputs to the pitch and yaw switching amplifiers is:

$$\text{acquisition Sun sensor disable} = \text{A7M2} + \text{ASG} \tag{3}$$

D. CANOPUS TRACKER

1. Canopus Tracker Power Enable

Canopus tracker power is inhibited during launch and after separation until Canopus acquisition is initiated by 7B. Power normally remains on for the remainder of the mission, but may be inhibited unconditionally by DC20F. Back up for 7B is provided by 7M1 and DC13F.

The logic equation for enabling Canopus tracker power is:

$$\begin{aligned}\text{Canopus tracker power enable} &= \text{CTPWR} \\ &= \overline{\text{DC20F}} \cdot (\text{7B} + \text{7M1} + \text{DC13F}) \qquad (4)\end{aligned}$$

2. Sun Shutter Power Enable

The Canopus tracker Sun shutter is powered whenever there is a possibility that the Canopus tracker might be pointed toward the Sun. The Sun shutter does not actually cover the Canopus tracker's field of view unless the tracker is pointed toward the Sun. An auxiliary sensor activates the shutter as required.

The logic equation used to enable power to the Sun shutter is:

$$\text{Sun shutter power enable} = \text{A} = \overline{\text{ASG}} + \text{other terms} \qquad (5)$$

The signal A is defined in Subsection IV-D-9. Signal A contains many terms in addition to $\overline{\text{ASG}}$, however, the absence of the Sun gate signal is a more than adequate criteria to use for controlling Sun shutter power.

3. Demodulator Enable

Perhaps the most crucial function performed by the Canopus tracker logic is to decide when to enable the output of the Canopus tracker demodulator. The Canopus tracker has a sensitivity that will permit it to detect any star of intensity greater than approximately 0.02 times that of Canopus. The integrated output of the demodulator is a dc signal proportional to the angular error between the line of sight to the star being sensed and the null plane of the spacecraft.[11] This signal is called the "roll error signal". The roll error signal is utilized for three distinct purposes:

(1) It is used as a feedback signal to the Canopus tracker roll deflection plates. This feedback signal causes the star image to be automatically centered in the scanned field of view.[12]

[11] See Appendix B for definition of null plane.

[12] See Subsection III-B-2 for definition of fields of view.

It is this automatic centering (or tracking) operation that distinguishes the Canopus tracker from the Canopus sensors employed on earlier Mariner spacecraft.[13] It should be noted that this centering is relative only to the roll (clock) axis, since the cone angle does not track the star image.

(2) The roll error signal is sent to the roll switching amplifier.[14] The roll switching amplifier in conjunction with the cold gas system then provides the control necessary to rotate the spacecraft so that the line of sight to the sensed star lies in the null plane.

(3) The roll signal is sent to two separate null detectors: one in the Canopus tracker and one in the ACE. These null detectors are used to generate logical signals that provide an alternate method of defining Canopus acquisition (i. e., an alternate to using star intensity to define Canopus acquisition).

The demodulator is enabled with the following logic equation:

$$\text{demodulator enable} = CA \cdot LG + DC15F \qquad (6)$$

The term CA • LG will enable the demodulator whenever a star of suitable brightness has been found. DC15F can be used to unconditionally enable the demodulator, and therefore bypass all the automatic star selection features incorporated in the term CA • LG.

CA is the Canopus acquisition signal, discussed in detail in Subsection IV-D-5. LG is the low-gate signal discussed in Subsection IV-D-4. In the absence of DC15F, CA • LG insures that the demodulator is enabled only when the star intensity falls between the lower and upper intensity gate settings discussed in the following subsection.

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[13] Mariners 6 and 7 were the first spacecraft to incorporate a Canopus tracker.

[14] See Subsection IV-D-9 for a discussion of the enabling of this roll error signal to the switching amplifier.

Star Intensity Gate Settings

The principal means of identifying the star Canopus is by its brightness. Two star intensity gates are used for this purpose: a lower and an upper gate. When a star having an intensity between these gate settings is encountered, it is identified as Canopus. Once the spacecraft has acquired the Sun, the brightest star in the field of view of the Canopus tracker is Canopus itself. Hence, the primary purpose of the high-gate setting is to prevent the tracker from locking onto a planet. The low gate is utilized to discriminate against stars dimmer than Canopus.

If a star enters the instantaneous field of view of the Canopus tracker with an intensity greater than the low-gate level, then the signal LG is output by the low-gate circuit. Similarly, the signal HG is output by the high-gate circuit if a star with intensity greater than the high-gate level is encountered.

The sketch below shows the signals LG and HG relative to the gate settings.

LOW GATE HIGH GATE

$$\longleftarrow \overline{LG} \cdot \overline{HG} \quad \vert\!\!\leftarrow \quad LG \cdot \overline{HG} \quad \rightarrow\!\!\vert \quad LG \cdot HG \longrightarrow$$

If the condition $LG \cdot \overline{HG}$ is satisfied, the Canopus tracker will track the star encountered that caused these signals to be generated. If there is more than one star (or object) in the field of view with brightness satisfying the gate requirements, the tracker will tend to follow the brightest object.

If the condition $\overline{LG} \cdot \overline{HG}$ is true, a star entering the instantaneous field of view will not be tracked unless DC15 has been sent [see Eq. (6)].

If the condition $LG \cdot HG$ is true, the star (or bright particle) will not be tracked unless the high gate has been disabled (see Subsection IV-D-4-b).

a. Low Gate Settings. The Canopus tracker utilizes an adaptive low gate having three possible level settings, progressively lower, numbered 1, 2, and 3. The adaptive gate feature was first implemented on Mariners 4 and 7. The intent was to provide an automatic safeguard against rejecting Canopus if for some reason the intensity signal generated by the tracker electronics fell below the nominal (number 1) low gate.[15]

Control of the low-gate setting is provided by signal 7C and DC12. The sequence of gate settings is 1, 2, 3, 1, ...[16] The logic equations mechanized are:

$$\text{low-gate step} = \text{DC12} + \text{X} \quad \text{7C} \tag{7}$$

where

$$\text{X SET} = \overline{\text{X RESET}} \cdot \text{7C} \tag{8}$$

$$\text{X RESET} = \text{PSUCA} + \text{7F} + \text{A} + \Delta\text{ACEPWR} \tag{9}$$

and

$$\Delta\text{ACEPWR} = \text{signal that pulses true at ACS power turn on}$$[17]

7C is a cyclic pulse output by the CC&S once every hour. X is set by the first 7C pulse received provided X RESET is not true. Once X is set, each subsequent pulse results in a gate level change. Note that each time X is reset, two 7C pulses are required to make a gate level change.

The symbols PSUCA and A are defined in Subsections IV-D-8 and IV-D-9 respectively. The condition $\overline{\text{X RESET}} = \overline{\text{PSUCA}} \cdot \overline{\text{7F}} \cdot \bar{\text{A}}$ insures that X is set only with the ACS in a roll search mode. $\overline{\text{7F}}$ implies lack of stray-light condition. If $\overline{\text{7F}}$ is true, $\overline{\text{PSUCA}}$ corresponds to a Canopus not acquired

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[15] The low-gate step was not required on the Mariner 6 and 7 missions.

[16] At Canopus tracker power turn on, the number 1 gate is set.

[17] This signal is not actually mechanized in the ACE, but is used here to show the manner in which some flip-flops are initialized at power turn on.

state, and output $\bar{A}$ enables the Canopus tracker error signal to the roll switching amplifier. These conditions taken together are sufficient to identify the roll search mode.[18]

b. High-Gate Enable. A fixed high gate is used. However, the high-gate signal (HG) is not directly used in the Canopus tracker logic. Instead HG is anded with the high-gate enable signal (HGen). Hence, if HGen is false, HG • HGen is false regardless of whether HG is true or false. Thus, HGen can be used to effectively remove the high gate.

The purpose of HGen is to avoid a problem experienced on the Mariner 4 mission. There were several instances when bright particles entered the field of view of the Canopus tracker after Canopus acquisition had been established. The resultant HG signal caused the ACS to go into a roll search. This very undesirable event was overcome, in part, by the use of the HGen signal.

HGen is the output of a flip-flop with logic:

$$\text{HGen SET} = \overline{\text{CA}} \cdot \text{LG} \tag{10}$$

$$\text{HGen RESET} = \text{CA} \cdot \Delta\text{ADEL} \cdot \text{RNULL} \tag{11}$$

where

(1) CA is the Canopus acquired signal defined in Subsection IV-D-5.

(2) RNULL is a signal generated by a null detector circuit in the Canopus tracker when the magnitude of the roll error signal is less than 0.5 deg.

(3) ΔADEL is a signal that pulses true 15 s after ANULL comes on

[18] The use of PSUCA in this equation is not required. PSUCA could be replaced with ACA [see Eq. (18)] and the same functional operation would be obtained. PSUCA was used here because it simplifies the hardware mechanization.

where

$$\text{ANULL} = \text{RNULL} \cdot \text{HGen} \tag{12}$$

HGen is set when a transition is made from a Canopus not acquired state ($\overline{\text{CA}}$) to a Canopus acquired state (CA). When this transition is made, the CA signal lags the LG signal by 10 ms, hence $\overline{\text{CA}} \cdot \text{LG}$ will be true during this 10-ms interval, and HGen is set.

HGen is also set when a transition is made from CA to $\overline{\text{CA}}$ under the action of DC21. When DC21 is issued (with DC18F off) CA is turned off, but LG will remain true for a short time thereafter. Hence, HGen is set (see Subsection IV-D-5). The high gate is disabled once a stable tracking condition has been obtained. Disabling is accomplished in the following manner:

Once the LG signal has been generated and the CA signal comes true, the Canopus tracker roll error signal is enabled to the roll switching amplifier (see Subsection IV-D-9). Based on this signal, the spacecraft is torqued about the roll axis to bring the star image into the null plane. The roll error signal is input to a null detector within the Canopus tracker, and the signal RNULL is generated when the magnitude of the roll error is less than 0.5 deg. At this time the signal ANULL comes on. Fifteen seconds after ANULL comes on a pulse is generated (ΔADEL) that resets HGen, provided Canopus is still being tracked. Hence HGen is not reset until a stable tracking condition has been obtained.

With HGen reset ($\overline{\text{HG}}$en true) the high gate is effectively disabled and a Canopus acquired signal is generated whenever LGDEL is true (see Subsection IV-D-5). Then, whether HG is true or false is of no importance as far as the operation of the Canopus tracker is concerned.

c. <u>High- and Low-Gate Override.</u> As mentioned before, the principal purpose of the star intensity gate settings is to determine when to enable the output of the Canopus tracker demodulator. Hence, as Eq. (6) indicates, DC15F can be used to effectively override this function of the star intensity gate settings. DC15F does not, however, actually remove the star intensity gates, nor is the output of these gates in any way affected.

5. Canopus Acquired Signal

The Canopus acquired signal (CA) is normally generated only when a star falls within the intensity gate settings. Once Canopus is acquired, however, the high gate is effectively removed (See Subsection IV-D-4-b) and CA will remain on so long as the low-gate setting is not violated. If the star intensity signal drops below the low gate, CA will remain on for 7s, during which time a fly back and sweep is performed (see Subsection IV-D-6-c). If Canopus is not reacquired during this 7-s interval, CA will be turned off and the ACS will go into a roll search.

The logic equations are:

$$CA = [\overline{D} + LGDEL] \cdot E + \Delta CTPWR \tag{13}$$

where

$$E\ SET = \overline{LGDEL} \tag{14}$$

$$E\ RESET = HG \cdot HGen + DC21 \cdot \overline{DC18F} \tag{15}$$

$\Delta CTPWR$ = signal that pulses true at Canopus tracker turn on

$$\overline{D}\ SET = CA \cdot \overline{LG} \cdot LGDEL \tag{16}$$

$$\overline{D}\ RESET = \Delta\overline{D}DEL \tag{17}$$

LGDEL = signal that lags LG by 10 ms

$\Delta\overline{D}DEL$ = pulse generated 7 s after $\overline{D}$ comes on; also generated at Canopus tracker power turn on.

The E flip-flop turns CA off if the high gate is violated (HG • HGen true), or by direct command (DC21 • $\overline{\text{DC18F}}$). E is set only if the star intensity falls below the low gate ($\overline{\text{LGDEL}}$ true). E will always be set at power turn on because LGDEL will always be off for at least 10 ms following power turn on.

$\overline{\text{D}}$ comes on during the 7-s interval following loss of a low gate signal. The term (CA • $\overline{\text{LG}}$ • LGDEL) is true only for 10 ms following loss of LG; this sets $\overline{\text{D}}$. $\overline{\text{D}}$ prevents the CA from being turned off during fly back and sweep, and hence prevents the gyros from being turned on during this interval (see Subsection IV-E-1). $\overline{\text{D}}$ is reset 7-s after being set by the pulse $\Delta\overline{\text{D}}$DEL. $\overline{\text{D}}$ is also reset at Canopus tracker power turn on.

The signal ΔCTPWR in Eq. (13) does not actually appear in the hardware mechanization of this equation, but the manner in which the flipflops $\overline{\text{D}}$ and E are set results in CA coming true momentarily at Canopus tracker power turn on. ΔCTPWR is included in this equation to indicate this phenomenon explicitly.

6. Alternate Canopus Acquired Signal.

Canopus acquisition is normally signaled by CA. However, if CA cannot be obtained, DC15 may be sent to override the star intensity gates. In this case, the roll error signal is used to obtain an alternate Canopus acquired signal. The roll error signal from the Canopus tracker is input to a null detector in the ACE. A logical signal (RECA = roll error Canopus acquired) is generated when the absolute value of the roll error is less than 2 deg. RECA then effectively replaces CA as the Canopus acquisition signal.

The logic equation identifying the alternate Canopus acquired state is:

$$\text{ACA} = \text{CA} \cdot \text{CTPWR} + \text{DC15F} \cdot \text{RECA} \qquad (18)$$

7. Scanned Field of View (SFOV) Bias Control

The Canopus tracker SFOV has two degrees of freedom: cone and clock.[19] The cone angle of the SFOV can be changed in discrete increments

[19] See Appendix B for definition of clock and cone angles.

only. The clock angle of the SFOV is controlled by three separate fixed biases in addition to the variable tracking bias that can be applied by the demodulator (see Subsection IV-D-3). Two of these, the fly back bias and the roll override bias, are hard biases in that if they are applied, they will predominate over both the demodulator tracking bias and the search bias. The search bias is a soft bias, in that it can be overridden by either of the hard biases or by the demodulator tracking bias. With DC15F on, however, the search bias will control the SFOV if no star is being tracked, in which case the demodulator, although enabled, has no output.

a. Cone Angle Updates. The SFOV can be biased to five discrete positions within the optical FOV to accommodate variations of the Canopus cone angle. The commands 7D and DC17 are used for updating the cone angle. These two signals are input directly to the Canopus tracker and are not routed through the ACE.[20] Upon receipt of either 7D or DC17, the cone angle is stepped to the next position. The sequence of position steps is 1 2 3 4 5 4 3 2 1•••.

The cone angle position is initialized prior to launch. Power turn on and turn off of the Canopus tracker does not affect the cone angle setting. The logic equation for updating the cone angle is:

$$\text{cone angle update} = 7D + DC17 \qquad (19)$$

b. Roll Override Bias. The roll override bias (or star reject bias) is used to position the SFOV against the plus clock edge of the optical field of view. This bias is applied by direct command and provides a means of breaking the Canopus tracker lock on any star being tracked. The logic equation is

$$\text{roll override bias} = DC21 \cdot \overline{DC18F} \qquad (20)$$

[20] Signal 7D and DC17 are the only two signals input to the Canopus tracker that are not first conditioned by the ACE.

$\overline{DC18F}$ prevents the roll override bias from being applied when DC21 is being used to perform roll one shots (see Subsection IV-E-5). With application of the roll override bias, the Canopus acquired signal is lost, the roll gyro is turned on, and a roll search for a new star is begun (see Subsections IV-E-1 and IV-E-2).

c. Search and Fly Back Bias. The search bias or acquisition bias is used to position the scanned field of view of the Canopus tracker against the plus clock edge of the optical field of view. (When the spacecraft is in roll search, the direction of roll is such that the star field will appear to move from the plus clock to the minus clock edge of the optical field of view.) The fly back bias positions the SFOV on the minus clock edge of the optical field of view. When a fly back is commanded, the search bias will be turned on if it is not already on. The fly back bias is applied for 10 ms and causes the SFOV to move to the minus clock edge of the optical FOV, overriding the search bias. When the fly back bias is removed, the search bias causes the SFOV to sweep towards the plus clock edge of the optical FOV.

The logic equation for search bias enabling is:

$$\text{search bias enable} = \overline{CA} + \overline{LG} + DC15F \tag{21}$$

The term $\overline{CA}$ insures that the search bias is applied whenever Canopus acquisition is lost. $\overline{LG}$ insures that the bias is enabled for fly back and sweeps.[21] DC15F is redundant and provides a back up for $\overline{CA}$ and $\overline{LG}$.[22]

The logic equation for fly back bias enabling is:

$$\text{fly back bias enable} = [\Delta FYBSW + \overline{C}] \cdot \overline{DC15F} \cdot \overline{DC21 \cdot \overline{DC18F}} \tag{22}$$

where

$$\overline{C} = CA \cdot \overline{LG} \cdot LGDEL \tag{23}$$

[21] Recall that CA is true during fly back and sweep (Subsection IV-D-5).

[22] DC15 will not normally be on unless CA cannot be obtained (i.e., $\overline{CA}$ is true).

This equation can be explained as follows:

(1) ΔFYBSW is a signal that pulses true for 10 ms after FYBSW changes from false to true.[23] FYBSW is a fly back command generated in the ACE. FYBSW will come on after a stray-light condition, if Canopus acquisition has been lost during the stray-light condition. (See Subsection IV-D-8.)

(2) $\overline{C}$ will be true only for the 10-ms interval after LG is lost and before LGDEL goes off. Hence if a star is being tracked and is lost, the fly back bias is enabled provided both DC15F and $DC21 \cdot \overline{DC18F}$ are false.

(3) The term $DC21 \cdot \overline{DC18F}$ is used to reject a star by positioning the SFOV against the plus clock edge of the swept field of view. This action will cause $\overline{C}$ to come true for 10 ms. Hence the term $\overline{DC21 \cdot \overline{DC18F}}$ must be included to prevent a fly back, otherwise the star rejected would be immediately reacquired.

(4) When DC15F is on, the roll error signal is supplied continuously to the ACE (see Subsection IV-D-3). If fly back and sweep were performed with DC15F on, the roll error signal would have large fluctuations causing large transients in the RCA torquing. $\overline{C}$ might come true, for example, if a bright particle swept through the tracker SFOV. To avoid a fly back and sweep under these circumstances, $\overline{DC15F}$ is included in the above equation.

It should be noted that anytime the FYBSW signal is generated the LG signal will be low. Hence, the search bias will be enabled and will cause the SFOV to sweep from the minus toward the plus clock edge of the optical field of view after the fly back bias is removed.

8. Fly Back and Sweep Signal

In addition to the logic internal to the Canopus tracker for generating a fly back bias, a fly back and sweep command can be generated external to

[23]No pulse corresponding to ΔFYBSW is actually generated in the Canopus tracker hardware, but the fly back bias is only enabled for 10 ms following a low-high transition of FYBSW.

the Canopus tracker by logic in the ACS. The fly back and sweep signal (FYBSW) is generated after a stray-light condition (signal 7F), if the Canopus acquisition signal has been lost during the stray-light condition. If Canopus can be reacquired with the fly back and sweep, a roll search is avoided.

The mechanization uses three flip-flops to generate FYBSW with the following logic:

(1) The PSUCA (pseudo Canopus acquired) flip-flop tracks the alternate Canopus acquired (ACA) signal in the absence of stray light:

$$\text{PSUCA SET} = \text{ACA} \cdot \overline{7\text{F}} \tag{24}$$

$$\text{PSUCA RESET} = \overline{\text{ACA}} \cdot \overline{7\text{F}} \tag{25}$$

Note that PSUCA will be reset during launch since ACA and signal 7F will be off.

(2) If ACA is lost during a stray-light condition, an auxiliary state (Z) is set by another flip-flop:

$$\text{Z SET} = \overline{\text{ACA}} \cdot 7\text{F} \cdot \text{PSUCA} \tag{26}$$

(3) If Canopus is not reacquired by the time the stray-light condition ends, FYBSW is set:

$$\text{FYBSW SET} = \text{Z} \cdot \overline{\text{ACA}} \cdot \overline{7\text{F}} \tag{27}$$

(4) Both Z and FYBSW are reset whenever PSUCA is set, and at initial ACS power turn on,

$$\text{Z RESET} = \text{FYBSW RESET} = \text{ACA} \cdot \overline{7\text{F}} + \Delta\text{ACEPWR} \tag{28}$$

Note that Z and FYBSW are never reset during a stray light condition.

From Eq. (28) it can be seen that FYBSW is not reset unless Canopus is reacquired.[24] If Canopus is not reacquired during the fly back and sweep then the ACS will not go into a roll search because the Canopus tracker integrator output to the roll switching amplifier will not be enabled (see Eq. 33).

In this case the ACS would remain in a constant attitude under gyro control. If this were to happen on a mission, direct commands would be required to get the ACS back into a celestial acquired cruise mode. There are two possibilities:

(1) The Canopus tracker power can be turned off and back on by DC20. This will cause CA to pulse true as indicated by Eq. (17) and reset FYBSW. With FYBSW reset the Canopus tracker roll error signal will be enabled to the Canopus tracker and a roll search will be initiated.

(2) DC18 and DC21 can be used to reposition the spacecraft so as to bring Canopus back into the field of view of the Canopus tracker (see Subsection IV-E-5).

Actually the probability of not reacquiring Canopus after stray light is small. The spacecraft would have to drift over 5 deg during stray light to place Canopus outside the limits of the field of view of the Canopus tracker during the fly back and sweep. Typical duration of a stray-light condition is on the order of 1 to 2 h.

It might first appear that the PSUCA flip-flop is unnecessary. If it were eliminated, FYBSW could be generated after stray light whenever ACA was off regardless of whether or not ACA was on or off prior to stray light. This is not desired, however, for the following reasons. If ACA was not on prior to the stray-light condition, the ACS would be in the roll search mode and Canopus would not be in the optical field of view. There is only a small probability that Canopus would be in the optical field of view after stray light if it were not there prior to stray light. Hence, a fly back and sweep would almost never result in Canopus being reacquired, and FYBSW would remain on. To continue roll search after stray light, however, the Canopus tracker

[24] ACS power is not turned off during the mission.

integrator output to the roll switching amplifier must be enabled (see Eq. 33). This cannot be accomplished with FYBSW on. Hence it is desirable to prevent FYBSW from being generated after stray light if Canopus was not acquired prior to stray light. This will allow the spacecraft to continue roll search after stray light ends.

9. Canopus Tracker Integrator Enable

The output of the Canopus tracker integrator is enabled to the roll switching amplifier by logic contained in the ACE. This enabling operation shall be called Canopus tracker integrator enabling and is not to be confused with enabling of the Canopus tracker demodulator discussed in Subsection IV-D-3. The output of the Canopus tracker integrator is the sum of the demodulator output and the fixed biases discussed in Subsections IV-D-7-b and IV-D-7-c. Ordinarily, if the demodulator is enabled, the fixed bias will not be on and vice versa. The exception to this is that when DC15 is on, the demodulator is enabled and the search bias is on.

When the search bias is input to the roll switching amplifier, the ACS begins a roll search. When the demodulator tracking bias is input to the roll switching amplifier, the ACS acts to bring the star image into the null plane. Hence, the Canopus tracker integrator output must be enabled to the roll switching amplifier during the roll search mode and during the Canopus acquired cruise mode. The equations that follow insure that these two primary functions are accomplished. Many additional terms are in the equations, however, for various reasons to be mentioned after stating the logic equations.

The logic equation mechanized is:

$$\text{Canopus tracker integrator enable} = \text{CTEN} = (\text{PSUCA} + \overline{\text{7F}}) \cdot \overline{\text{A}} \qquad (29)$$

where

$$\begin{aligned} \text{A} = {} & \text{RSI} + \text{DC20F} + \overline{\text{ASG}} + \\ & (\text{PSUCA} \cdot \text{7F} + \text{DC18F} + \text{A7M2} + \text{FYBSW}) \cdot \text{GPWRONDE} \end{aligned} \qquad (30)$$

GPWRONDE = output of the 3-min timer used in conjunction with gyro power turn on (see Subsection IV-E-1); GPWRONDE comes on three minutes after gyro power is applied.

