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OUTER PLANETARY EXPLORER (OPE) CONTROL SYSTEM STUDY

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GODDARD SPACE FLIGHT CENTER
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James H. Donohue
Stabilization and Control Branch

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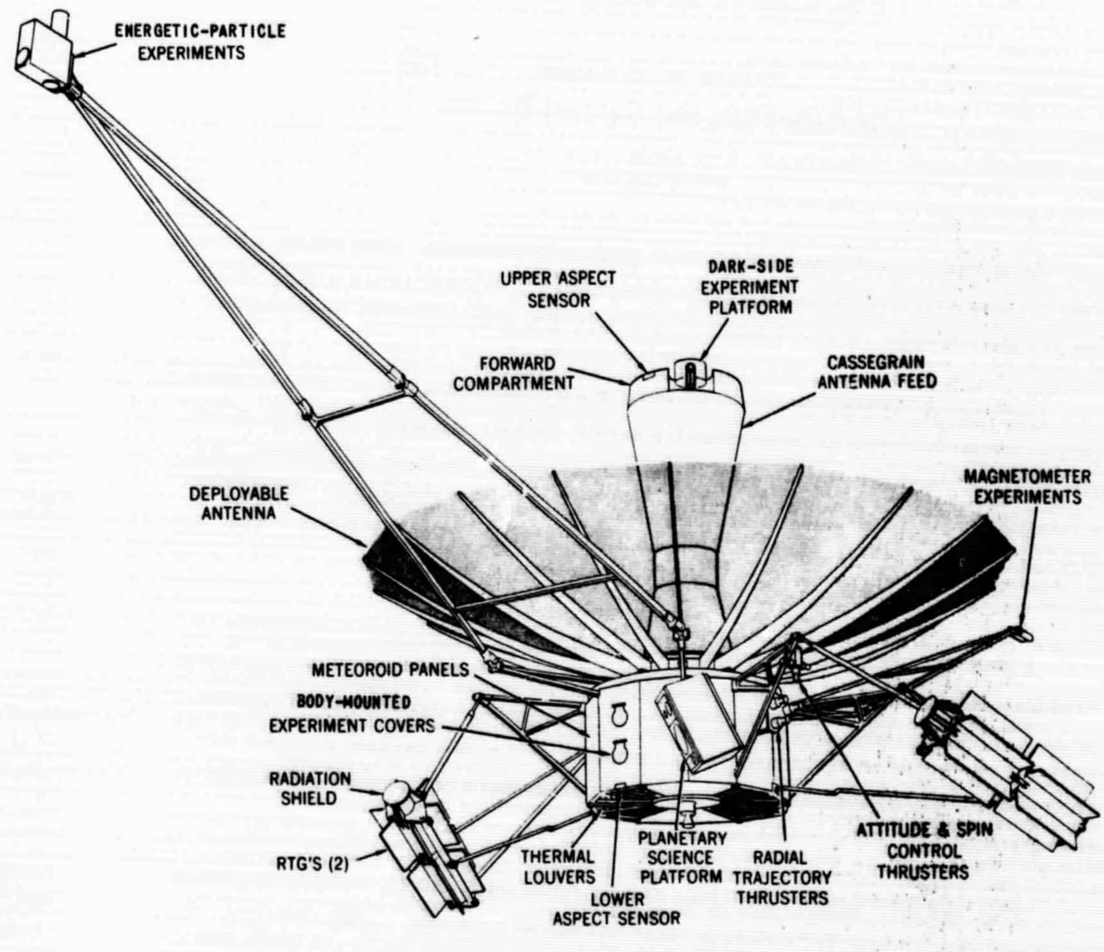
SUMMARY

This document describes the author's efforts, beginning February 1969, to establish a control-system concept compatible with the requirements of the Outer Planetary Explorer (OPE) spacecraft.

During the early portion of this study, the program did not require experiments to continually point at the planet. The project felt that the scan provided by a spinning satellite was adequate. As a result, a spin-stabilized control system was designed which is similar to the system described in the Galactic Jupiter Probe Study, X-701-67-566, Volume 2.

Later requirements dictated planetary pointing, resulting in significant changes to the control-system design so that the vehicle would be spin stabilized during the cruise phase and three-axis stabilized during the planetary encounter. This document covers both the three-axis stabilization system and the spin-stabilization system for OPE.