RSI is the output of a flip-flop with logic:

$$\text{RSI SET} = \text{roll counterclockwise gas valves on continuously for more than 30 s} \tag{31}$$

$$\text{RSI RESET} = \text{DC21} \cdot \overline{\text{DC18F}} + \text{A7M1} + \Delta\text{ACEPWR} \tag{32}$$

Taking the complement of A, and substituting into Eq. (29) we have, after some manipulation, the following equivalent equation:

$$\begin{aligned} \text{CTEN} = {} & \overline{\text{RSI}} \cdot \overline{\text{DC20F}} \cdot \text{ASG} \cdot \Big[\overline{\text{DC18F}} \cdot \overline{\text{A7M2}} \cdot \overline{\text{FYBSW}} \cdot \overline{\text{7F}} \\ & + \overline{\text{GPWRONDE}} \cdot (\text{PSUCA} + \overline{\text{7F}})\Big] \end{aligned} \tag{33}$$

The complement of Eq. (29) is also given for reference:

$$\begin{aligned} \overline{\text{CTEN}} = {} & \text{RSI} + \text{DC20F} + \overline{\text{ASG}} \\ & + (\text{DC18F} + \text{A7M2} + \text{FYBSW}) \cdot \text{GPWRONDE} \\ & + (\overline{\text{PSUCA}} + \text{PSUCA} \cdot \text{GPWRONDE}) \cdot \text{7F} \end{aligned} \tag{34}$$

The conditions for enabling CTEN are such that CTEN is on during the roll search mode when GPWRONDE is true and during the Canopus acquired cruise mode when GPWRONDE is false [see Eq. (33)].

Each term in Eq. (34) will now be explained.

(1) DC20F: DC20 is used to place the ACS in the roll drift mode, and DC20F disables the Canopus tracker integrator unconditionally as a part of this mode change.

(2) $\overline{\text{ASG}}$: during Sun acquisition (ASG off) the integrator is automatically disabled. The use of $\overline{\text{ASG}}$ for this purpose insures that the Sun is acquired prior to Canopus. With the Sun acquired, the probability of acquiring some star other than Canopus is minimized.

(3) RSI: RSI is the roll search inhibit. Its primary function is to guard against spin up of the spacecraft while in the roll search mode in the event that the roll gyro rate signal fails. During roll search the search bias is input to the roll switching amplifier. If the roll gyro rate signal fails, this constant bias would result in a constant torque being applied about the roll axis by the cold gas system. A spacecraft spin up would result. This spin up would be such that the roll counterclockwise gas valves were on continuously and RSI would be set according to Eq. (31). The Canopus tracker integrator would then be disabled and would remove the search bias from the roll switching amplifier.

(4) DC18F • GPWRONDE: DC18 places the roll axis under gyro control and the Canopus tracker integrator output is disabled as soon as GPWRONDE comes on (3-min after DC18 is sent).

(5) A7M2 • GPWRONDE: A7M2 enables the gyro integrators to all three axes and places the ACS in the inertial mode. Ordinarily signal A7M1 precedes A7M2 so that GPWRONDE is already on. Hence GPWRONDE acts as a back up precaution in this term if A7M1 fails.

(6) FYBSW • GPWRONDE[25]: if after a stray-light condition a flyback and sweep is performed, the integrator is not enabled until FYBSW goes off. FYBSW will be reset if Canopus is reacquired during the fly back and sweep (see Subsection IV-D-8). If Canopus is not reacquired, the integrator will not be enabled thus preventing the ACS from going into a roll search. It is desirable

[25] The term GPWRONDE is not important in this expression since the gyros are normally on when FYBSW is generated. If they were off, it would be because of a direct command [see Eq. (36)].

to prevent a roll search under these circumstances since almost certainly Canopus is still in the field of view, and if it was not reacquired, the ACS is almost certainly in a failure mode. Under these conditions it is preferable to direct the spacecraft by direct command.

(7) $(\overline{\mathrm{PSUCA}} + \mathrm{PSUCA} \cdot \mathrm{GPWRONDE}) \cdot 7\mathrm{F}$: the intent of this term is to disable the Canopus tracker integrator during a stray-light condition when the output of the Canopus tracker is unreliable.[26]

E. INERTIAL REFERENCE UNIT

1. Gyro Power Logic Signals

Provisions are made so that all gyros may be operated simultaneously, or so that under certain conditions the roll gyro may be turned on by itself. Before any gyro can be powered the logical word GPWR must be turned on. GPWR controls a 3-min timer whose outputs (GPWRD, INVPWR, and GPWRONDE) are illustrated in Fig. 50 as a function of GPWR. INVPWR is used to actuate a relay in the electrical power subsystem that turns on gyro power. GPWRD is used to control power switching to the pitch and yaw gyros. GPWRONDE is used to identify the time intervals when the gyro output information is usable. The function of these signals is discussed in more detail in conjunction with their use in various logic equations.

---

[26] This intent would be better accomplished if PSUCA were eliminated and signal 7F used by itself. PSUCA was originally used in the equations to perform a function in conjunction with a stray-light output from a stray-light sensor. The Mariner Mars 1971 design does not utilize a stray-light sensor, however; instead the CC&S command 7F is used to signal stray light. When this design change was made, PSUCA was not removed from the logic equations, although it serves no purpose except in the generation of the fly back signal FYBSW (see Subsection IV-D-8). The use of PSUCA has the following consequence: it is possible to have the gyros off in a stray-light condition and have PSUCA true; for example, DC40 could be sent (see Eq. 35). In this case CTEN would remain on. If CA was off, the search bias would be enabled and the spacecraft could conceivably spin up about the roll axis; so RSI might turn on. This undesirable possibility could be eliminated if PSUCA were removed from the equations.

The logic equation for GPWR is as follows:

$$
\begin{aligned}
\mathrm{GPWR} = {} & [\overline{\mathrm{SWAMPR}} + \overline{\mathrm{ASG}} + (\overline{\mathrm{ACA}} + \mathrm{7F}) \cdot \overline{\mathrm{DC20F}} \\
& + \mathrm{A7M1} + \mathrm{A7M2} + \mathrm{DC18F} \\
& + \mathrm{DC63F}] \cdot (\overline{\mathrm{7G}} + \mathrm{DC63F}) \cdot \overline{\mathrm{DC40F}}
\end{aligned}
\tag{35}
$$

The complement of Eq. (35) is also given for reference:

$$
\begin{aligned}
\overline{\mathrm{GPWR}} = {} & \mathrm{SWAMPR} \cdot \mathrm{ASG} \cdot [\mathrm{ACA} \cdot \overline{\mathrm{7F}} \\
& + \mathrm{DC20F}] \cdot \overline{\mathrm{A7M1}} \cdot \overline{\mathrm{A7M2}} \cdot \overline{\mathrm{DC18F}} \cdot \overline{\mathrm{DC63F}} \\
& + \mathrm{7G} \cdot \overline{\mathrm{DC63F}} + \mathrm{DC40F}
\end{aligned}
\tag{36}
$$

The purpose of each term in the above equations can be explained as follows:

(1) SWAMPR is the logic signal that enables the roll switching amplifier (see Subsection IV-F-3. The SWAMPR is off only during launch, hence this term insures that GPWR is on during launch.

(2) ASG is the alternate Sun gate signal. $\overline{\mathrm{ASG}}$ insures that GPWR remains on during Sun acquisition.

(3) The term $\overline{\mathrm{ACA}} \cdot \overline{\mathrm{DC20F}}$ causes GPWR to come on whenever Canopus acquisition is lost, unless DC20F is on. This will allow the ACS to go into a roll search for Canopus. DC20F is used to permit the ACS to operate in a roll drift mode with the gyros off and Canopus not acquired.

(4) 7F $\cdot$ $\overline{\mathrm{DC20F}}$ causes the gyros to come on during stray light so that the ACS may operate in the inertial mode. DC20F can be used to override this operation.

(5) A7M1 is sent to turn on the gyros prior to the command turn mode where gyro control is required.

(6) A7M2 is a back up to A7M1 in these equations.

(7) DC18F is used to turn on the roll gyro prior to 2-deg roll increment commands (see Subsection IV-E-5).

(8) DC63F is used to turn on the roll gyro and overrides all other direct commands except DC40F.

(9) 7G is used to turn off the gyros and overrides all other commands except DC63F.

(10) DC40F is used to unconditionally inhibit gyro power.

2. Gyro Power Enable

Power to the roll gyro is turned on whenever INVPWR comes on. Hence we have

$$\text{roll gyro power on} = \text{INVPWR} \tag{37}$$

INVPWR is not reset until 3 min after GPWR is turned off so that Canopus acquisition can be completed with gyro rate damping.

Power to the pitch and yaw gyros is controlled by the following logic equation:

$$\text{pitch and yaw gyro power on} = \text{INVPWR} \cdot \overline{\text{RONY}} \tag{38}$$

The logical signal RONY (roll gyro only) is the output state of a flip-flop. RONY controls a relay that can interrupt inverter power to the pitch and yaw gyros and hence provides the option of operating the subsystem with the roll gyro only powered.

Rony is set with the following logic:

$$\text{RONY SET} = [\overline{\text{ACA}} + 7\text{F} + \text{DC18F} + \text{DC63F}] \cdot \overline{\text{GPWRD}} \tag{39}$$

Note that RONY may be set only when GPWRD is off. The 90-ms delay by which the signals INVPWR and GPWRD follow GPWR assures that RONY is set and power input to the pitch and yaw gyros is disabled before power is supplied from the inverter to the gyros. During the time GPWRD is on, RONY cannot be set. This prevents opening of the power input switch to the pitch and yaw gyros by the RONY switch after power has been applied and eliminates a potential electrical arcing problem.

If GPWRD is off, then RONY may be set in any of the following ways:

(1) When Canopus acquisition is lost ($\overline{ACA}$).

(2) When a stray-light condition occurs (7F).

(3) By direct command (DC18F or DC63F).

RONY is reset with the following logic:

$$\text{RONY RESET} = [\text{ACA} \cdot \overline{\text{7F}} \cdot \overline{\text{DC18F}} \cdot \overline{\text{DC63F}}] \cdot \overline{\text{GPWRD}} + \overline{\text{ASG}} + \text{A7M1} + \text{A7M2} \tag{40}$$

RONY RESET may be interpreted as follows:

(1) $\text{ACA} \cdot \overline{\text{7F}} \cdot \overline{\text{DC18F}} \cdot \overline{\text{DC63F}}$ is the inverse of an expression containing like terms in the RONY SET equation. Hence, if GPWRD is off, RONY will be reset when Canopus is reacquired provided that 7F, DC18, and DC63 are all off.

(2) $\overline{\text{GPWRD}}$ is logically anded with the terms in (1) above to prevent RONY from being reset and enabling the pitch and yaw gyro power before the inverter power is turned off (i.e., to prevent momentary pitch and yaw gyro turn-on when one of the conditions calling for RONY disappears).

(3) Loss of Sun acquisition ($\overline{\text{ASG}}$ true) will reset RONY. This term assures that all gyros are on for Sun acquisition. It also assures that RONY is reset prior to spacecraft separation since ASG is off during launch.

(4) A7M1 and A7M2 initiate full gyro control of spacecraft attitude and reset RONY.

3. Gyro Integrator Enable

During certain portions of the mission the celestial outputs to the switching amplifiers are disconnected. When this occurs, rate and position information are normally obtained from the gyro outputs and supplied to the switching amplifier. Rate information is supplied by the gyros directly, whereas position information is obtained by integrating the rate. Provisions are made to control the roll channel integrator separately from the pitch and yaw channel integrators.

Normally the rate integrators in the pitch and yaw channel are operated only during the commanded turn and TVC modes. However, they will be turned on any time the gyros are used as the only reference for attitude control. The integrator outputs are connected directly to the switching amplifier inputs, so that whenever the integrators are turned on, their outputs are automatically input to the switching amplifiers.

The logic equation for enabling the pitch and yaw gyro integrators is:

$$\text{pitch and yaw gyro integrators enable} = \text{A7M2}$$

The roll gyro integrator is controlled by the following logic equation:

$$\text{roll gyro integrator enable} =$$

$$RI_{en} = (\text{A7M2} + \text{PSUCA} \cdot \text{7F} + \text{FYBSW} + \text{DC18F}) \cdot \text{GPWRONDE} \qquad (42)$$

Hence, the condition that turns on the pitch and yaw integrators (signal A7M2) also turns on the roll integrator.

In addition, when GPWRONDE is on:

(1) A stray-light condition (7F) will turn on the roll integrator.[27] This insures that the roll axis is in the inertial mode during stray light.

[27] The term PSUCA should have been eliminated from this equation. It is desirable to place the roll axis in the inertial mode during stray light regardless of whether PSUCA is true or not. See footnote 24, Subsection IV-D-9.

(2) A fly back and sweep (FYBSW) will hold the roll gyro integrator on after coming out of a stray-light condition. This insures that the ACS remains in the roll inertial mode until Canopus is reacquired.

(3) DC18F will turn on the roll gyro integrator since it is required to perform 2-deg roll increments.

The GPWRONDE signal is used to prevent turning on the roll integrator during the 3-min delay allowed for gyro spin up and drift stabilization. This delay is not required in the logic for the pitch and yaw gyro integrator since the gyros are always on well before signal A7M2 is issued.

4. Turn Commands

Turn commands are provided to the inertial reference unit during the commanded turn mode.

Prior to execution of the turn commands, the gyros will have been turned on by signal A7M1 and the gyro rate integrators enabled by A7M2. The turn polarity is then selected with the following logic equations:

$$\text{positive turn select} = \text{A7M3} \tag{43}$$

$$\text{negative turn select} = \overline{\text{7M3}} \cdot \overline{\text{DC13}} \tag{44}$$

The logic used to execute turns is as follows:

$$\text{roll turn execute} = \text{A7M4} \tag{45}$$

$$\text{yaw turn execute} = \overline{\text{7M4}} \cdot \text{A7M5} \tag{46}$$

The magnitude of the roll and yaw turns is determined by the duration of signals A7M4 and A7M5 respectively.

5. Two-Degree Roll Turn Increments

An option is provided for positioning the spacecraft about its roll axis through the use of direct commands in increments of 2 deg. To use this option (also referred to as "two-degree, one-shot option") the ACS is first placed in the roll inertial mode by DC18. With this accomplished, each subsequent transmission of DC18 and DC21 causes the spacecraft to be torqued about the roll axis according to the following logic equations:[28]

$$+2 \text{ deg roll increment} = \text{DC18} \cdot \text{GPWRONDE} \tag{47}$$

$$-2 \text{ deg roll increment} = \text{DC21} \cdot \text{GPWRONDE} \cdot \text{RI}_{en} \tag{48}$$

It should be noted that if the gyros are already powered (GPWRONDE on) when the first DC18 is sent, then a positive 2-deg roll increment will result. If DC18 is used to turn on the gyros, then at least 3-min must elapse before DC18 and DC21 can be used to produce the roll increments because of the time delay required before GPWRONDE comes on (see Subsection IV-E-1.

6. Gyro Enable

The rate outputs of the gyros are enabled continuously. The system mechanization is such that whenever the gyros are powered their rate outputs are connected directly to their respective switching amplifiers. There are no separate switches to delay this rate input until the gyros come up to speed. (See also Subsection IV-F-2.)

7. Accelerometer Power Enable

The accelerometer power is enabled at the same time power to the pitch and yaw gyros is enabled (see Subsection IV-E-2). There is no requirement for the accelerometers to be powered, except for the TVC mode, but enabling in conjunction with the pitch and yaw gyros simplifies the hardware mechanization.

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[28] These two equations are not mechanized as such in the ACE. DC18 and DC21 are supplied to the IRU one-shot circuit directly. The 2-deg roll increments will not occur, however, unless these equations are satisfied.

8. Accelerometer Capture Loop Disable

The accelerometer is held in the capture loop mode of operation during launch and separation to prevent damage from the high-acceleration environment. Release from the capture mode is normally accomplished at the start of Canopus acquisition with signal 7B. Backup for 7B is provided by 7M1 and DC13.

The logic equation for disabling the accelerometer capture loop is:

$$\text{Accelerometer capture loop disable} = 7B + 7M1 + DC13F \qquad (49)$$

9. Accelerometer Enable

The accelerometer output is enabled continuously. No logic equation is required.

F. SWITCHING AMPLIFIERS

1. Power enable

Power to the ACE (with the exception of the TVC circuitry) is supplied throughout the mission. This includes power to the switching amplifiers, logic elements, and several minor circuits. No logic equations are required.

2. Derived Rate Enable

The derived rate networks are normally enabled only while celestial references are acquired. Gyro turn on is used to disable derived rate. The logic equations are:

$$\text{pitch and yaw derived rate disable} = INVPWR \cdot \overline{RONY} \qquad (50)$$

$$\text{roll derived rate disable} = INVPWR \qquad (51)$$

Simultaneous with gyro turn on, the derived rate inputs to the switching amplifiers are disconnected and the gyro rates replace them as inputs (see Subsection IV-E-6).

3. Switching Amplifier Enable

The switching amplifiers for the pitch, yaw, and roll channel are normally enabled simultaneously at spacecraft separation by a signal from the pyrotechnic arming switch (PAS). The roll switching amplifier then remains on throughout the mission. Backup for PAS is provided by signals 7A, 7B, 7M1, and DC13.

The logic equation for enabling the roll switching amplifier is:[29]

$$\text{roll switching amplifier enable} = \text{SWAMPR} = \text{PAS} + 7\text{A} + 7\text{B} + 7\text{M1} + \text{DC13F} \tag{52}$$

The pitch and yaw switching amplifiers are disabled during TVC when pitch and yaw control is provided by the TVC electronics. Disabling is accomplished with signal A7M6.

The logic equation for enabling the pitch and yaw switching amplifiers is:

$$\text{pitch and yaw switching amplifier enable} = \text{SWAMPPY} = \text{SWAMPR} \cdot \overline{\text{A7M6}} \tag{53}$$

G. THRUST VECTOR CONTROL

1. Power Enable

The TVC electronics (also called the autopilot) is a part of the attitude control electronics, but separate power switching is provided. Application of TVC power powers both the TVC electronics and the gimbal actuators.

[29] The use of PAS in this equation has undesirable consequences since PAS is issued prior to solar panel deployment. This is particularly true if separation takes place in Sunlight.

The TVC electronics will normally be powered only during launch and motor burn. Power is supplied to the TVC electronics during launch to prevent gimballed engine flopping under the effects of the launch environments. The logic equation for TVC power enabling is:

$$\mathrm{TVCPWR} = \mathrm{A7E} + \mathrm{A7M1} + \overline{\mathrm{SWAMPR}} \cdot \mathrm{GPWRONDE} \tag{54}$$

signal A7E preceeds the commanded turn and TVC modes, and A7M1 provides backup for A7E. The roll switching amplifier is disabled only during launch so $\overline{\mathrm{SWAMPR}} \cdot \mathrm{GPWRONDE}$ insures that the TVC power is on for launch.[30]

2. Preaim Bias Select

There are two different preaim backup bias levels: preinsertion and postinsertion. These biases provide backup for the normal preaim register inputs to the gimbal servo amplifiers. The logic controlling the selection of these biases is as follows:

$$\text{preinsertion preaim backup bias} = \mathrm{DC62F} \cdot \overline{\mathrm{DC64F}} \tag{55}$$

$$\text{postinsertion preaim backup bias} = \mathrm{DC62F} \cdot \mathrm{DC64F} \tag{56}$$

If DC62 is not issued, the normal preaim registers will supply the required biases.

$$\text{normal preaim register bias} = \overline{\mathrm{DC62F}} \tag{57}$$

3. Path Guidance Enable

The path guidance circuitry is enabled just prior to motor burn (TVC) by signal A7M6. The logic equation is:

$$\text{path guidance enable} = \mathrm{A7M6} \tag{58}$$

---

[30] GPWRONDE is used in this term to assure proper logic initialization during ACS power turn on. It also reduces power turn on transients.

4. TVC Enable

The TVC output signals that drive the gimbal servos during motor burn are enabled following launch in conjunction with the roll switching amplifier enabling. The logic equation for enabling the TVC module output signals is:

$$\text{TVC enable} = \text{SWAMPR} \tag{59}$$

Since the TVC electronics is not powered after launch except when 7E or 7M1 has been issued, TVC enabling has no effect on system operation until A7E or A7M1 come on.

H. TELEMETRY

Five telemetry channels are assigned to the ACS. The channels and signals telemetered are given in Section I. Channel 105 is shared between the cruise Sun sensor error output for the pitch axis, the pitch gyro rate output, and the pitch gyro integrator output. Only one signal is telemetered at a time. The logic that selects the signal to be telemetered is given in Section I also. Channel 106 is identical to 105 except yaw axis information is carried.

Channel 107 is shared between the Canopus tracker integrator output, the roll gyro rate output and the roll gyro integrator output.

Channel 112 carries the combined Sun sensor outputs[31] for the pitch axis and the TVC "A" gimbal position. Channel 113 is identical to channel 112 except it carries the yaw Sun sensor outputs and the TVC "B" gimbal position.

I. LOGIC SUMMARY

1. Input Commands

A brief description of each of the CC&S commands is given in Table 15. Table 16 gives a brief description of the flight command subsystem (FCS) direct commands.

[31] The sum of the acquisition and cruise sensor error signals.

2. Tabulation of Equations and Symbols

The logic equations are repeated in Tables 17 through 22 for reference. Table 23 gives a list of symbols used in Section IV, a short name, and an index to their definition.

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Table 1. Position deadband widths

Mode	Pitch and yaw axes, mrad	Roll axis, mrad
Celestial cruise	±4.3	±4.3
All-axes inertial	±3.0	±1.5
Roll inertial	±4.3	±1.5

Table 2. Turn rates

Mode	Pitch and yaw axes, mrad/s	Roll axis, mrad/s
Sun acquisition	±5.2	Not applicable
Roll search	Not applicable	-4.4
Commanded turn	±3.14 (yaw only)	±3.14

Table 3. ACS weight summary (nominal)

Assembly	Weight, kg	(lbs)
Sun sensors	0.454	(1.0)
Canopus tracker	4.2	(9.3)
Inertial reference unit	4.4	(9.8)
Reaction control assemblies (dry)	11.7	(25.8)
Attitude control gas	2.4	(5.4)
Gimbal actuators	2.8	(6.2)
Attitude control electronics	3.7	(8.2)
Total ACS	28.9	(65.7)

Table 4. Typical average power requirements[a]

Operating mode	Canopus Tracker 2.4 kHz	IRU 2.4 kHz/400 Hz	Gimbal actuators 30 Vdc	ACE[b] 2.4 kHz	Subtotal 2.4 kHz	Total ACS
Launch	0	13/9	6	18	31	46
Sun acquisition	0	13/9	0	9[c]	22	31
Roll search	5	13/9	0	8	26	35
Celestial cruise	5	0/0	0	4	9	9
All-axes inertial	5	13/9	6	17	35	50
Roll inertial	5	7/3	0	10	22	25
Commanded turn	5	17/9[d]	6	17	39	54
TVC	5	13/9	6	19[e]	37	52

[a]All requirements expressed in watts.

[b]Includes approximately 1/4 W for Sun sensors and a negligible amount for jet valves (but see notes c and e).

[c]In addition, 5 to 15 W may be required to operate jet valves during the time required for initial spacecraft rate reductions, nominally 10 to 20 s.

[d]The 2.4-kHz power includes 4 W associated with the commanded turn bias.

[e]Includes 2 W for jet valves.

Table 5. Performance parameters for the cruise sun sensors

Parameter	Specification
Nominal scale factor in linear region	4.75 Vdc/deg[a]
Combined mechanical to electrical null offset relative to mounting plane	<0.08 deg
Rise time to step input	<10 ms
Output noise, peak-to-peak	<50 mV

[a] 1.51 V/deg into switching amplifier summing resistor of 82.5 kΩ

Table 6. Roll error signal characteristics

Parameter	Value
Nominal scale factor in linear region	4 Vdc/deg
Null pointing error	<0.05 deg
Time constant	<0.5 s
Output noise, peak-to-peak	<0.2 V

Table 7. Canopus acquired signal characteristics

Parameter	Value
Nominal output for acquired condition	4.5 Vdc
Nominal output for nonacquired condition	0 Vdc
Output noise, peak-to-peak	< 1 V

Table 8. Single axis IRU rate errors (3σ)

Mode	Mission estimate, deg/h	Measured day to day, deg/h
Commanded turn	1.5	0.5
Inertial modes	0.5	0.05

Table 9. Accelerometer parameters

Parameter	Value
Pulse scaling (nominal)	0.03 m/s
Bias offset (3σ mission estimate)	150 μg
Scale factor error (3σ mission estimate)	0.2%
Maximum acceleration capability	4.5 m/s^2

Table 10. Gas requirement estimates

Source of gas consumption	Full-gas system, kg (lb)	Half-gas system, kg (lb)
Cruise		
Limit cycling		
Transit (200 days)	0.170 (0.368)	0.101 (0.222)
Orbit (90 days)	0.061 (0.136)	0.036 (0.078)
Stray light	0.037 (0.082)	0.009 (0.020)
Gravity gradient	0.020 (0.046)	0.020 (0.046)
Scan platform stepping (10 slews/orbit)	0.321 (0.708)	0.321 (0.708)
Leakage (3 cm^3/h/valve)	0.297 (0.655)	0.149 (0.328)
Subtotal	0.906 (1.995)	0.636 (1.402)
Discrete events		
Initial rate reduction	0.003 (0.006)	0.003 (0.006)
Acquisitions (6)	0.012 (0.026)	0.012 (0.026)
Roll searches (6)	0.010 (0.023)	0.010 (0.023)
Roll overrides (50)	0.088 (0.193)	0.088 (0.193)
Commanded turns (6)	0.035 (0.077)	0.040 (0.089)
Motor burns (20 min total)	0.006 (0.014)	0.006 (0.014)
Subtotal	0.154 (0.339)	0.159 (0.351)
Total	1.06 (2.33)	0.795 (1.75)

Table 11. Nominal RCA parameters[a]

Parameter	Pitch	Yaw	Roll
Spacecraft moments of inertia, kg-m^2 (slug-ft^2)			
Launch to orbit insertion	441 (324)	465 (342)	545 (401)
After orbit insertion	359 (264)	379 (279)	462 (340)
Effective moment arms for jet valves, m (in.)	3.26 (128.5)	3.26 (128.5)	3.05 (120.0)
Thrust per valve, N (mlb)	0.033 (7.4)	0.033 (7.4)	0.074 (16.6)
Torque per valve couple N-m (in.-lb)	0.215 (1.9)	0.215 (1.9)	0.452 (4.0)
Angular acceleration[b] (mrad/s^2)			
Launch to orbit insertion	0.49	0.46	0.83
After orbit insertion	0.60	0.57	0.98
Effective thrusting time per pulse (ms)	23	23	21
Angular velocity increment per pulse[b] (μrad/s)			
Launch to orbit insertion	11	11	17
After orbit insertion	14	13	21

[a] These are preflight estimates. Postflight estimates for certain of these parameters were as much as 50% different.

[b] Per thrust valve couple.

Table 12. Gimbal actuator performance parameters

Parameter	Value
Backlash with 2.27-kg (5-lb) load	<0.010 cm (0.004 in.)
Null position relative to actuators center stroke	<0.005 cm (0.002 in.)
Response time	<25 ms

Table 13. Nominal performance parameters for the RCA electronics[a]

Parameter	Value
Rate to position gain	
Acquisition modes[b]	7 s (pitch and yaw) 12 s (roll)
Inertial modes	5 s (pitch and yaw) 4 s (roll)
Minimum on-time	20 ms
Derived rate	
Charging time constant	7.4 s
Discharging time constant	22 s
Maximum feedback to preamplifier	43 μA

[a]Position deadbands and turn rates are given in Subsection II-C.

[b]Sun acquisition and roll search.

Table 14. Nominal performance parameters for the TVC electronics

Parameter	Value
Forward filter amplifier	
Dc transfer gain	9.1 V/μA
Minimum saturated output	11 V
Corner frequency of lag	1.45 Hz
Path guidance loop	
Dc transfer gain	1.0
Minimum saturated output	11 V
Corner frequency of lag	0.0328 Hz
Actuator driver	
Dc transfer gain	1260 V/mA
Saturated output	±28 Vdc
Feedback compensation	$T_1 = 4.83$ ms
	$T_2 = 0.105$ ms
	$T_3 = 84$ μs
$\dfrac{K(T_1 S + 1)}{(T_2 S + 1)(T_3 S + 1)(T_4 S + 1)\left(\dfrac{S^2}{\omega^2} + \dfrac{2\zeta}{\omega} S + 1\right)}$	$T_4 = 35$ μs
	$\omega = 636.7$ Hz
	$\zeta = 0.47$
	$K = 1.4$ mA/cm (3.58 mA/in.)
Preaim command range	
A axis	±4 deg
B axis	±2 deg

Table 15. CC&S commands to the ACS

CC&S command number	Primary function and purpose
7A	Switching amplifiers enable backup to pyroarming switch
7B	1. Enable Canopus tracker power 2. Discontinue accelerometer captured loop mode 3. Switching amplifiers enable, backup
7C	Canopus tracker brightness low-gate step command
7D	Canopus tracker cone angle update (directly from CC&S to the Canopus tracker)
7E	1. Enable power to TVC electronics and gimbal servos 2. Clear preaim circuitry
7F	Signals a stray-light condition
7G	Inhibit gyro power
7M1	Turn on Gyro and accelerometer power
7M2	1. Disable celestial sensor outputs to switching amplifiers 2. All gyros to inertial mode
7M3	Sets positive turn polarity
7M4	Command roll turn
7M5	Command yaw turn
7M6	1. Disable pitch and yaw switching amplifiers 2. Enable path guidance circuitry and TVC power

Table 16. FCS command to the ACS

FCS command number	Function and purpose
DC12	Canopus tracker brightness low-gate step command (backup to signal 7C)
DC13	1. Inhibit CC&S commands 7M1 through 7M6 and 7E 2. Enable switching amplifiers and Canopus tracker power (backup to PAS, and signals 7A, 7B, and 7M1)
DC14	Reset DC-13, DC-61, DC-62, and DC-64
DC15	Canopus tracker high- and low-brightness gate override
DC17	Canopus tracker cone angle update (backup to CC&S signal 7D; directly from FCS to Canopus tracker)
DC18	1. Roll gyro to inertial mode 2. Inhibit roll error signal to switching amplifier 3. Enable Sun shutter power 4. Command +2 deg roll turn increments on subsequent transmission 5. Inhibit DC-21 pulse to Canopus tracker
DC19	Resets DC-15, DC-18, DC-20, DC-40, DC-63
DC20	1. Disable Canopus tracker power 2. Turn off gyro power if Sun is acquired 3. Inhibit roll error signal to switching amplifier

Table 16 (contd)

FCS command number	Function and Purpose
DC20 (contd)	4. Enable Sun shutter power 5. Inhibit Canopus tracker adaptive low-gate step command
DC21	1. Transfer to Canopus search, provided DC-18 signal is absent 2. Transmission while in roll inertial (by DC-18, signal 7M2, or signal 7F) shall produce -2-deg roll turn increments.
DC40	Inhibit gyro power
DC61	Backup to the Sun gate signal
DC62	1. Inhibits ACE preaim registers 2. Enable selection of fixed preaim bias
DC63	Turn on roll gyros; override signal 7G commanded function from CC&S
DC64	Select postinsertion preaim backup bias

Table 17. Sun sensor logic equations

Equation No.	Logical expression
(1)	Alternate Sun acquired signal: ASG = SG + DC61F
(2)	Cruise Sun sensor disable = A7M2
(3)	Acquisition Sun sensor disable = A7M2 + ASG

Table 18. Canopus tracker logic equations

Equation No.	Logical expression	Where mechanized
	Canopus tracker power enable:	
(4)	$CTPWR = \overline{DC20F} \cdot (7B + 7M1 + DC13F)$	ACE[a]
	Sun shutter power enable:	
(5)	$A = \overline{ASG}$ + other terms	ACE
(6)	Demodulator enable $= CA \cdot LG + DC15F$	CT[b]
(7)	Low-gate step $= DC12 + X \cdot 7C$	ACE
(8)	$X\ SET = \overline{X\ RESET} \cdot 7C$	ACE
(9)	$X\ RESET = PSUCA + 7F + A + \Delta ACEPWR$	ACE
(10)	$HGen\ SET = \overline{CA} \cdot LG$	CT
(11)	$HGen\ RESET = CA \cdot \Delta ADEL \cdot RNULL$	CT
(12)	$ANULL = RNULL \cdot HGen$	CT

[a]Attitude control electronics assembly.

[b]Canopus tracker assembly.

[c]These are logical equivalents to Eq. (29), but are not mechanized as such.

Table 18 (contd)

Equation No.	Logical expression	Where mechanized
	Canopus acquired signal:	
(13)	$CA = (\overline{D} + LGDEL) \cdot E + \Delta CTPWR$	CT
(14)	$E\ SET = \overline{LGDEL}$	CT
(15)	$E\ RESET = HG \cdot HGen + DC21 \cdot \overline{DC18F}$	CT
(16)	$\overline{D}\ SET = CA \cdot \overline{LG} \cdot LGDEL$	CT
(17)	$\overline{D}\ RESET = \Delta\overline{D}DEL$	CT
	Alternate Canopus acquired signal:	
(18)	$ACA = CA \cdot CTPWR + DC15F \cdot RECA$	ACE
(19)	Cone angle update $= 7D + DC17$	CT
(20)	Roll override bias $= DC21 \cdot \overline{DC18F}$	ACE
(21)	Search bias enable $= \overline{CA} + \overline{LG} + DC15F$	CT
(22)	Fly back bias enable $= \left[\Delta FYBSW + \overline{C}\right] \cdot \overline{DC15F} \cdot \overline{DC21 \cdot \overline{DC18F}}$	CT
(23)	$\overline{C} = CA \cdot \overline{LG} \cdot LGDEL$	CT
(24)	$PSUCA\ SET = ACA \cdot \overline{7F}$	ACE
(25)	$PSUCA\ RESET = \overline{ACA} \cdot \overline{7F}$	ACE
(26)	$Z\ SET = \overline{ACA} \cdot 7F \cdot PSUCA$	ACE

Table 18 (contd)

Equation No.	Logical expression	Where mechanized
(27)	$\text{FYBSW SET} = Z \cdot \overline{\text{ACA}} \cdot \overline{\text{7F}}$	ACE
(28)	$\text{Z RESET} = \text{FYBSW RESET} = \text{ACA} \cdot \overline{\text{7F}} + \Delta\text{ACEPWR}$	ACE
	Canopus tracker integrator enable:	
(29)	$\text{CTEN} = (\text{PSUCA} + \overline{\text{7F}}) \cdot \overline{\text{A}}$	ACE
(30)	$\text{A} = \text{RSI} + \text{DC20F} + \overline{\text{ASG}}$	ACE
	$+ (\text{PSUCA} \cdot \text{7F} + \text{DC18F} + \text{A7M2} + \text{FYBSW}) \cdot \text{GPWRONDE}$	
(31)	RSI SET = roll CCW jet valves on continuously for over 30 s	ACE
(32)	$\text{RSI RESET} = \text{DC21} \cdot \overline{\text{DC18F}} + \text{A7M1} + \Delta\text{ACEPWR}$	ACE
(33)	$\text{CTEN} = \overline{\text{RSI}} \cdot \overline{\text{DC20F}} \cdot \text{ASG} \cdot \left[\overline{\text{DC18F}} \cdot \overline{\text{A7M2}}\right.$	(footnote c)
	$\left.\overline{\text{FYBSW}} \cdot \overline{\text{7F}} + \overline{\text{GPWRONDE}} \cdot (\text{PSUCA} + \overline{\text{7F}})\right]$	
(34)	$\overline{\text{CTEN}} = \text{RSI} + \text{DC20F} + \overline{\text{ASG}}$	(footnote c)
	$+ (\text{DC18F} + \text{A7M2} + \text{FYBSW}) \cdot \text{GPWRONDE}$	
	$+ (\overline{\text{PSUCA}} + \text{PSUCA} \cdot \text{GPWRONDE}) \cdot \text{7F}$	

Table 19. Inertial reference unit logic equations

Equation No.	Logical expression
	Gyro power logic signal:
(35)	$\mathrm{GPWR} = \left[\overline{\mathrm{SWAMPR}} + \overline{\mathrm{ASG}} + (\overline{\mathrm{ACA}} + \mathrm{7F}) \cdot \overline{\mathrm{DC20F}} + \mathrm{A7M1} + \mathrm{A7M2} + \mathrm{DC18F} + \mathrm{DC63F}\right] \cdot (\overline{\mathrm{7G}} + \mathrm{DC63F}) \cdot \overline{\mathrm{DC40F}}$
(36)	$\overline{\mathrm{GPWR}} = \mathrm{SWAMPR} \cdot \mathrm{ASG} \cdot \left[\mathrm{ACA} \cdot \overline{\mathrm{7F}} + \mathrm{DC20F}\right] \cdot \overline{\mathrm{A7M1}} \cdot \overline{\mathrm{A7M2}} \cdot \overline{\mathrm{DC18F}} \cdot \overline{\mathrm{DC63F}} + \mathrm{7G} \cdot \overline{\mathrm{DC63F}} + \mathrm{DC40F}$
(37)	Roll gyro power on $= \mathrm{INVPWR}$
(38)	Pitch and yaw gyro power on $= \mathrm{INVPWR} \cdot \overline{\mathrm{RONY}}$
(39)	$\mathrm{RONY\ SET} = (\overline{\mathrm{ACA}} + \mathrm{7F} + \mathrm{DC18F} + \mathrm{DC63F}) \cdot \overline{\mathrm{GPWRD}}$
(40)	$\mathrm{RONY\ RESET} = (\mathrm{ACA} \cdot \overline{\mathrm{7F}} \cdot \overline{\mathrm{DC18F}} \cdot \overline{\mathrm{DC63F}}) \cdot \overline{\mathrm{GPWRD}} + \overline{\mathrm{ASG}} + \mathrm{A7M1} + \mathrm{A7M2}$
(41)	Pitch and yaw gyro integrators enable $= \mathrm{A7M2}$
(42)	Roll gyro integrator enable $= \mathrm{RI}_{en} = (\mathrm{A7M2} + \mathrm{PSUCA} \cdot \mathrm{7F} + \mathrm{FYBSW} + \mathrm{DC18F}) \cdot \mathrm{GPWRONDE}$

Table 19 (contd)

Equation No.	Logical expression
(43)	Positive turn select = A7M3
(44)	Negative turn select = $\overline{\mathrm{7M3}} \cdot \overline{\mathrm{DC13}}$
(45)	Roll turn execute = A7M4
(46)	Yaw turn execute = $\overline{\mathrm{7M4}} \cdot \mathrm{A7M5}$
(47)	+2 deg roll increment = DC18 · GPWRONDE
(48)	-2 deg roll increment = DC21 · GPWRONDE · RI_{en}
(49)	Accelerometer capture loop disable = 7B + 7M1 + DC13F

Table 20. Switching amplifier logic equations

Equation No.	Logical expression
(50)	Pitch and yaw derived rate disable $= \mathrm{INVPWR} \cdot \overline{\mathrm{RONY}}$
(51)	Roll derived rate disable $= \mathrm{INVPWR}$
(52)	Roll switching amplifier enable:
	$\mathrm{SWAMPR} = \mathrm{PAS} + 7\mathrm{A} + 7\mathrm{B} + 7\mathrm{M1} + \mathrm{DC13F}$
(53)	Pitch and yaw switching amplifier enable:
	$\mathrm{SWAMPPY} = \mathrm{SWAMPR} \cdot \overline{\mathrm{A7M6}}$

Table 21. TVC logic equations

Equation No.	Logical expression
(54)	TVC power enable:
	$\mathrm{TVCPWR} = \mathrm{A7E} + \mathrm{A7M1} + \overline{\mathrm{SWAMPR}} \cdot \mathrm{GPWRONDE}$
(55)	Preinsertion preaim backup bias $= \mathrm{DC62F} \cdot \overline{\mathrm{DC64F}}$
(56)	Postinsertion preaim backup bias $= \mathrm{DC62F} \cdot \mathrm{DC64F}$
(57)	Normal preaim bias $= \overline{\mathrm{DC62F}}$
(58)	Path guidance enable $= \mathrm{A7M6}$
(59)	TVC enable $= \mathrm{SWAMPR}$

Table 22. Telemetry channel assignments[a]

Channel	Logic	Signal telemetered
105	$\mathrm{RONY} + \overline{\mathrm{INVPWR}}$	Cruise Sun sensors (pitch)
105	$\mathrm{INVPWR} \cdot \overline{\mathrm{RONY}} \cdot \overline{\mathrm{A7M2}}$	Gyro rate (pitch)
105	A7M2	Gyro integrator (pitch position)
106	$\mathrm{RONY} + \overline{\mathrm{INVPWR}}$	Cruise Sun sensor (yaw)
106	$\mathrm{INVPWR} \cdot \overline{\mathrm{RONY}} \cdot \overline{\mathrm{A7M2}}$	Gyro rate (yaw)
106	A7M2	Gyro integrator (yaw position)
107	$\overline{\mathrm{INVPWR}}$	Canopus tracker integrator output (fine scaling)
107	$\mathrm{INVPWR} \cdot \overline{\mathrm{RI}_{en}}$	Gyro rate (roll)
107	RI_{en}	Gyro integrator (roll position)
112	$\overline{\mathrm{TVCPWR}}$	Combined Sun sensors (pitch)
112	TVCPWR	Autopilot "A" gimbal position
113	$\overline{\mathrm{TVCPWR}}$	Combined Sun sensors (yaw)
113	TVCPWR	Autopilot "B" gimbal position

[a]Switched channels only; see Appendix A for unswitched channels.

Table 23. List of logic symbols in Section IV

Symbol	Short name	Subsection where defined
A	A switch	D-9
ACA	Alternate Canopus acquired	D-6
ANULL	roll error signal null	D-4-b
ASG	Alternate Sun gate	C-2
A7E, A7M_	Alternate CC&S commands	B
$\overline{C}$	Logic signal	D-7-c
CA	Canopus acquired	D-5
CTEN	Canopus tracker integrator enable	D-9
CTPWR	Canopus tracker power	D-1
$\overline{D}$	Logic signal	D-5
DC__	Direct command	I-1
DC__F	Direct command flip-flop	B
E	Logic signal	D-5
FYBSW	Fly back and sweep	D-8
GPWR	Gyro power signal	E-1
GPWRD	Gyro power turn on delay	E-1
GPWRONDE	Gyro power on delay	E-1
HG	High gate	D-4
HGen	High-gate enable	D-4-b
INVPWR	Gyro power turn on delay	E-1
LG	Low gate	D-4
LGDEL	Low gate delayed 10 ms	D-5

Table 23 (contd)

Symbol	Short name	Subsection where defined
PAS	Pyro arming switch	B
PSUCA	Pseudo Canopus acquired	D-8
RECA	Roll error Canopus acquired	D-6
RI_{en}	Roll gyro integrator enable	E-3
RNULL	Roll error null	D-4-b
RONY	Roll gyro only	E-2
RSI	Roll search inhibit	D-9
SG	Sun gate	C-2
SWAMPPY	Pitch and yaw switching amplifier enable	F-3
SWAMPR	Roll switching amplifier enable	F-3
TVCPWR	TVC power	G-1
X	Logic signal	D-4-a
Z	Logic signal	D-8
ΔACEPWR	ACS power on pulse	D-4-a
ΔADEL	ANULL delay pulse	D-4-b
ΔCTPWR	Canopus tracker power on pulse	D-5
$\Delta\overline{D}$DEL	$\overline{D}$ turn on delay pulse	D-5
ΔFYBSW	Fly back and sweep signal on pulse	D-7
7_, 7M_	CC&S commands	I-1