The author acknowledges the assistance of H. Hoffman, who contributed to the three-axis stabilization concept and design; B. Zimmerman, who offered helpful suggestions in the area of backup control-system mechanization; W. Raskin and L. Draper, for their conceptual designs of the Canopus sensor and planet tracker; and E. Scobey, for his error analysis of the RF-interferometer earth sensor.



Frontispiece. Outer Planetary Explorer

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OUTER PLANETARY EXPLORER
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CONTROL SYSTEM STUDY

INTRODUCTION

A highly reliable and moderately accurate control scheme is necessary to:

- Fulfill the long mission-life requirement
- Point a high-gain antenna beam (2 degrees) toward the earth from long distances (30 astronomical units (AU's))
- Point a scientific platform at the planet during encounter

The spin-stabilization approach is the most desirable control scheme for the long cruise phases of the mission because of:

- The inherent stability of a spinning spacecraft
- The relatively small spin-axis precession required to track the earth
- The simplicity of the precession system

Assuming that no real requirement exists for spacecraft scan capability during planetary encounter, a logical control system appears to be one which maintains the gyroscopic stability feature by transferring its spin momentum to a momentum wheel and which stabilizes the spacecraft in three axes. The most practical control scheme for the encounter phase of the mission appears to be a single-gimbaled scientific platform which will track the planet by rolling the spacecraft and by changing the gimbal angle as dictated by a platform-mounted planet tracker.

After the radioisotope thermal generator (RTG) booms, the experiment booms, and the high-gain antenna are deployed and before separation, Burner II will orient the spacecraft and spin it up to the desired 1 rpm. Because of thermal considerations, the angle between the spin axis and the sunline must be kept below 70 degrees. Figure 1 shows the time history of this angle with the spin axis initially pointed toward the earth. With this initial orientation, earth tracking could be delayed for more than a month from a thermal standpoint.

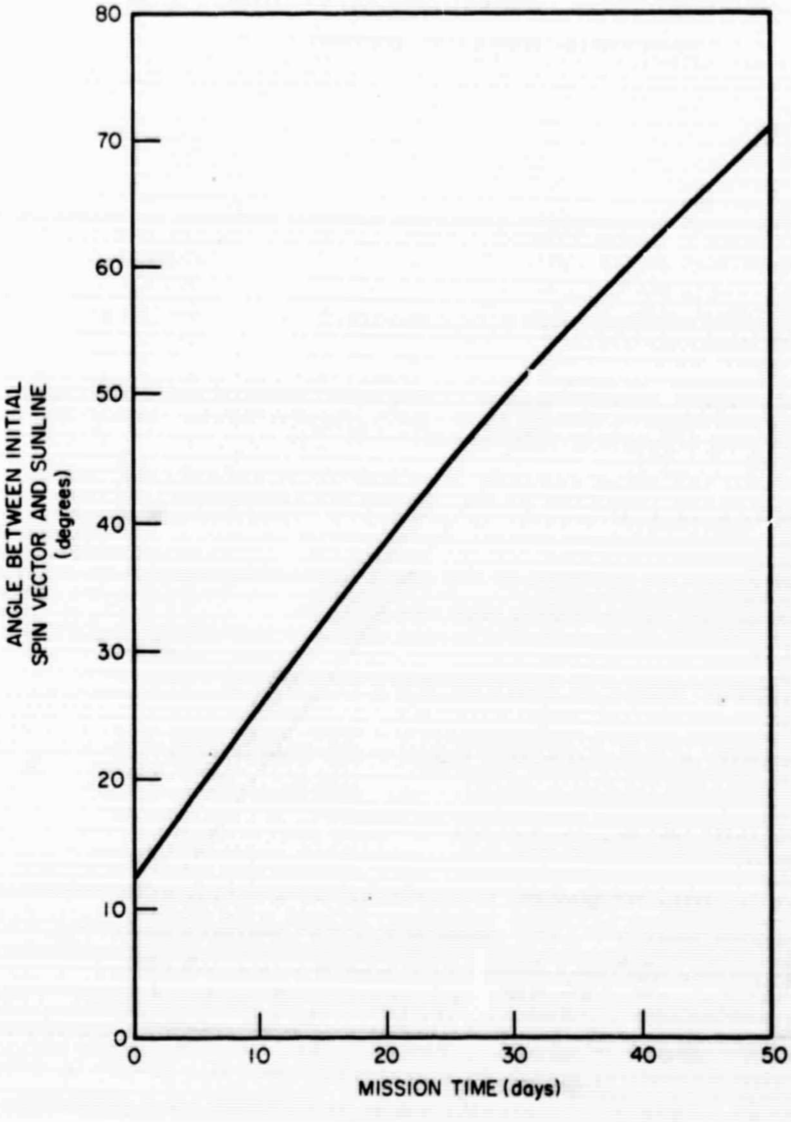

Figure 1. Time History of Angle Between Spin Axis and Sunline (Spin Axis Initially Pointed at Earth)