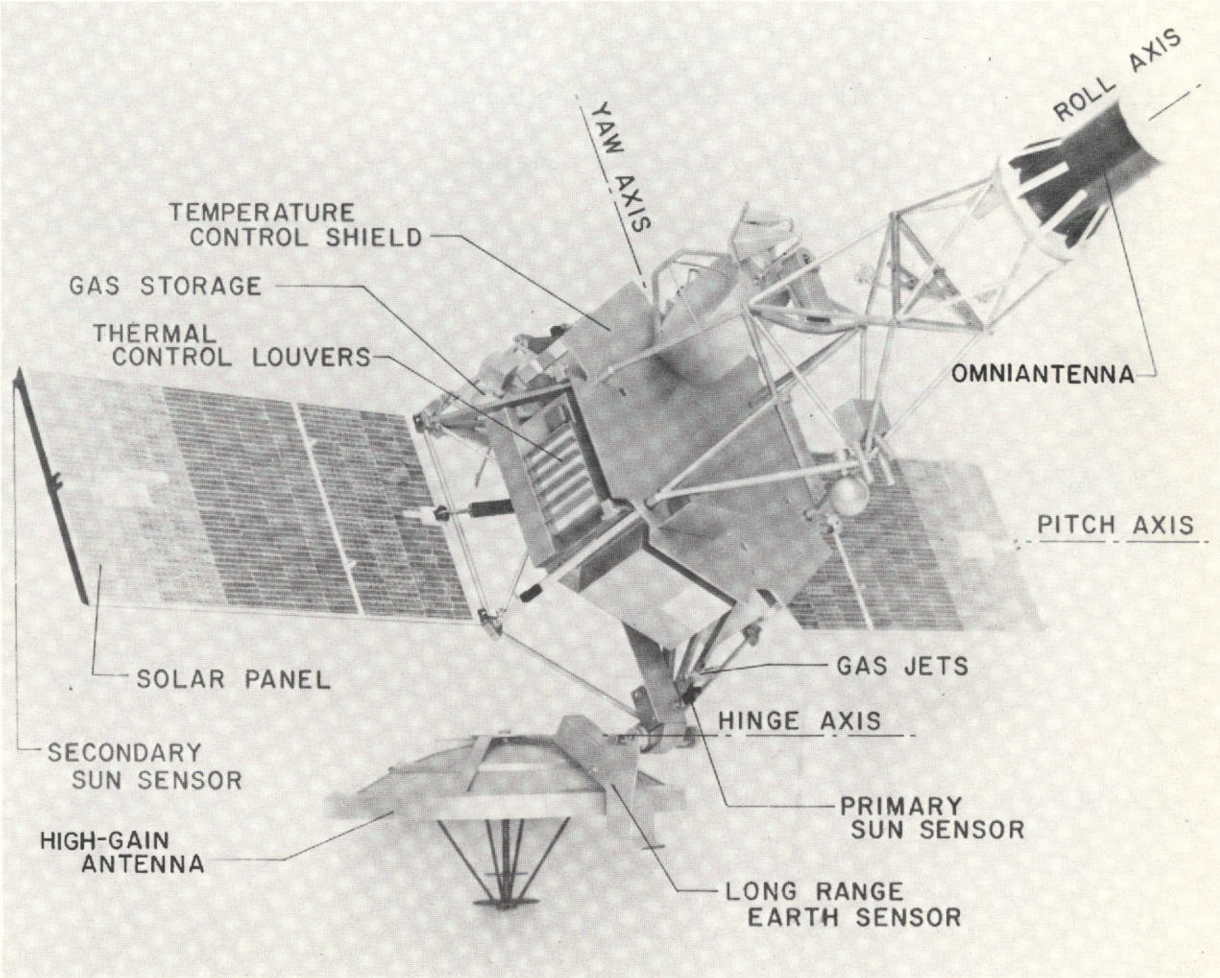


Fig. 1. Mariners 1 and 2

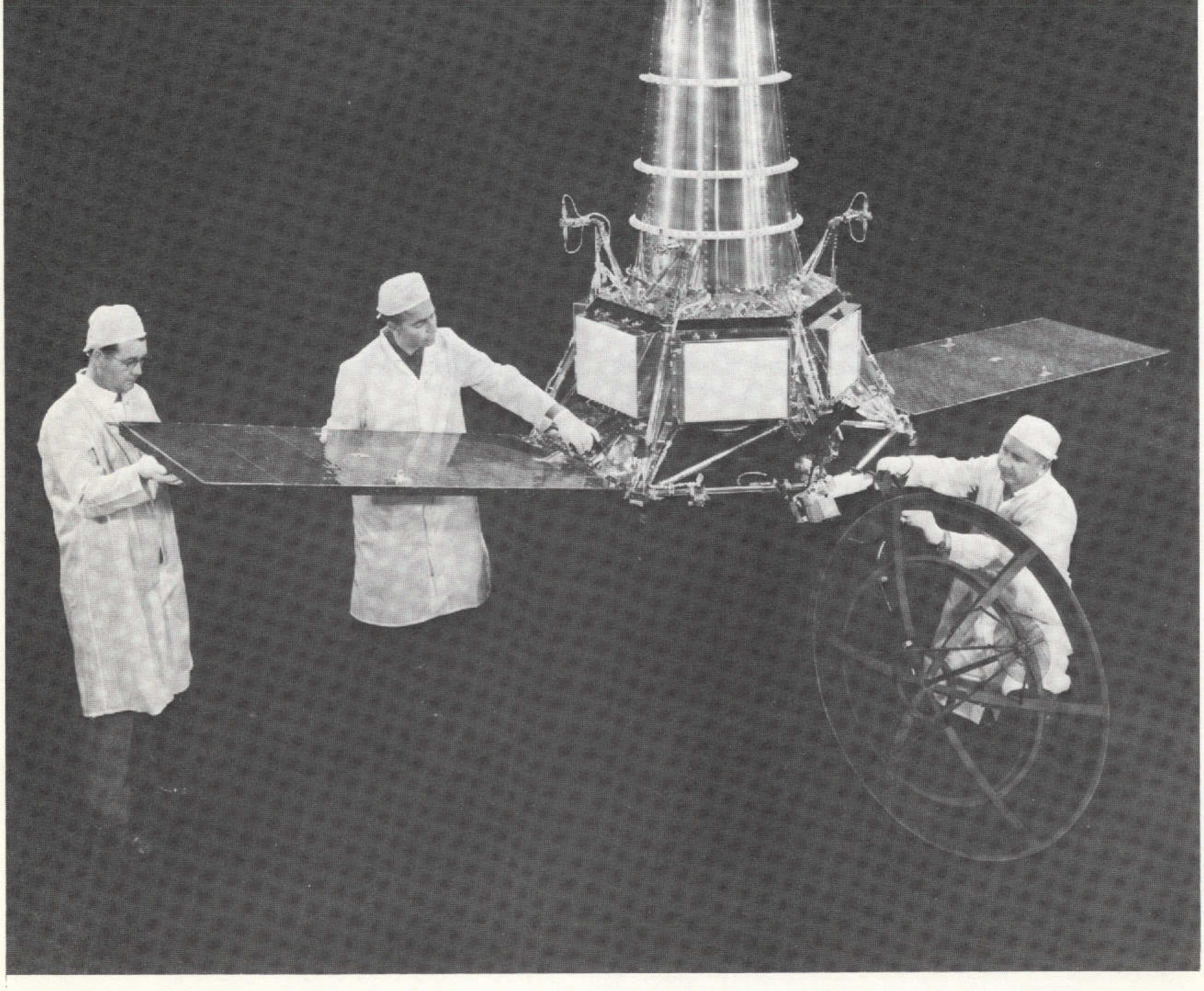


Fig. 2. Ranger Block III

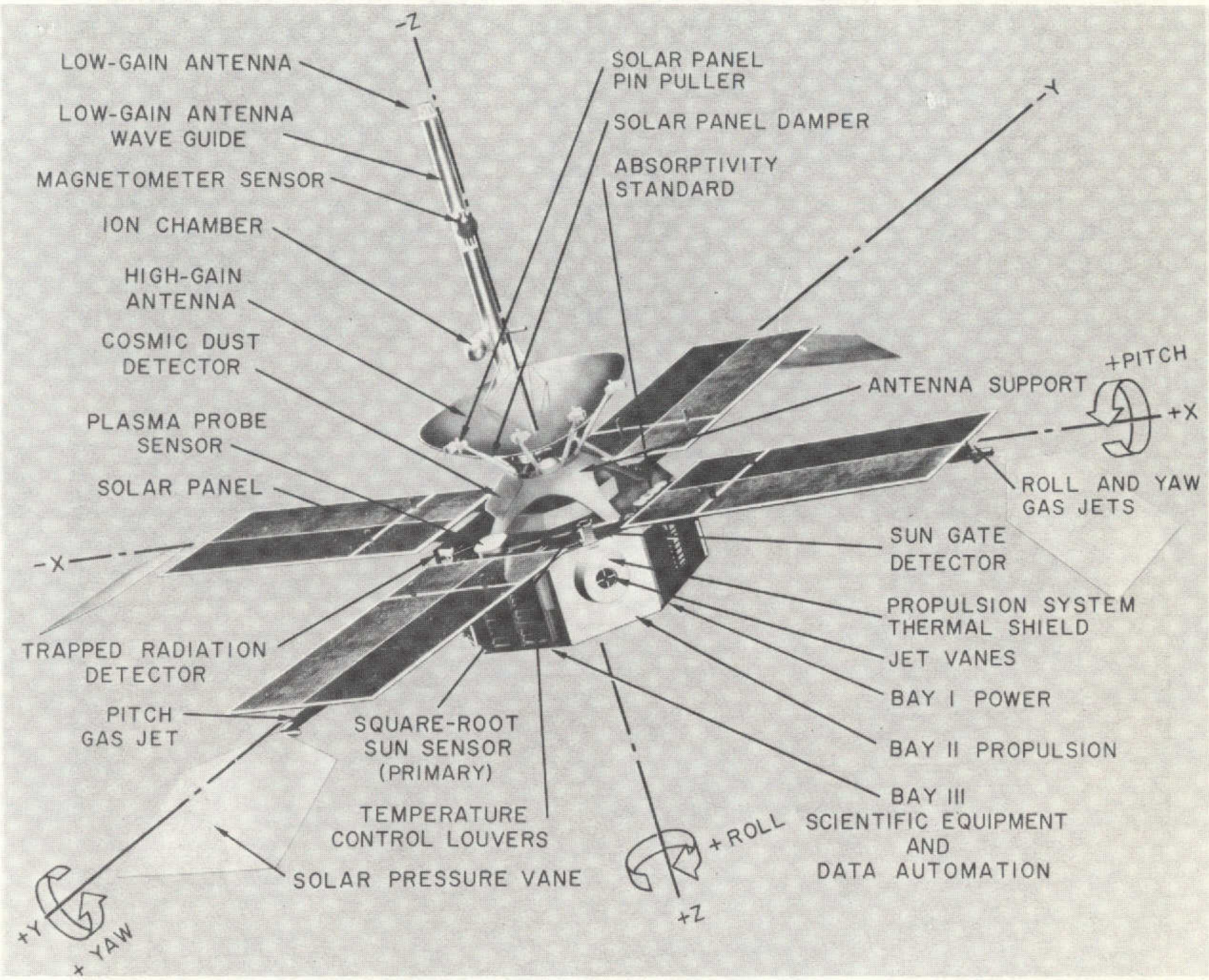


Fig. 3 Mariners 3 and 4

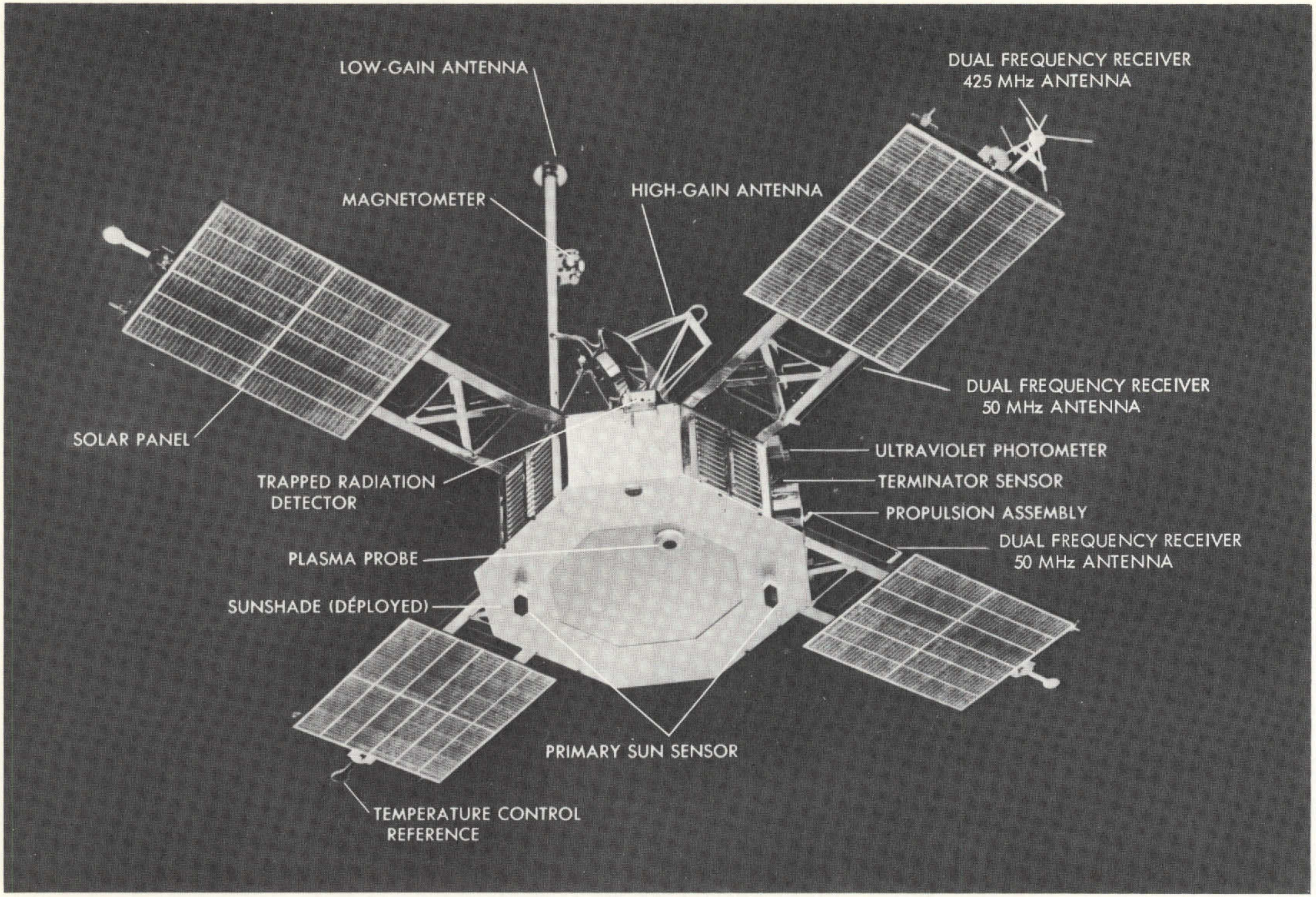


Fig. 4. Mariner 5

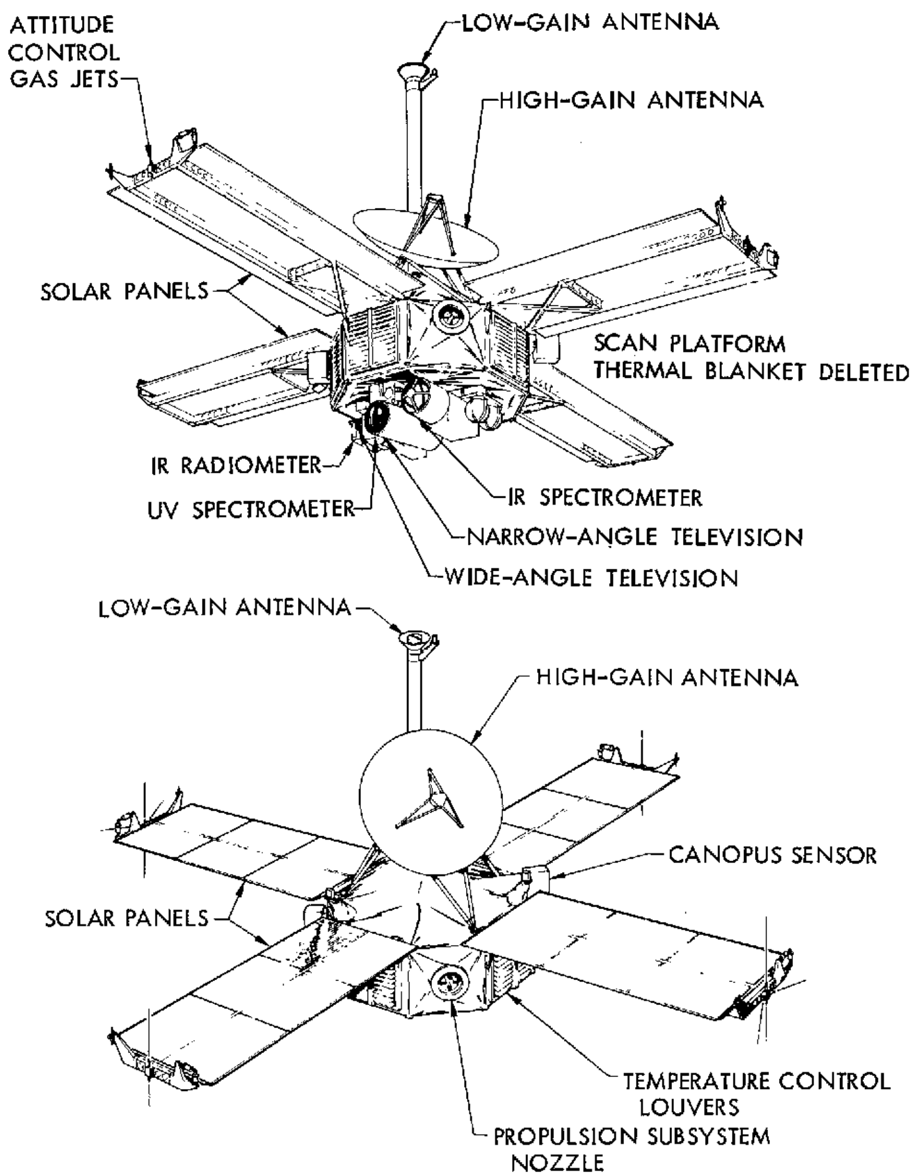


Fig. 5. Mariners 6 and 7

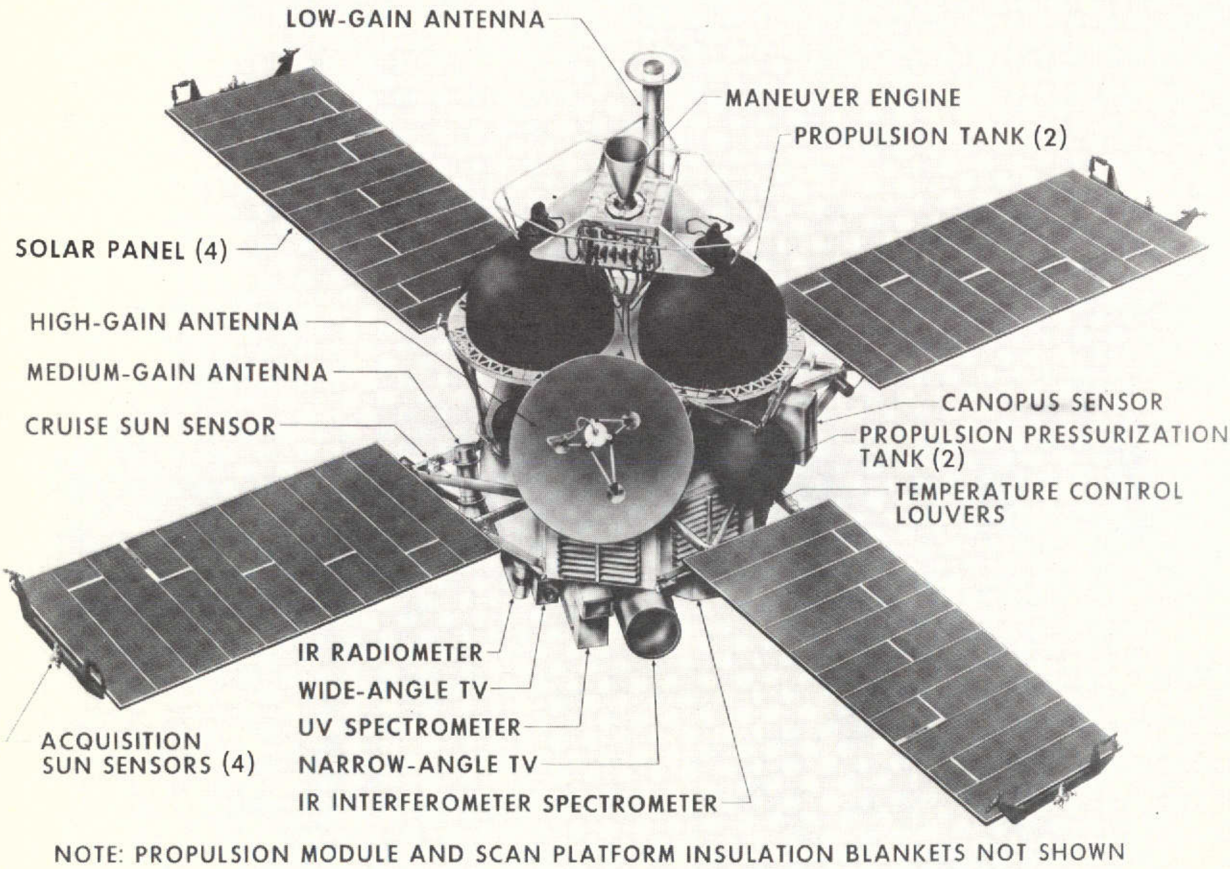


Fig. 6. Mariners 8 and 9

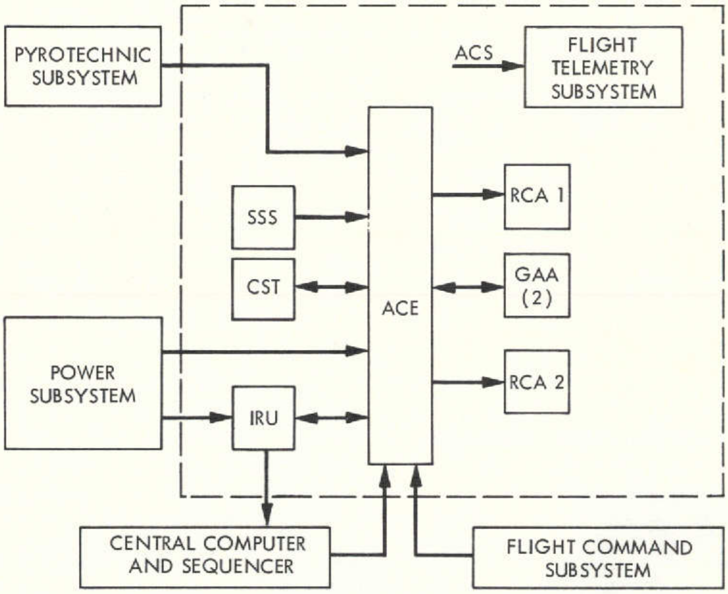


Fig. 7. Mariner Mars 1971 ACS block diagram

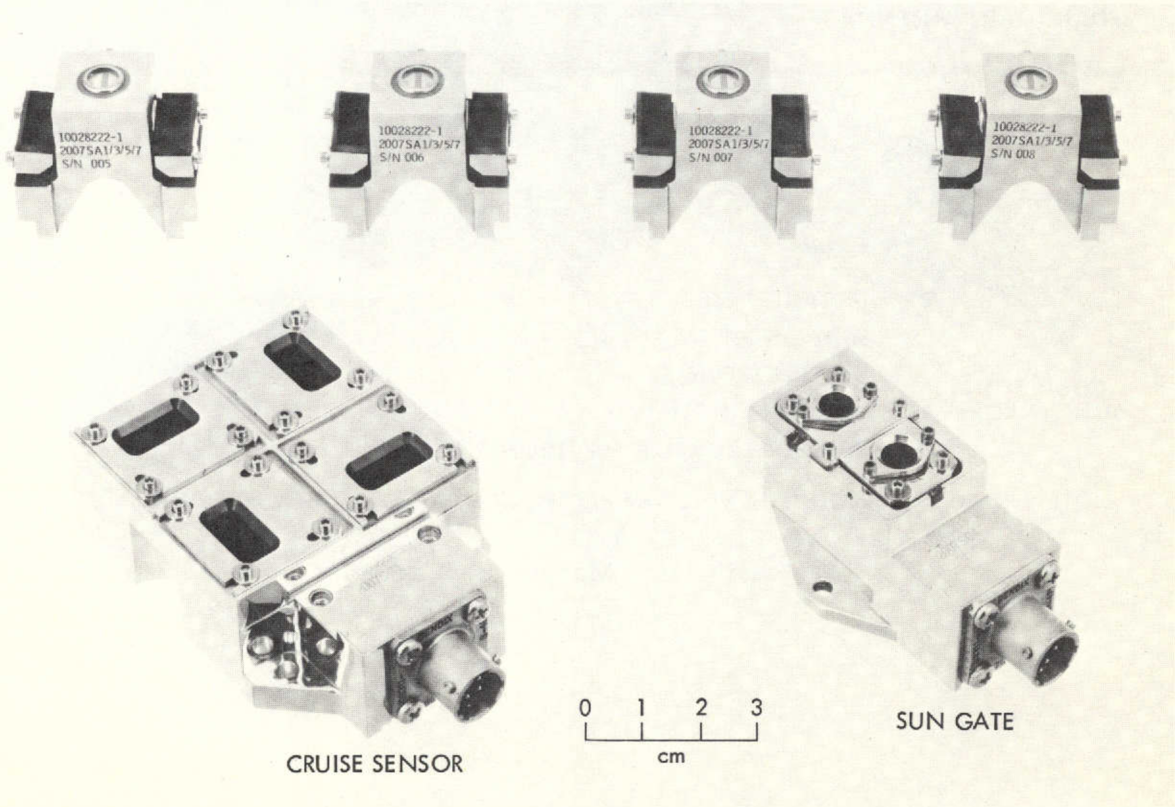
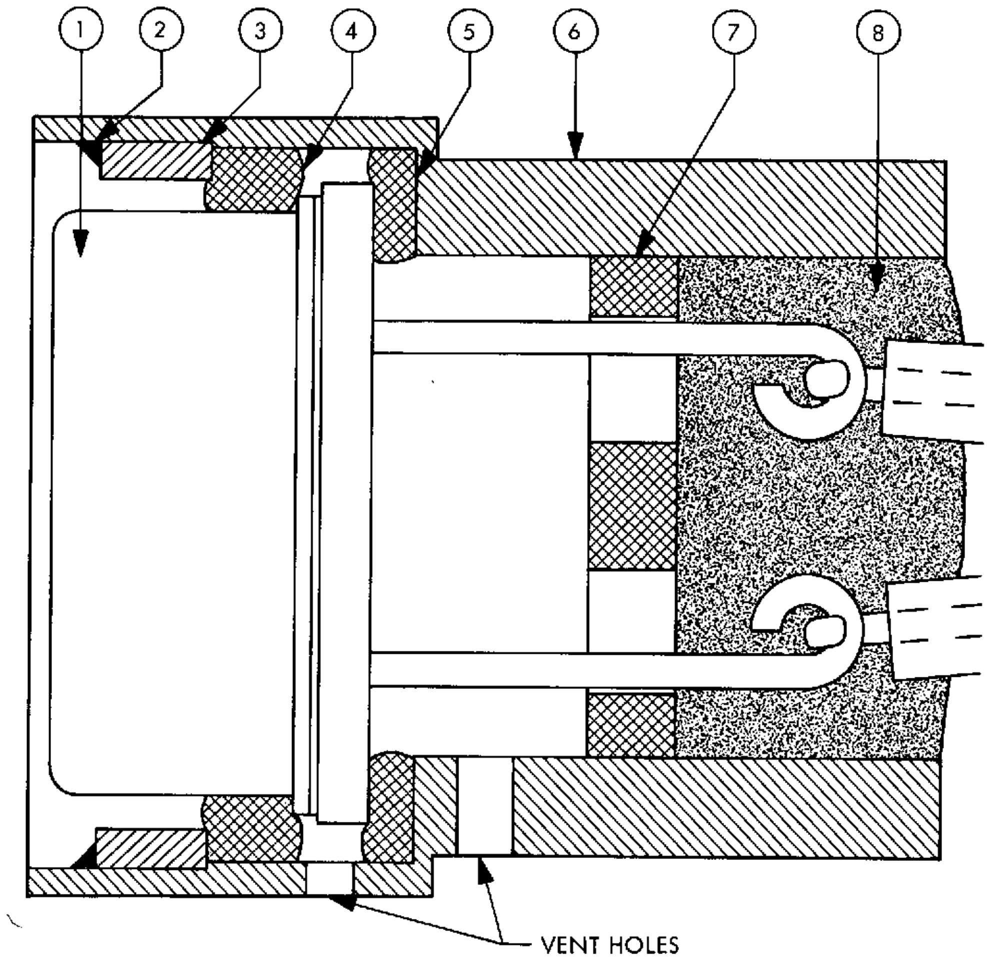


Fig. 8. Sun sensors for the Mariner 9 spacecraft



1	PHOTODETECTOR	5	LOWER ISOLATION WASHER
2	ADHESIVE	6	PHOTODETECTOR HOUSING
3	RETENSION RING	7	SEPARATION DISC
4	UPPER ISOLATION WASHER	8	POTTING COMPOUND

Fig. 9. Photodetector resilient-mount assembly

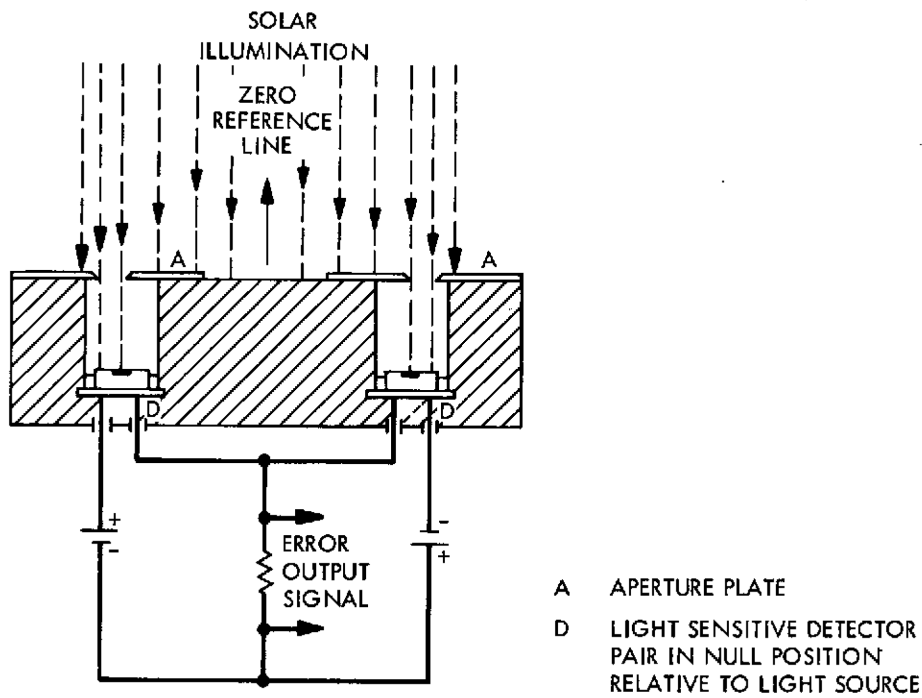


Fig. 10. Electro-optical schematic

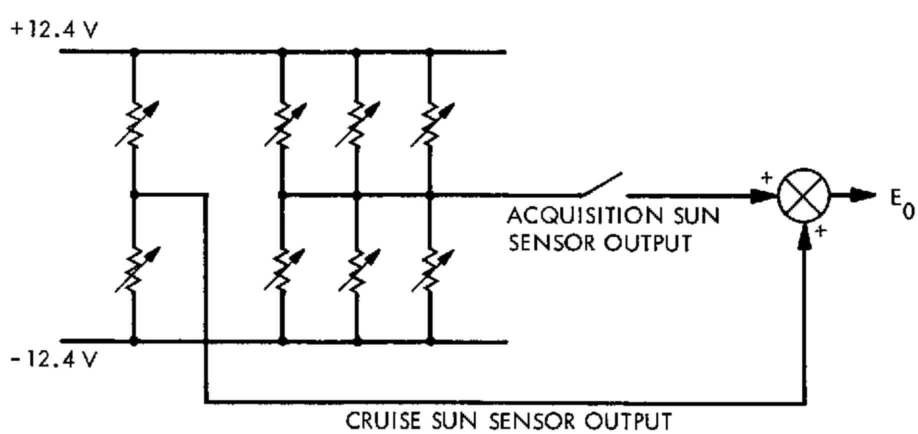


Fig. 11. Simplified Sun sensor circuit

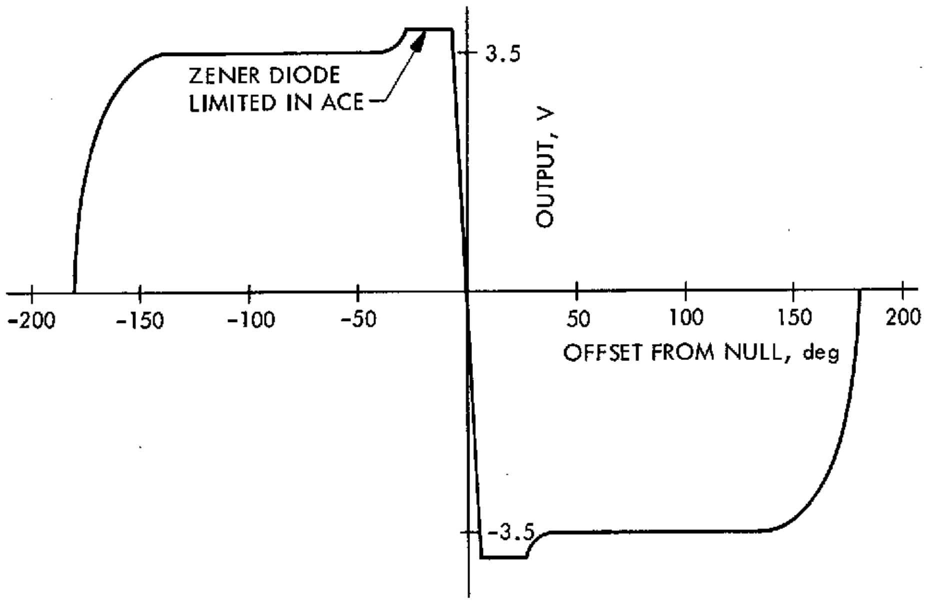


Fig. 12. Combined cruise and acquisition Sun sensor output, single axis

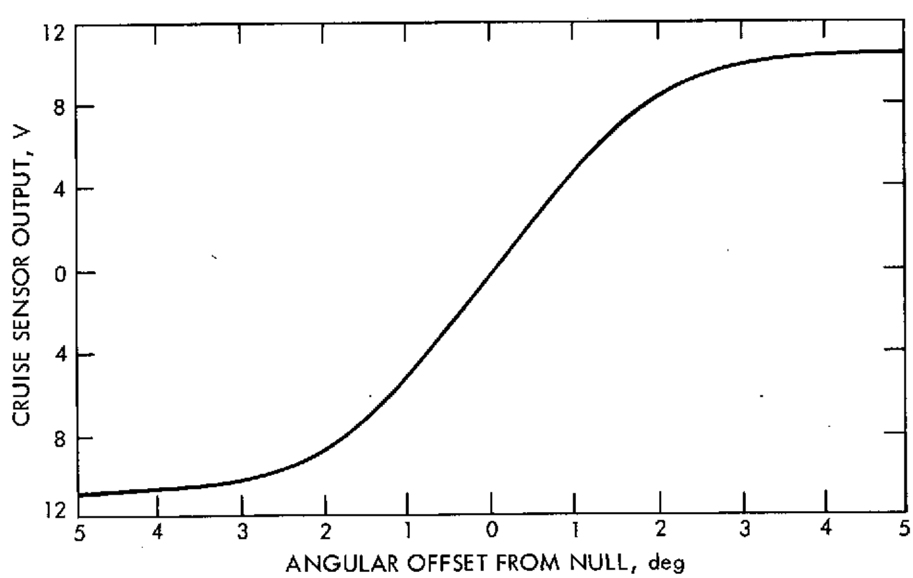


Fig. 13. Cruise sensor output, single axis

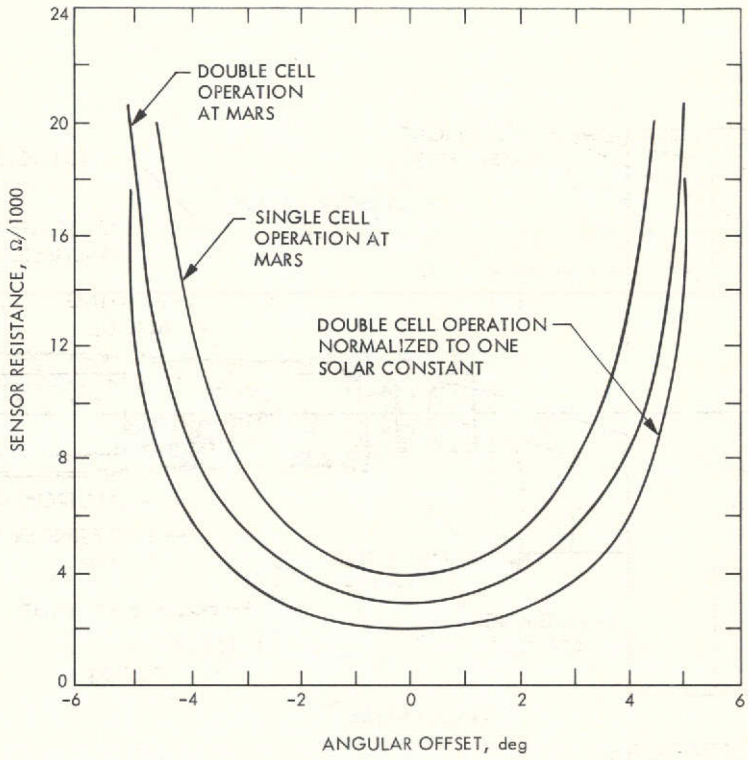


Fig. 14. Sun gate characteristics

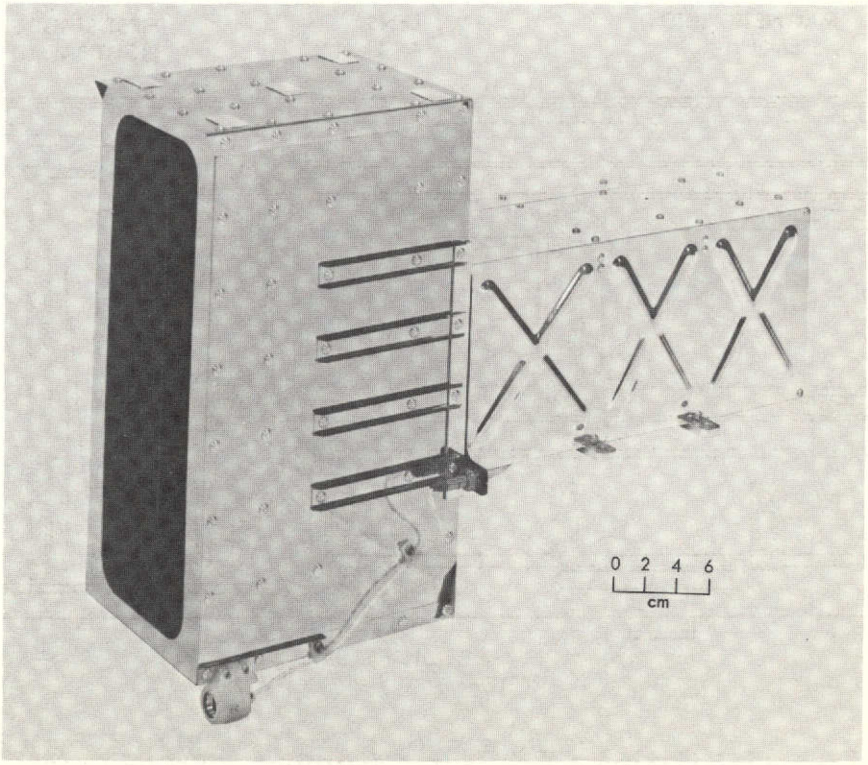


Fig. 15. Canopus Tracker Assembly

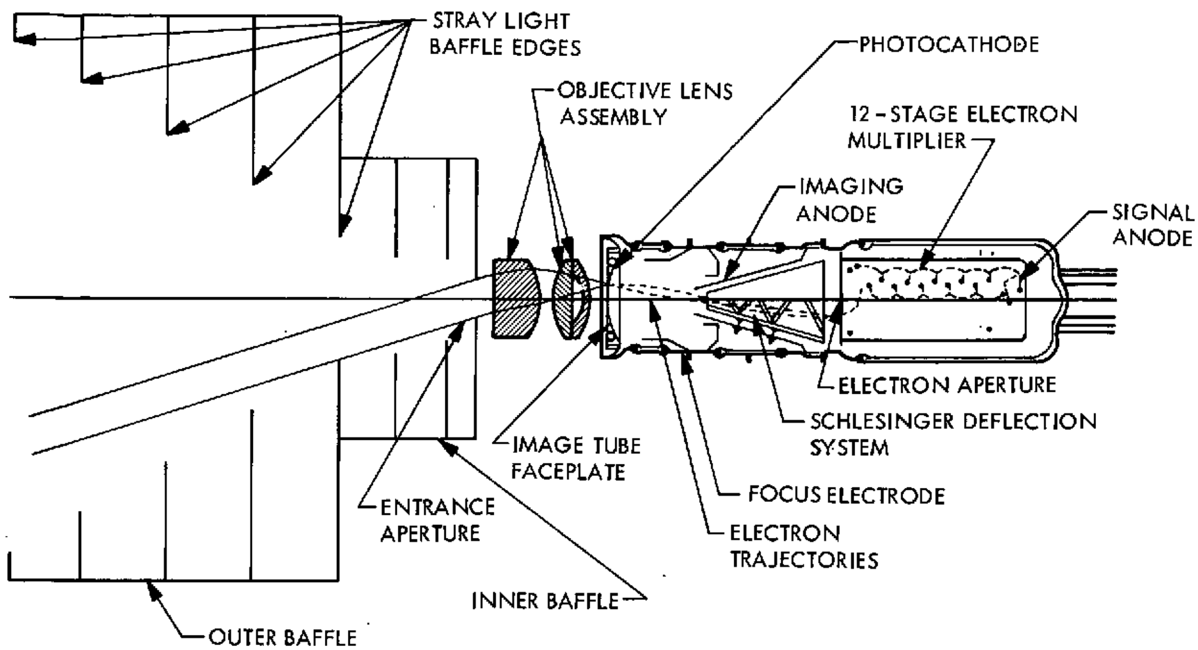


Fig. 16. Canopus tracker optical layout

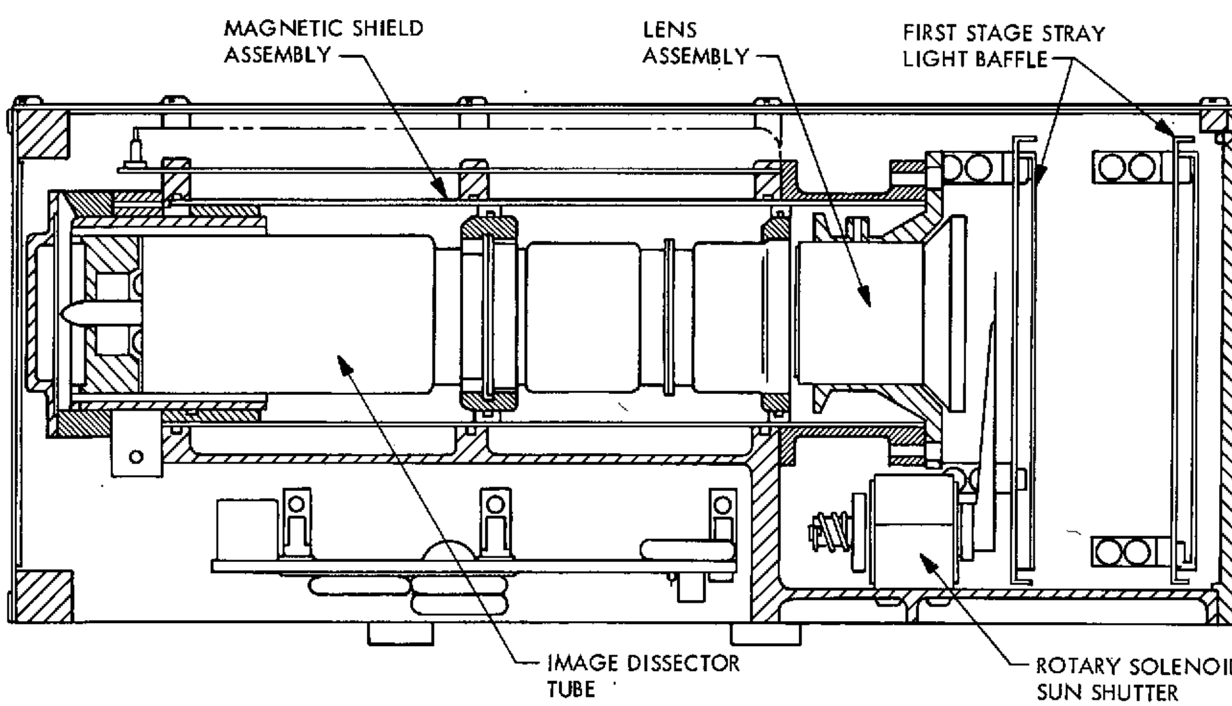


Fig. 17. Canopus Tracker Assembly, cutaway view

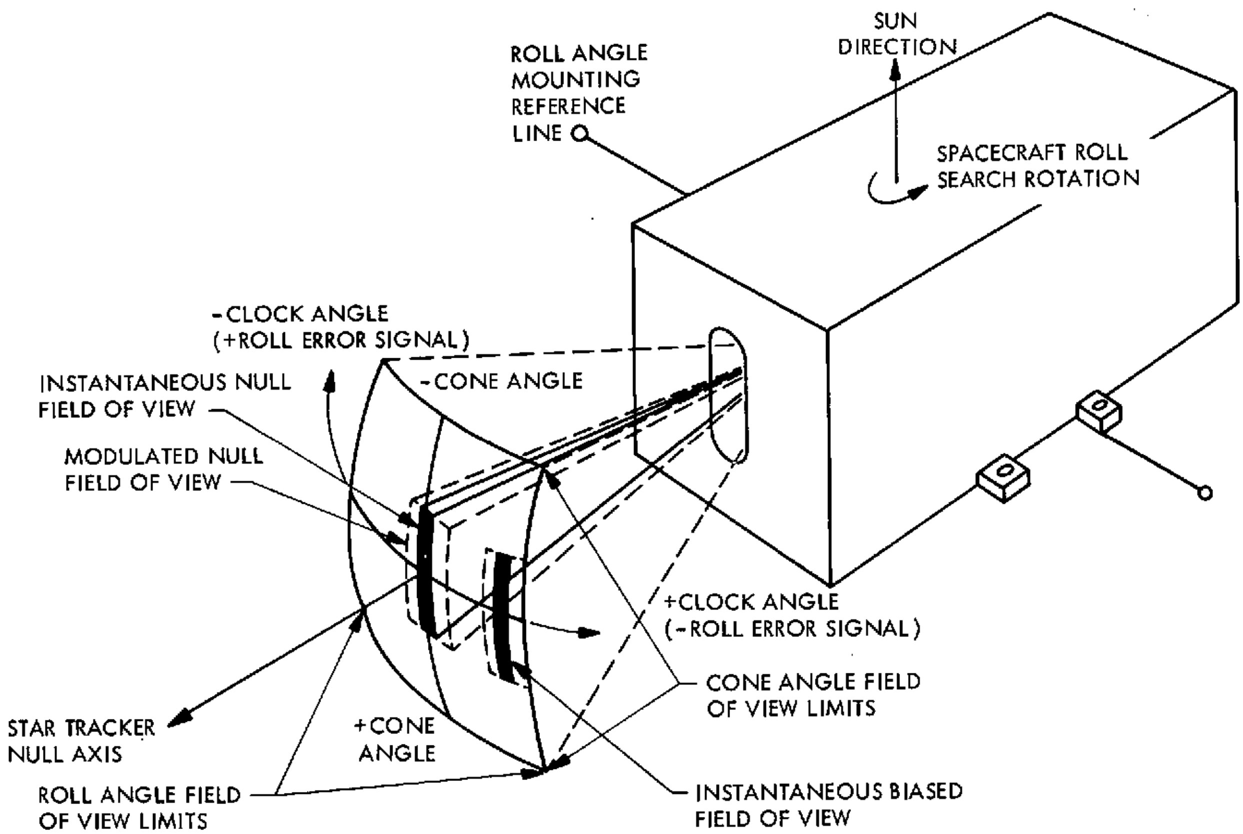


Fig. 18. Canopus tracker fields of view, three-dimensional diagram

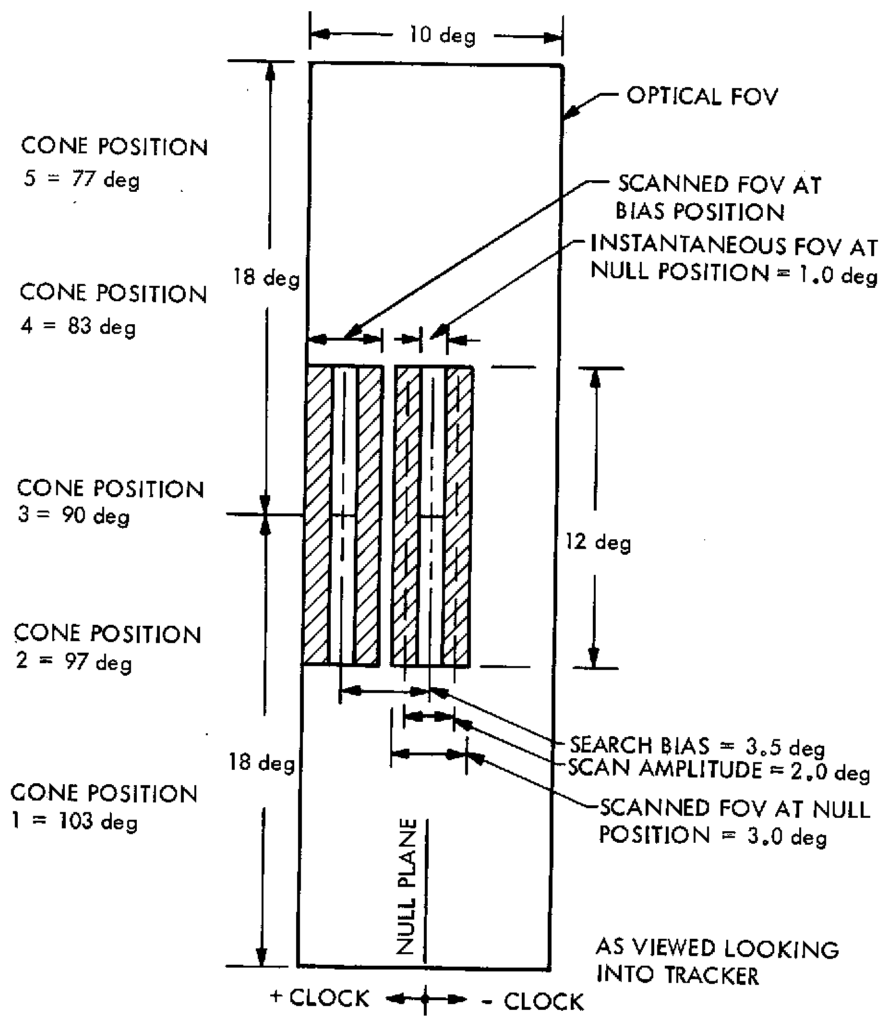


Fig. 19. Canopus tracker fields of view as viewed looking into tracker

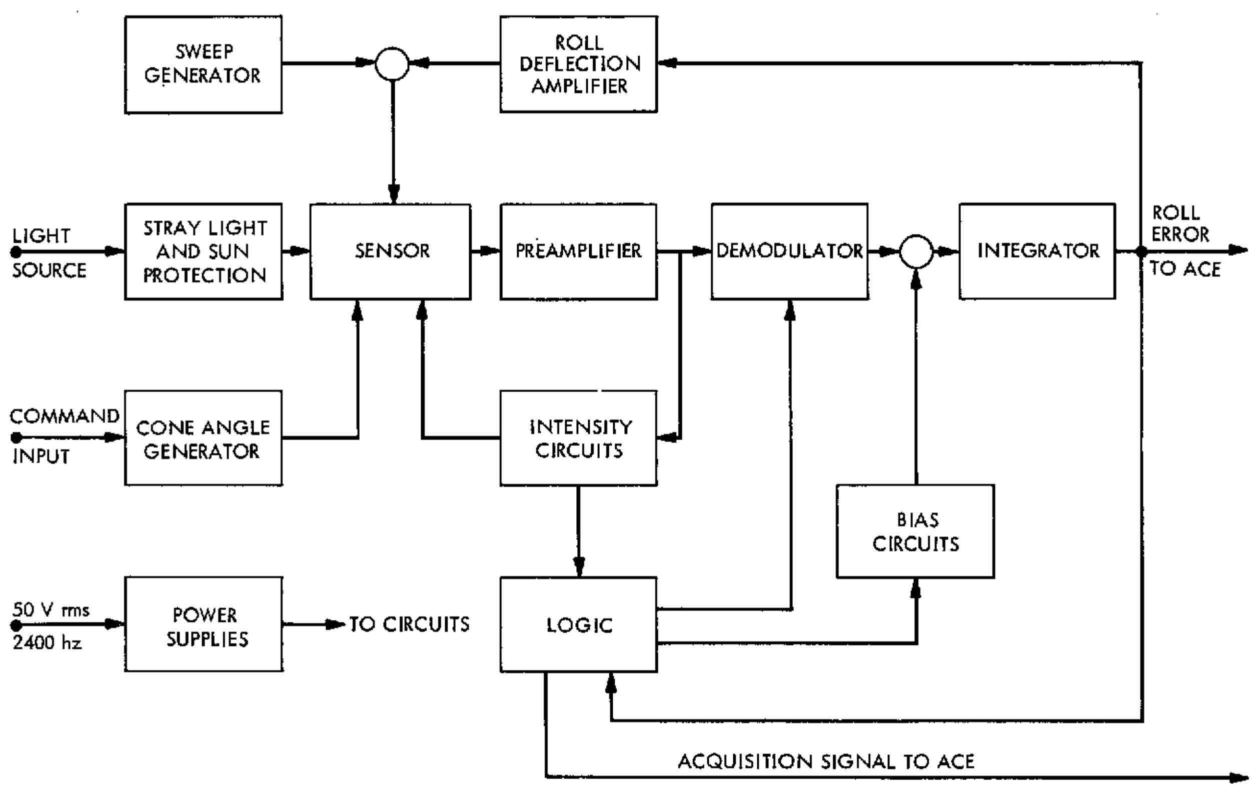


Fig. 20. Canopus Tracker Assembly, simplified block diagram

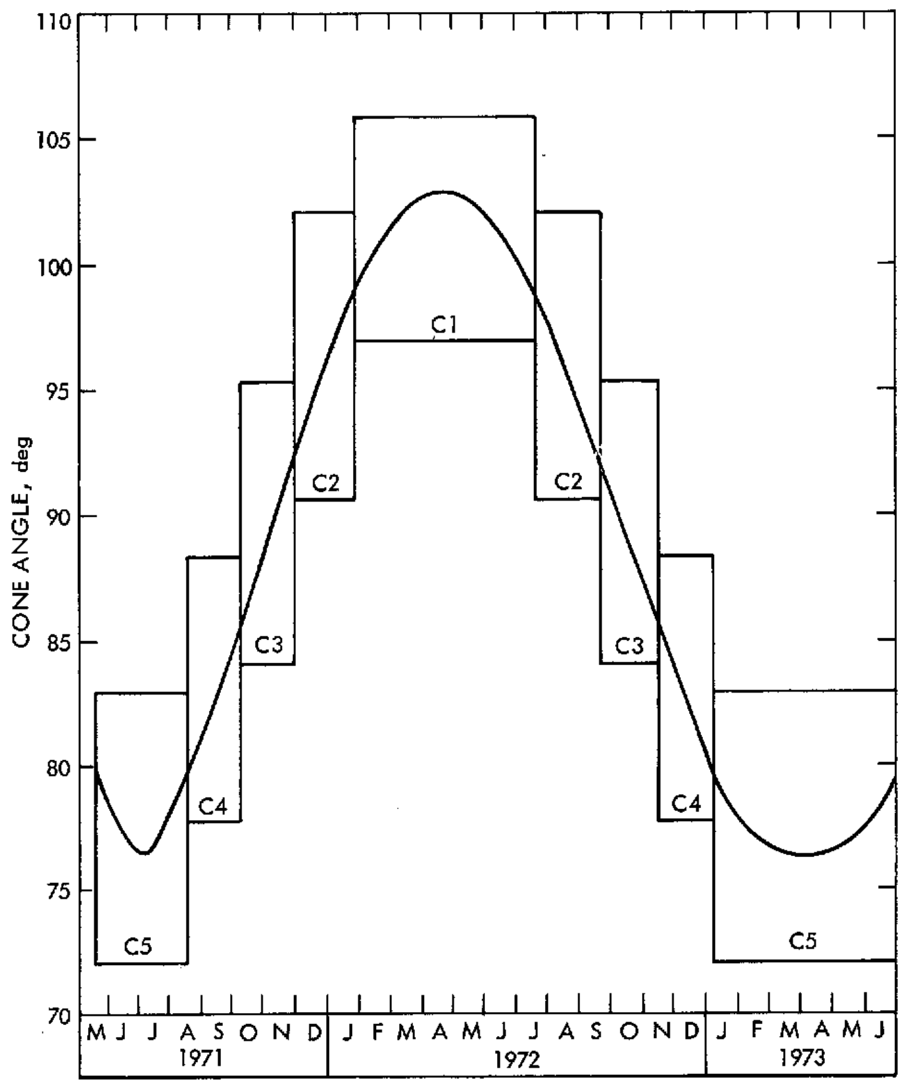


Fig. 21. Canopus-probe-Sun angle vs calendar date

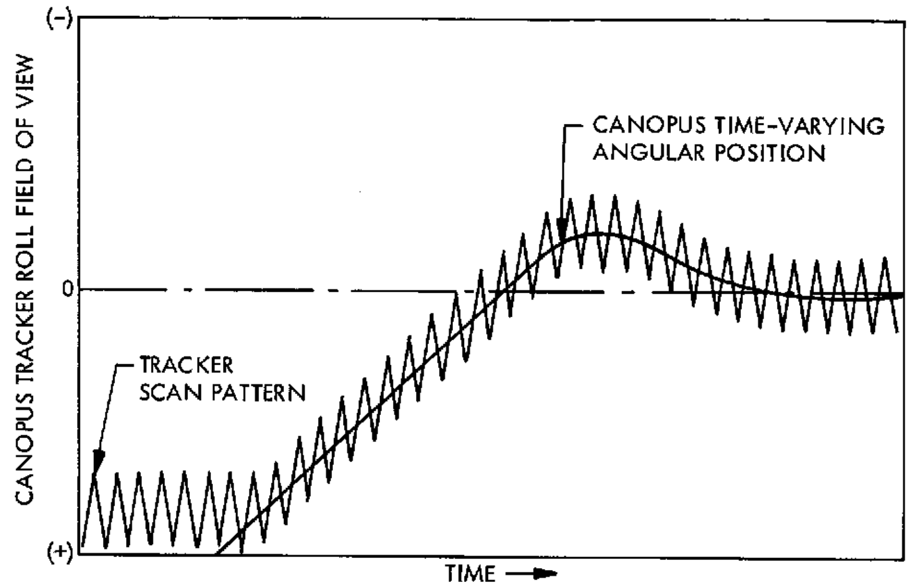


Fig. 22. Canopus tracker scan pattern during acquisition

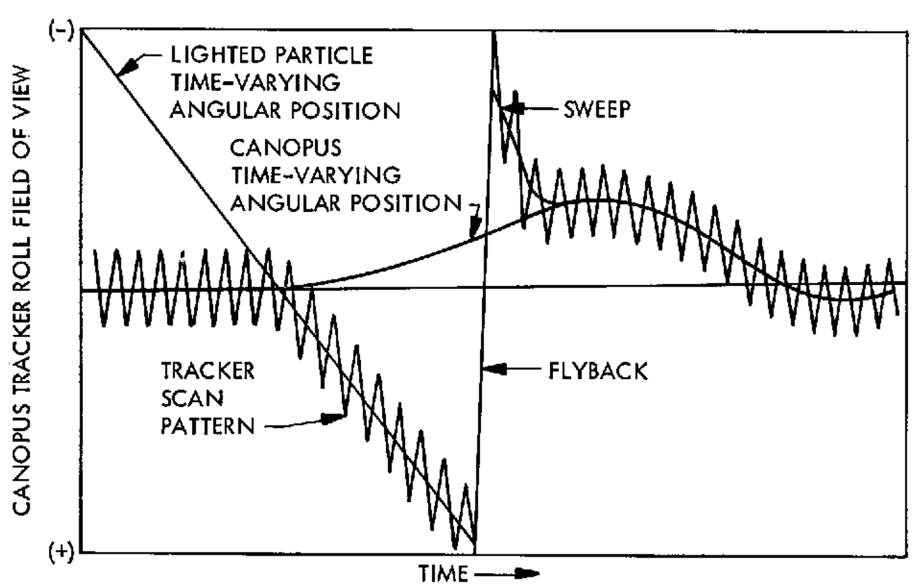


Fig. 23. Canopus tracker scan pattern when distracted by lighted particles

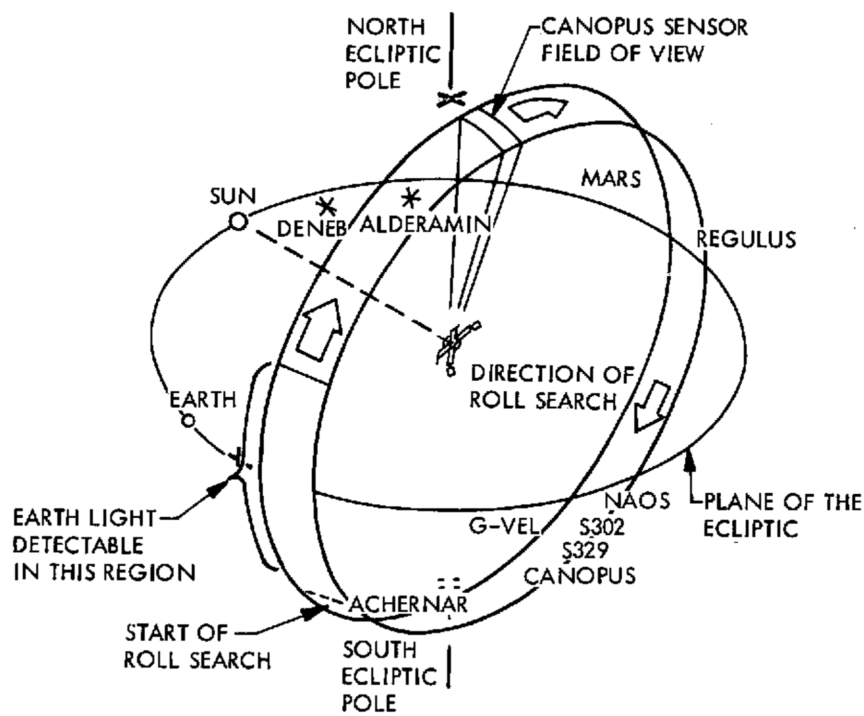


Fig. 24. Roll search pattern

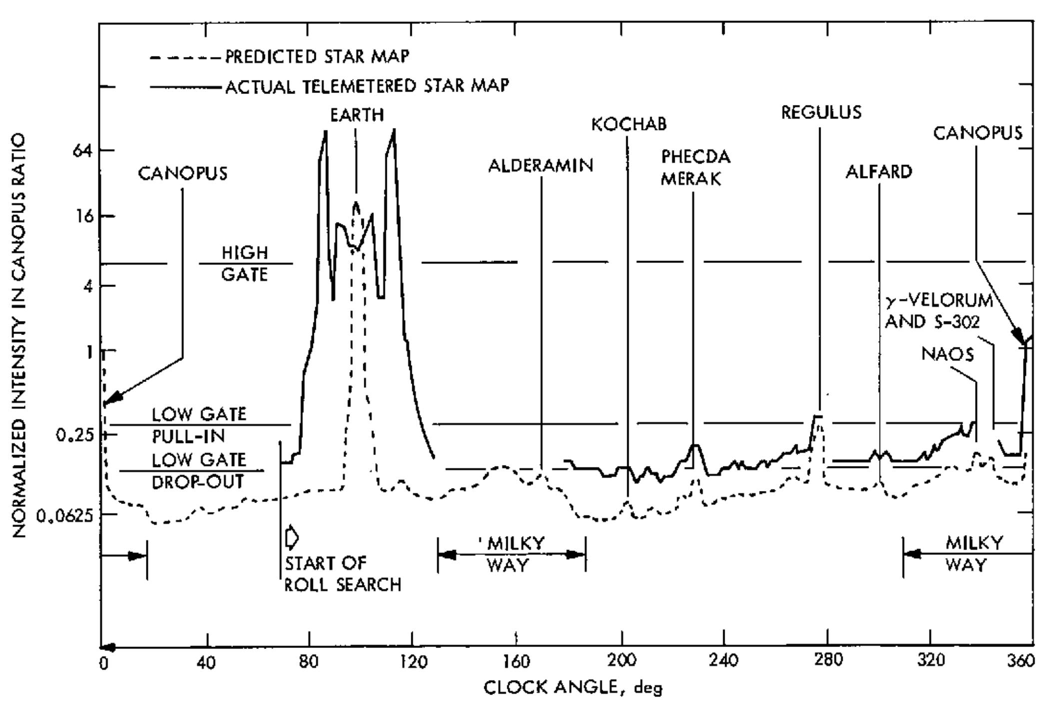


Fig. 25. Predicted and actual Canopus sensor output voltage vs clock angle at time of first roll search

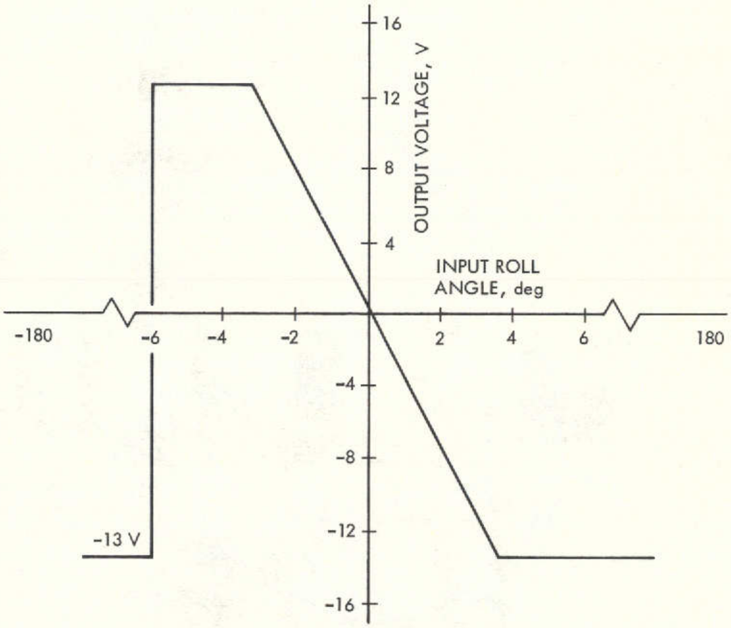


Fig. 26. Canopus tracker roll error output characteristic

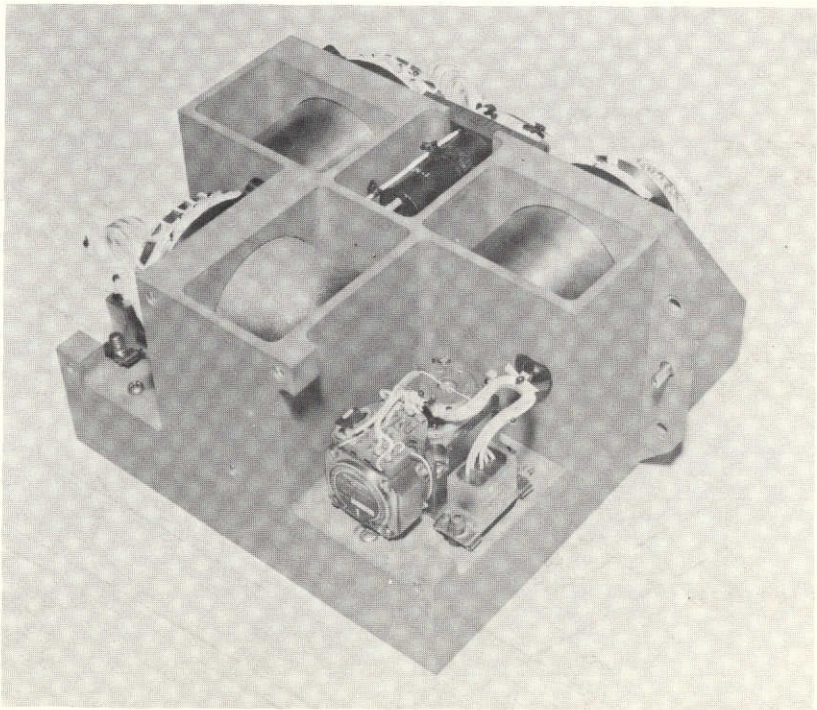


Fig. 27. Inertial sensors subassembly

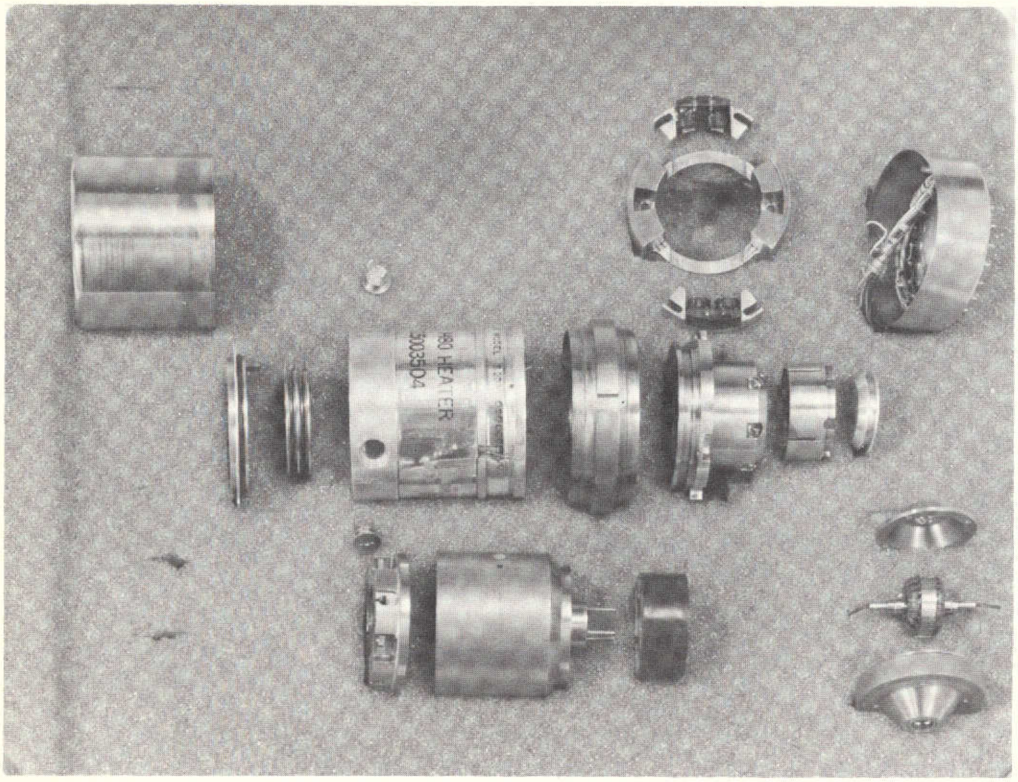


Fig. 28. Gyro cutaway

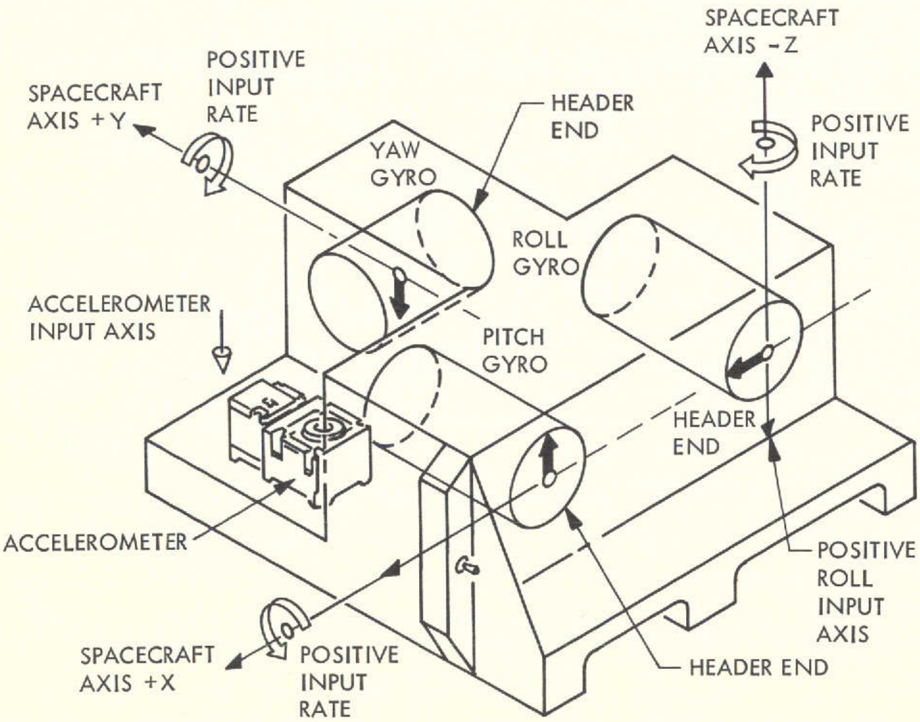


Fig. 29. Inertial sensors mounting configuration

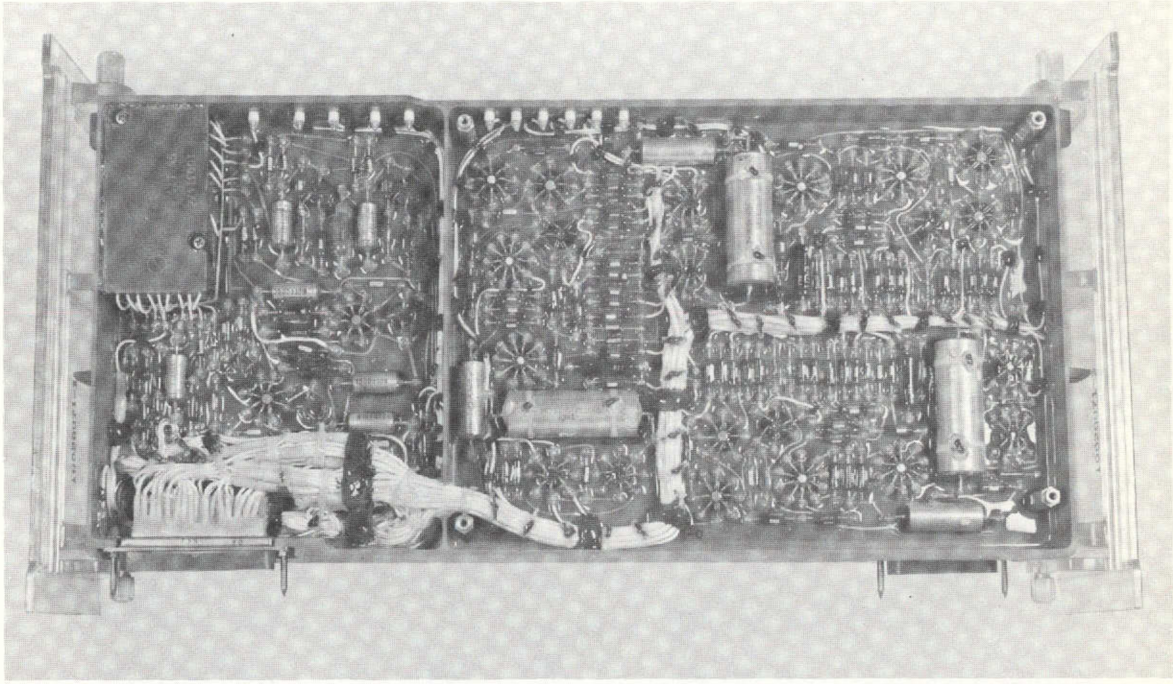


Fig. 30. Inertial electronics subassembly

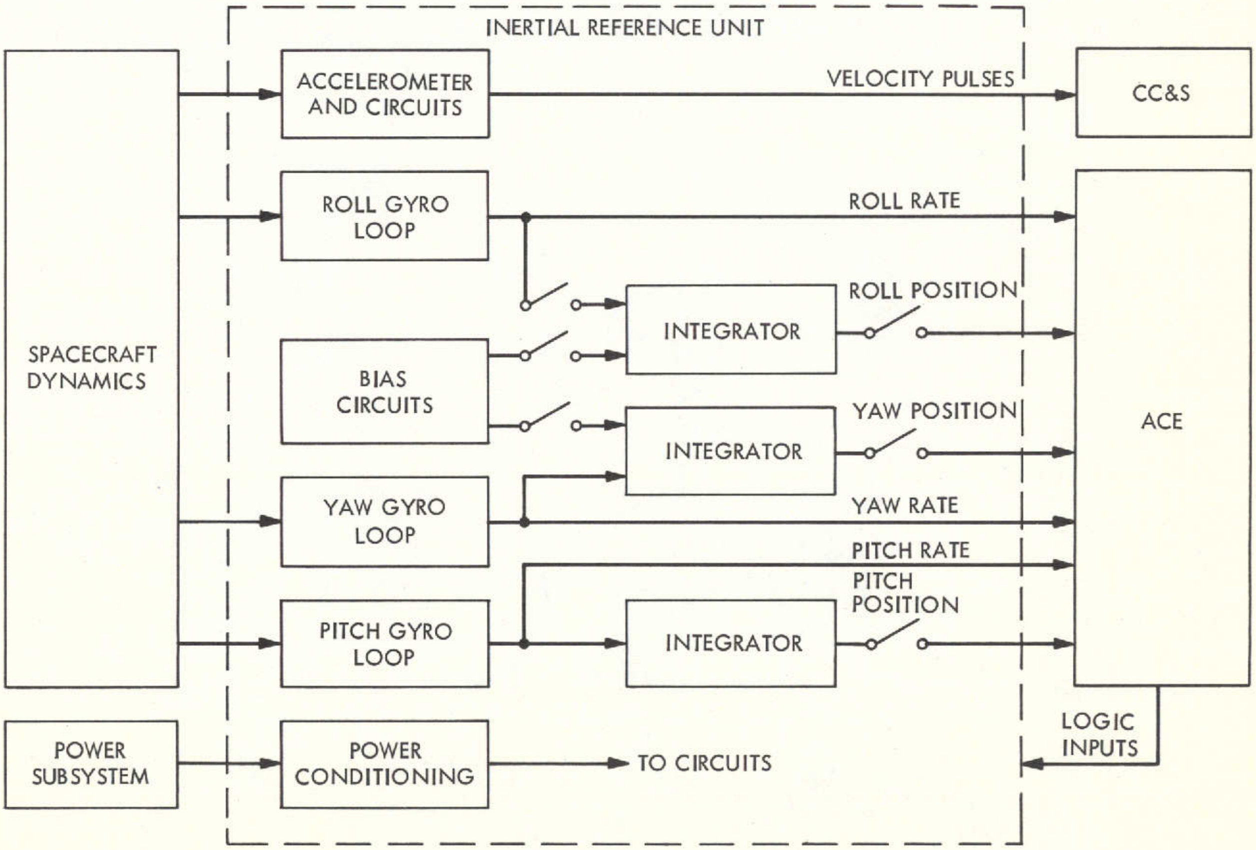


Fig. 31. Inertial reference unit functional block diagram

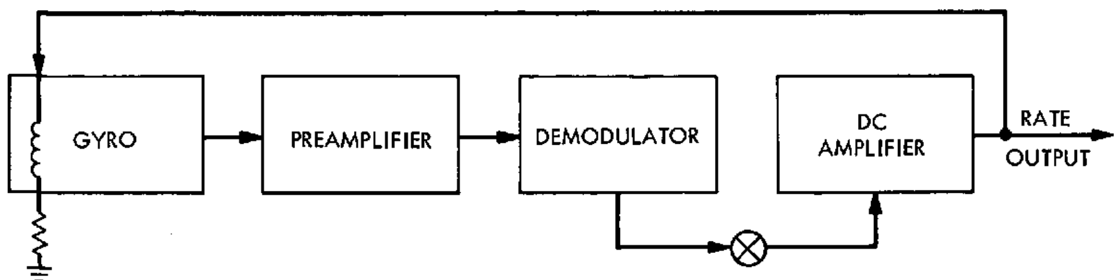


Fig. 32. Gyro loop, single axis

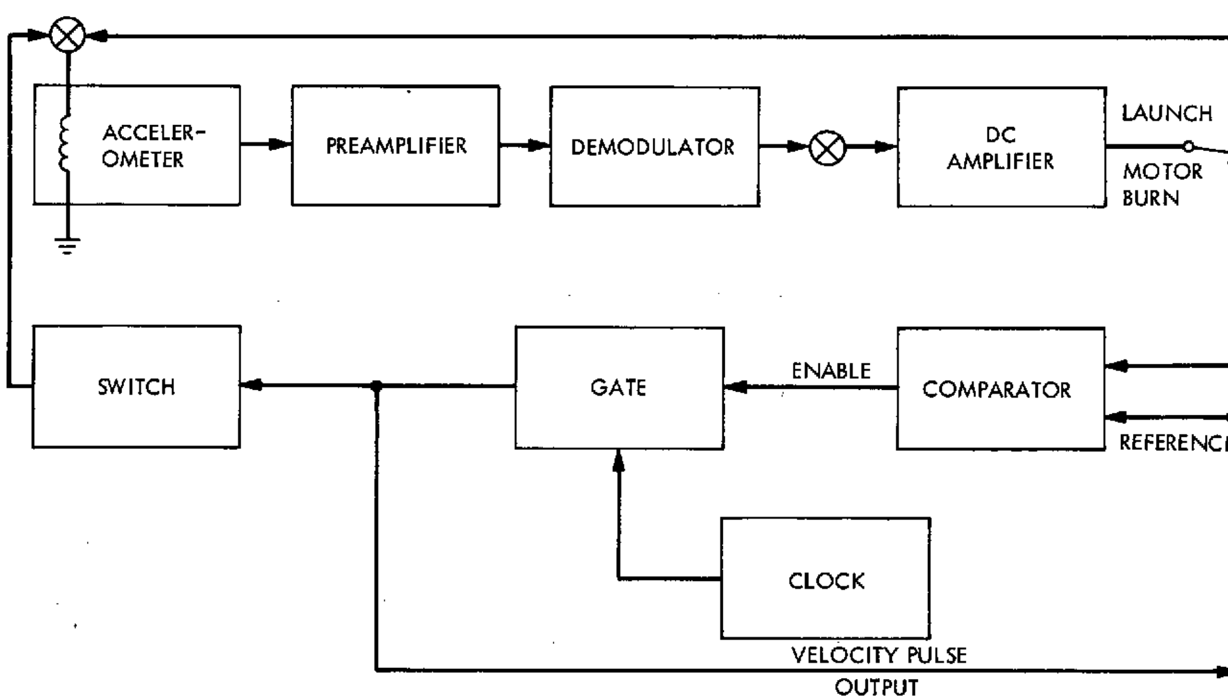


Fig. 33. Accelerometer loop

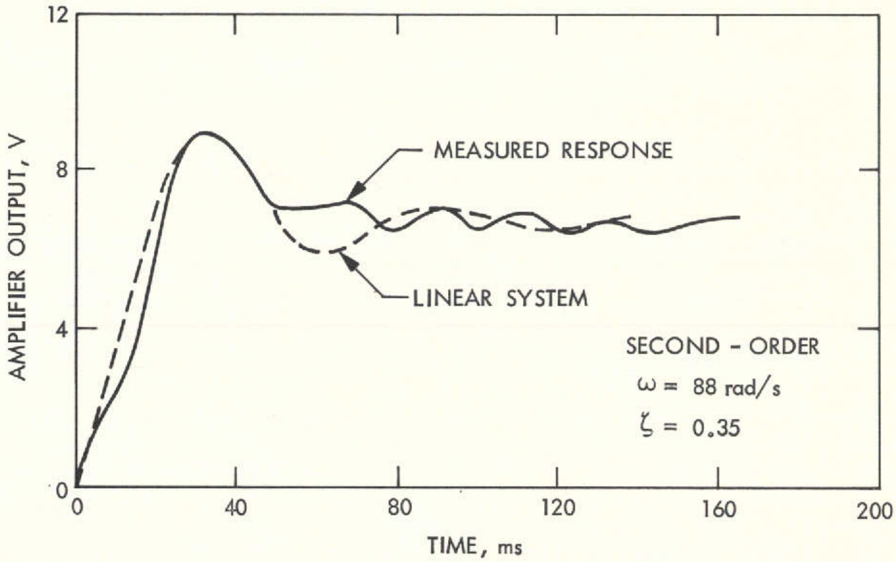


Fig. 34. Gyro step response

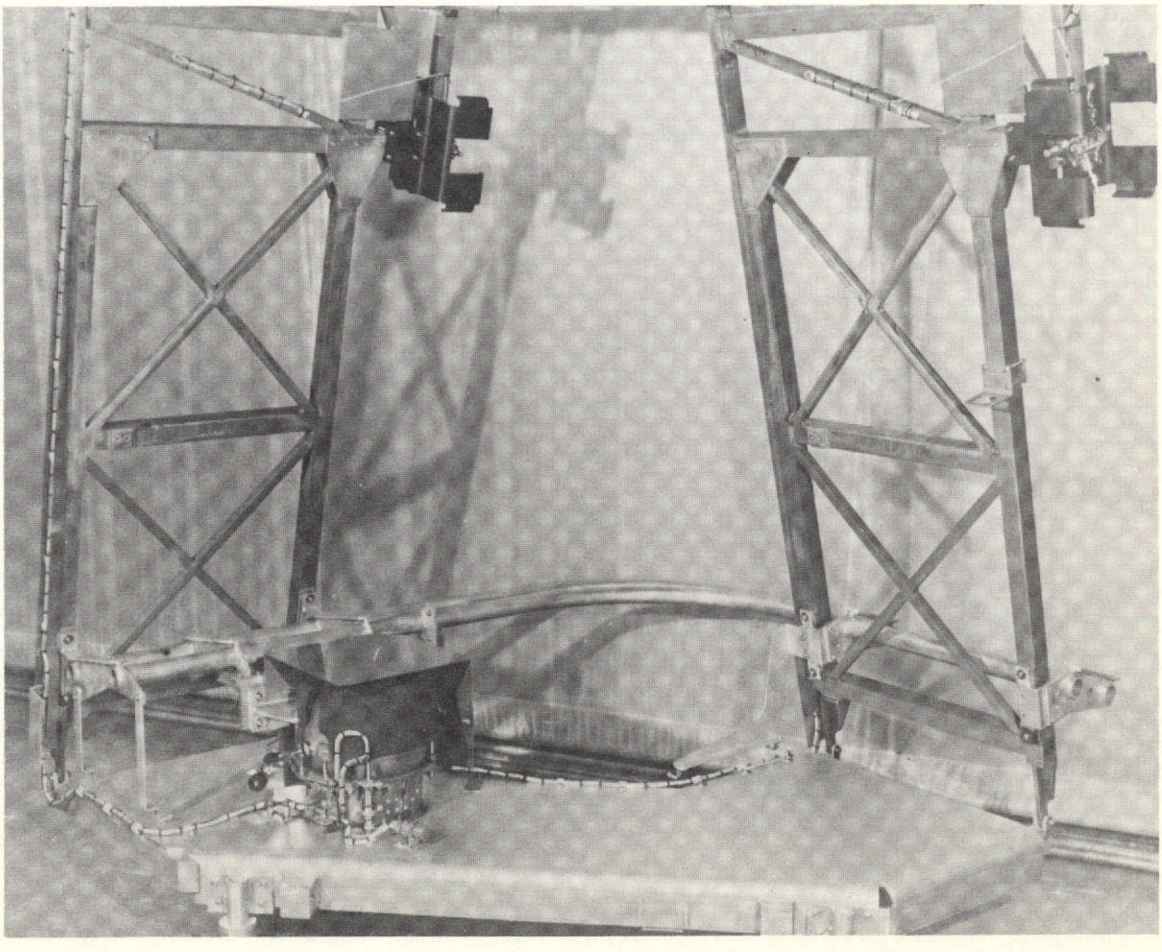


Fig. 35. RCA mounted on handling fixture

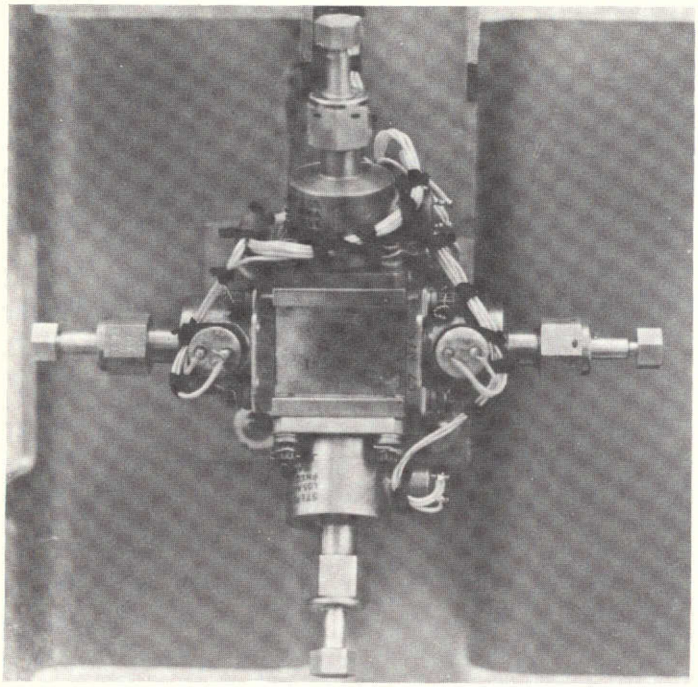


Fig. 36. Four valve unit with test nozzles

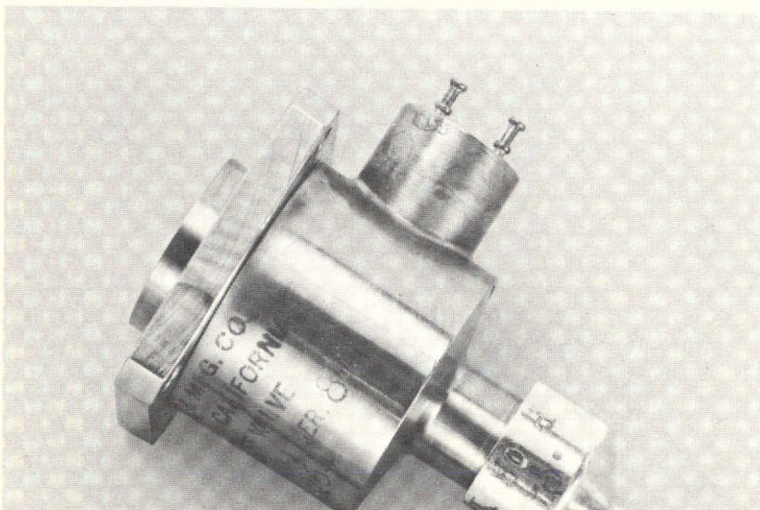


Fig. 37. Jet valve with flight nozzle

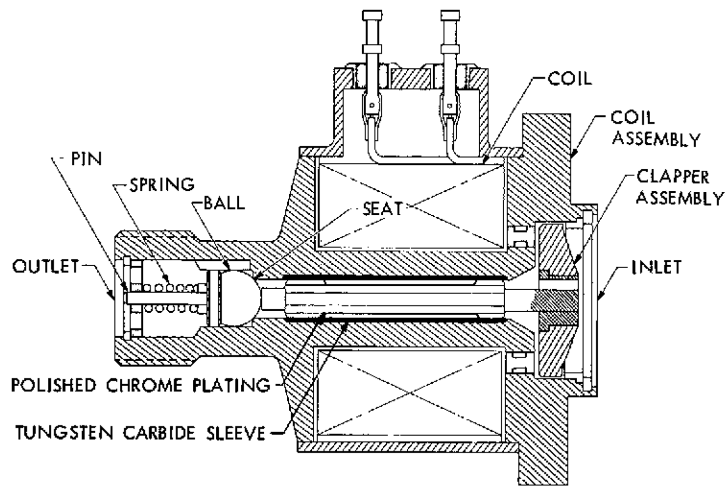


Fig. 38. Jet valve cross section

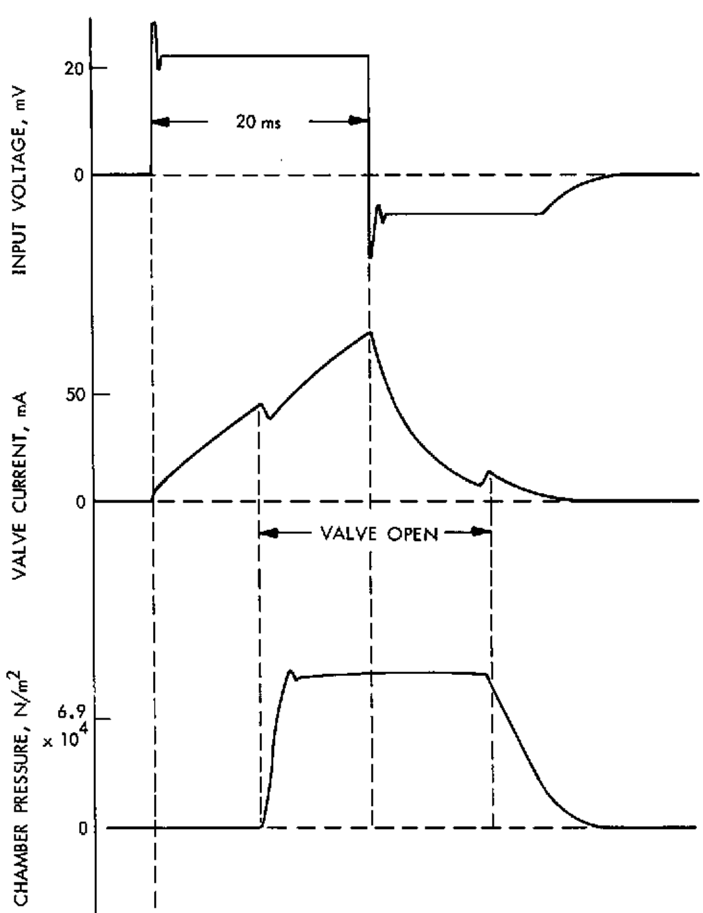


Fig. 39. Typical response of valve to command excitation (0. 5-mm (0. 020-in.) nozzle throat diameter

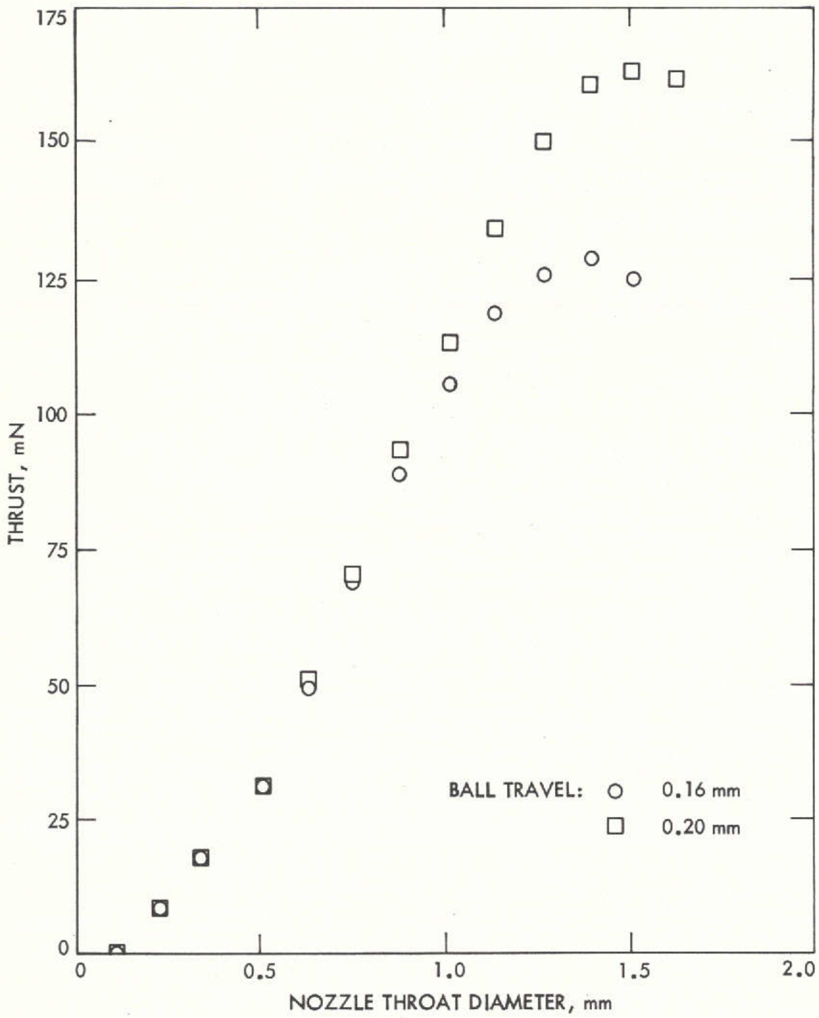


Fig. 40. Jet valve maximum thrust capability

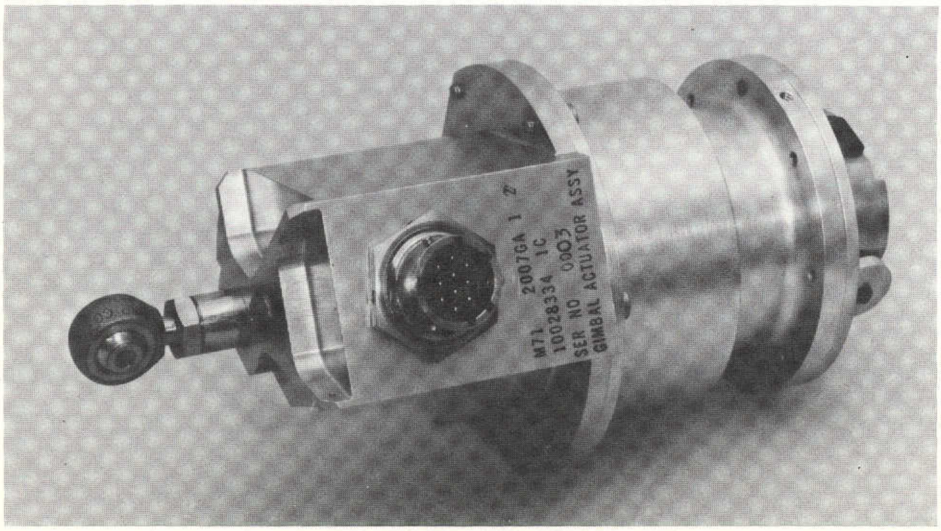


Fig. 41. Gembal Actuator Assembly

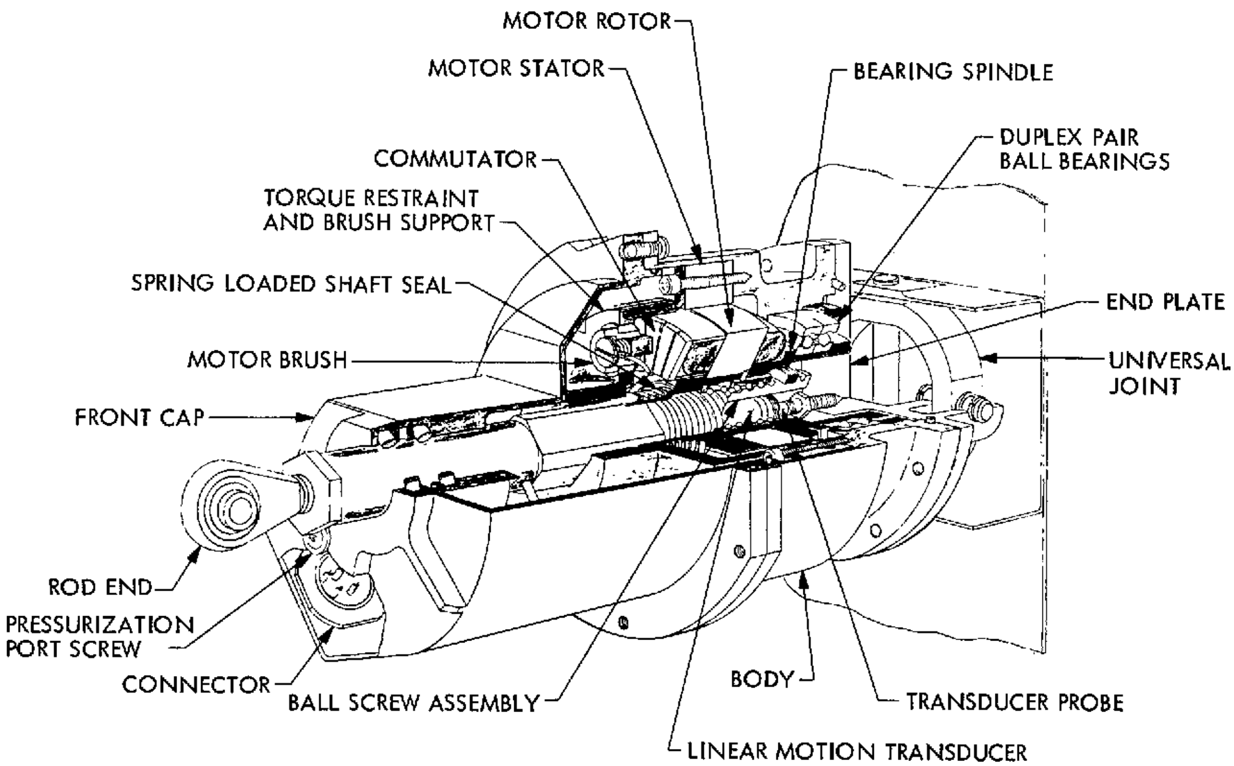


Fig. 42. Gimbal Actuator Assembly, cutaway view

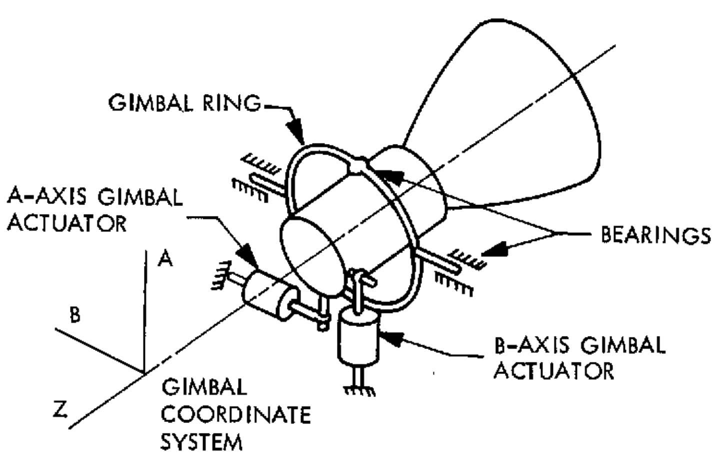


Fig. 43. Engine and gimbal sketch

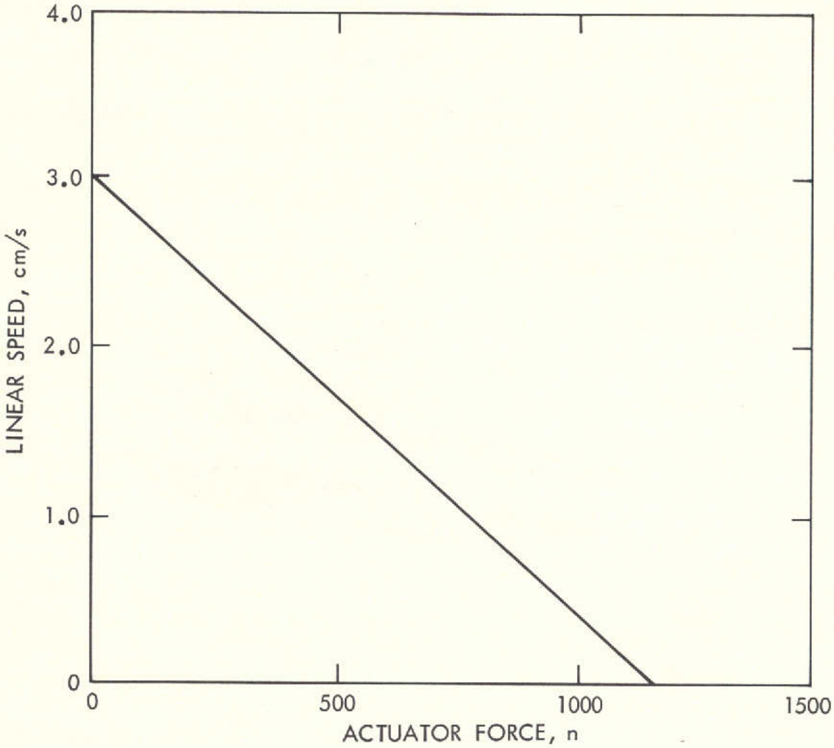


Fig. 44. Gimbal actuator load-spped performance

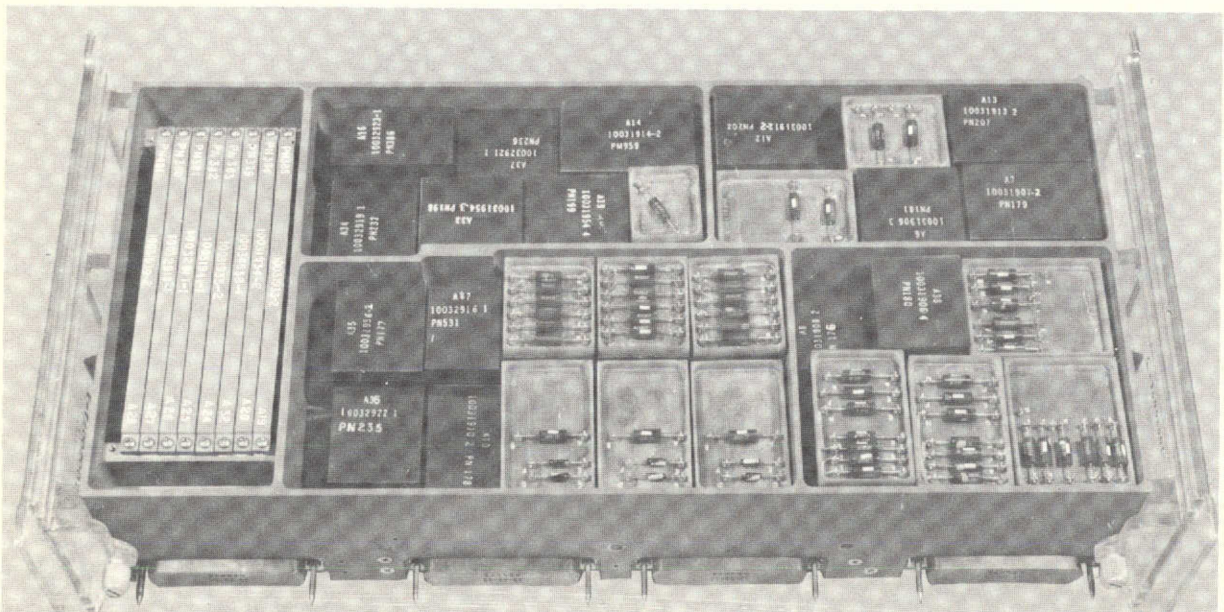


Fig. 45. Attitude control electronics

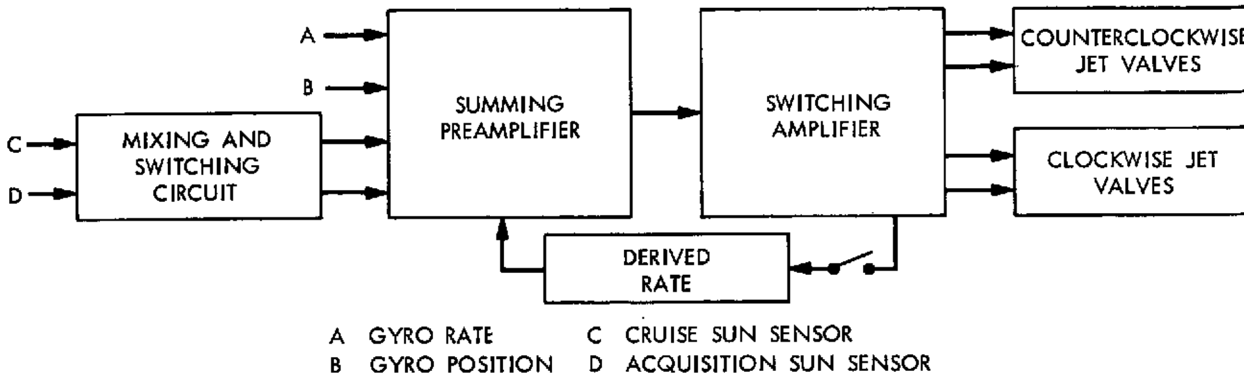


Fig. 46. RCA electronics for the pitch (yaw) axis

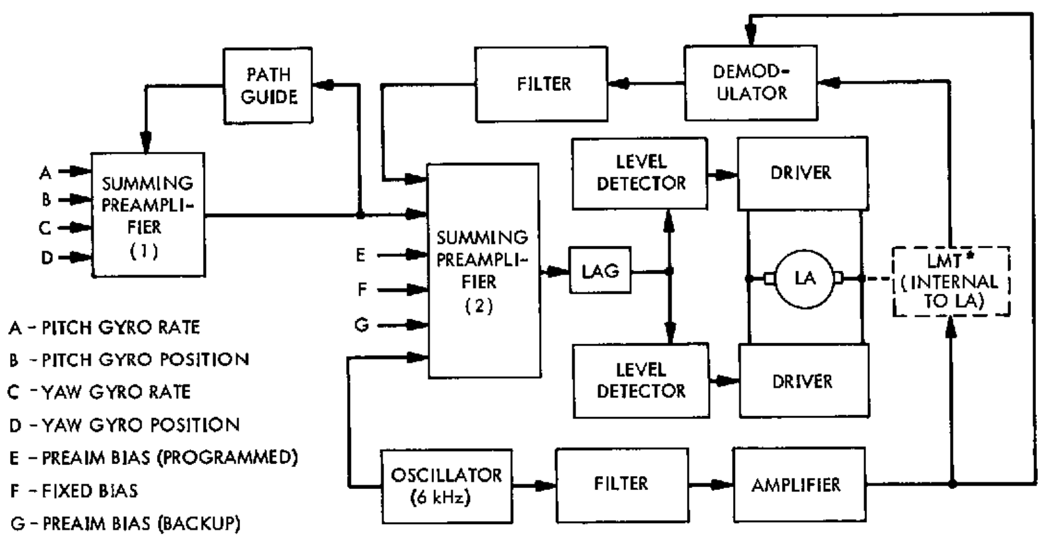


Fig. 47. TVC electronics, single axis

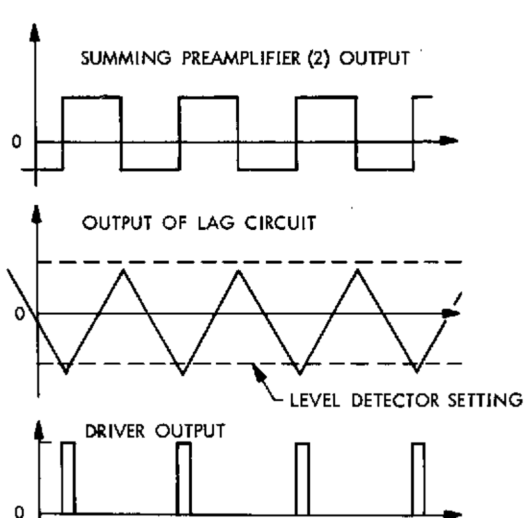


Fig. 48. Pulse width modulator operation

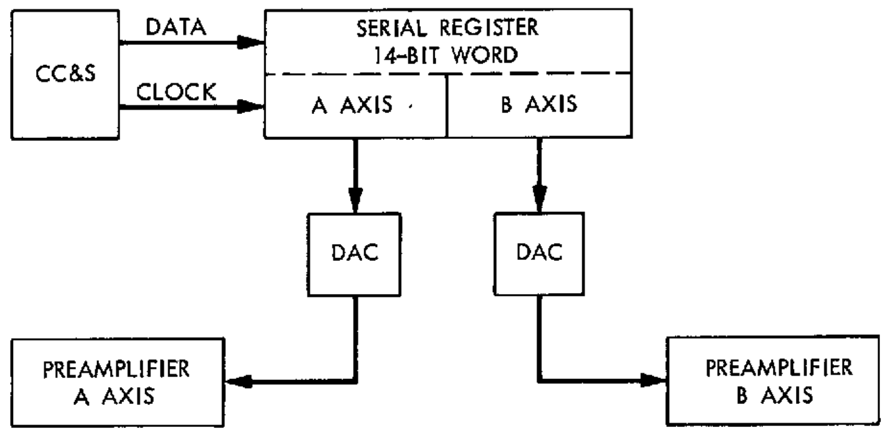


Fig. 49. Programmed preaim bias

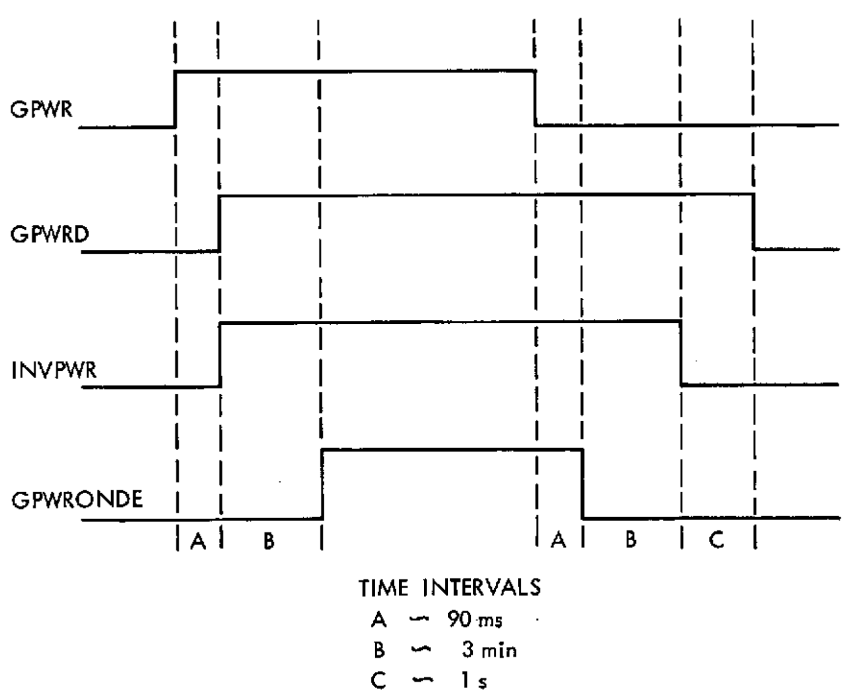


Fig. 50. Three-minute timer

APPENDIX A
TELEMETRY

The telemetry subsystem has 91 channels (or words) for engineering data.[32] Of these, fourteen 7-bit words and 2 bits from another word are allocated to the ACS. Each word has a 7-bit length, so a maximum resolution of 1/128 is available.

The data to be transmitted is arranged into four decks. The sampling rate for words in the high deck is a function of the data transmission rate, which for engineering data is either 33-1/3 bits/s or 8-1/3 bits/s. The medium, low, and low-low decks have sampling rates that are 1/10, 1/100, and 1/200 of the high-deck rates respectively. The sampling rates for words in each deck are listed in Table A-1, along with the total number of words in each deck and the number allocated to the ACS.

To maximize the useful information output, five channels assigned to the ACS monitor different parameters depending upon the ACS operating mode. The outputs for these channels are listed in Table A-2, all of which are located in the high (100) deck (see also Subsection IV-H).

The remaining 9-2/7 words are unswitched and are tabulated in Table A-3. The channel numbers 100, 200, 300, and 400 correspond to the high, medium, low and low-low decks respectively. Channel 412, Bay III, temperature is not assigned to ACS, however it is this temperature measurement that is used to determine gyro and accelerometer temperatures, and for this reason it is included in the table.

[32] Excluding 3 words used for synchronization and indexing.

Table A-1. Sampling intervals and word count

Deck	Rate, bits/s		Total number of words	ACS number of words
	33 1/3	8 1/3		
High (100)	4.2 s	16.8 s	14	7 2/7
Medium (200)	42 s	2.8 min	27	0
Low (300)	7 min	28 min	10	2
Low-low (400)	14 min	56 min	40	5

Table A-2. ACS telemetry assignments (switched)

Channel	Signal	Mode[a]								Typical calibration range	Typical resolution
		1	2	3	4	5	6	7	8		
105	Cruise Sun sensors (pitch)				X				X	-1.0 to + 1.0 deg	0.015 deg
105	Gyro rate (pitch)	X	X	X						-2.0 to +2.0 deg/s	0.04 deg/s
105	Gyro integrator (pitch position)					X	X	X		-6 to +6 deg	0.1 deg
106	Cruise Sun sensors (yaw)				X				X	-1.0 to +1.0 deg	0.015 deg
106	Gyro rate (yaw)	X	X	X						-2.0 to +2.0 deg/s	0.04 deg/s
106	Gyro integrator (yaw position)					X	X	X		-6 to +6 deg	0.1 deg
107	Canopus tracker integrator output (roll position, fine)				X					-2.0 to +2.0 deg	0.03 deg
107	Gyro rate (roll)	X	X	X						-1.0 to +1.0 deg/s	0.02 deg/s
107	Gyro integrator (roll position)					X	X	X	X	-6 to +6 deg	0.1 deg
112	Combined Sun sensors (pitch)		X	X	X				X	-180 to +180 deg	0.04 deg[b]
112	Gimbal actuator position (A)	X				X	X	X		-9 to +9 deg	0.15 deg
113	Combined Sun sensors (yaw)		X	X	X				X	-180 to +180 deg	0.04 deg[b]
113	Gimbal actuator position (B)	X				X	X	X		-9 to +9 deg	0.15 deg

[a] 1 = launch; 2 = sun acquisition; 3 = roll search; 4 = celestial cruise; 5 = all-axes inertial; 6 = commanded turn; 7 = TVC; 8 = roll inertial

[b] Valid for outputs between -1.0 and +1.0 deg only.

Table A-3. ACS telemetry assignments (unswitched)

Channel	Signal	Typical calibration range	Typical resolution
108	Canopus tracker intensity output	dark to 4 × Canopus	Not applicable (N/A)
114	Canopus tracker integrator output (roll position, coarse)	-3.0 to +3.0 deg	0.06 deg
118	Event counter		
	Gyros on (1 bit)	N/A	N/A
	Sun acquired (1 bit)	N/A	N/A
307	Canopus tracker adaptive low-gate setting	3 discrete levels	N/A
309	Canopus tracker cone angle	5 discrete levels	N/A
407	Canopus tracker temperature	255 to 325 D (0 to 125°F)	0.55 K (1.0°F)
412	Bay III temperature	273 to 325 K (32 to 125°F)	0.55 K (1.0°F)
417	Sun sensor temperature	225 to 325 D (0 to 125°F)	0.55 K (1.0°F)
426	-X/+Y N_2 pressure	0 to 27.6×10^6 N/m^2 (0 to 4000 psia)	2.8×10^5 N/m^2 (40 psia)
427	+X/-Y N_2 pressure	0 to 27.6×10^6 N/m^2 (0 to 4000 psia)	2.8×10^5 N/m^2 (40 psia)
438	+X/-Y N_2 temperature	273 to 325 K (32 to 125°F)	0.55 K (1.0°F)

APPENDIX B
SPACECRAFT COORDINATE SYSTEM

The pitch, yaw, and roll axes of the spacecraft are illustrated in Fig. B-1. The roll axis is perpendicular to the plane containing the solar panels and is directed away from the Sun whenever the ACS is operating in a celestial cruise mode. The pitch and yaw axes are contained in the plane of the solar panels. For most purposes, this coordinate system is adequate; however, several other parameters must be identified for purposes of discussing the Canopus tracker.

The Canopus tracker null axis lies along the center of the Canopus tracker optical field of view and is perpendicular to the spacecraft roll axis.[33] The null plane of the spacecraft contains the roll axis and the Canopus tracker null axis, and is defined by these axes. The clock angle is measured about the minus roll axis and is referenced to the null plane.

The cone angle of a celestial object (such as Canopus) is defined as the angle between the line of sight to the Sun and the line of sight to the celestial object as seen by an observer on the spacecraft. The cone angle of a line referenced to the spacecraft is defined as the angle between the minus roll axis and the line of interest.

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[33] See Subsection III-B for definition of Canopus tracker fields of view.

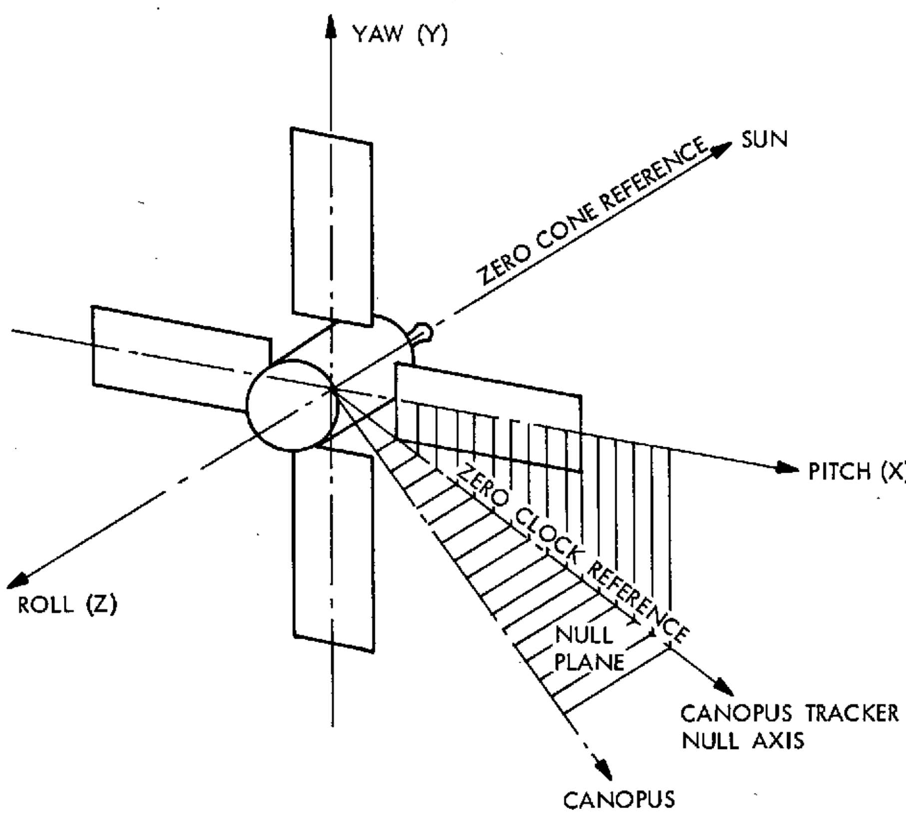

Fig. B-1. Spacecraft coordinate system

NASA — JPL — Coml., L.A., Calif.