However, a more reasonable initial spin-axis orientation would align the spin axis along the perpendicular to the critical plane for the T + 5-day trajectory correction. The desired inertial velocity direction is normal to this plane. Using radial thrusting over a portion of the spin cycle would eliminate undesired velocity components in the critical plane. Time-delaying the radial-thruster firings with respect to the sensed sun or Canopus crossings would accomplish the inertial phasing.

As Table 1 shows, the angle of the critical-plane normal vector for a T + 5-day correction with respect to the sunline and earthline is located nearly in the ecliptic plane. No thermal problem exists for this initial spin-axis orientation.

Table 1

T+5-Day Trajectory Correction

Mission	Angle Between Critical-Plane Normal Vector and Sunline (degrees)	Angle Between Critical-Plane Normal Vector and Earthline (degrees)
1974 Jupiter flyby	46.7	41.2
1977 Earth/Jupiter/Saturn/Pluto	42.8	43.4
1979 Earth/Jupiter/Uranus/Neptune	45.4	50.8

After the T + 5-day trajectory correction, commands through the ground link will periodically energize the cruise-control system, causing the high-gain antenna dish to discretely track the earth. At about 30 to 40 planet radii from Jupiter, ground command will turn on the encounter-control system. The 100 ft-lb-sec cruise spin momentum will transfer to the roll momentum wheel, reducing the spacecraft spin rate to 1/60 rpm (1 revolution per hour) for planet search.

After planet acquisition, the spacecraft spin rate will be essentially reduced to zero. The antenna dish will point toward the earth, and the single-gimbaled scientific platform will track the planet in both the daytime and nighttime phases of the encounter by changing its gimbal angle and by rolling about the earthline as the planet tracker or stored program dictates.

The maximum range for closed-loop tracking is conservatively based on the need for a total planet disc size of 3 degrees. This requirement stems from the envisioned size of the detectors on the planet tracker. Table 2 gives a conservative estimate of the maximum hours before radius of closest approach (RCA) at which closed-loop tracking will be possible. The relatively distant RCA (30 Rp*) causes the small acquisition range at Pluto.

*Rp = radius of planet

Table 2

Maximum Planet Acquisition Hours Before RCA

Launch	Jupiter	Saturn	Uranus	Neptune	Pluto
1974	41				
1977	41	34			3
1979	41		13	8	

After completing the encounter phase of the mission, the spacecraft will return to the spinning-cruise mode until it encounters the next planet. Required trajectory corrections will be made in the spinning mode, using axial and radial thrusters so that the high-gain antenna will always point toward the earth.

Requirements of the attitude-control system (ACS) are:

- To provide spin-axis (antenna-axis) earth-pointing capability with ±0.5-degree accuracy
- To provide pointing of the planetary directional experiments with ±0.2-degree accuracy
- To provide earth search and acquisition capability if the antenna is offset from earth for any reason
- To provide attitude-reference signals for experiments, telemetry, and other spacecraft systems (e.g., trajectory correction)

SYSTEM DESCRIPTION

Cruise (Spin-Stabilized)

The cruise-control system (Figure 2) consists of RF earth sensors, sun and Canopus sensors, electronics, and Freon-14 pneumatic systems. In the primary mode of operation, the RF earth sensors will detect the angular error between the spin axis (high-gain antenna axis) and the spacecraft/earthline. In addition, the RF sensor will provide a reference pulse when the earthline is in a particular spacecraft reference plane. When the antenna axis is not pointing toward the earth, the spinning motion of the spacecraft about the antenna axis will cause the earthline to rotate in the spacecraft coordinate system. By determining when the earthline is in a specific spacecraft plane, the electronics will cause the precession jet to fire for a short period relative to the spin period, causing the antenna axis to precess toward the earth. The jet can be fired at the time the reference pulse is generated, or it can be time-delayed, depending on the location of the precession jets relative to the reference plane.

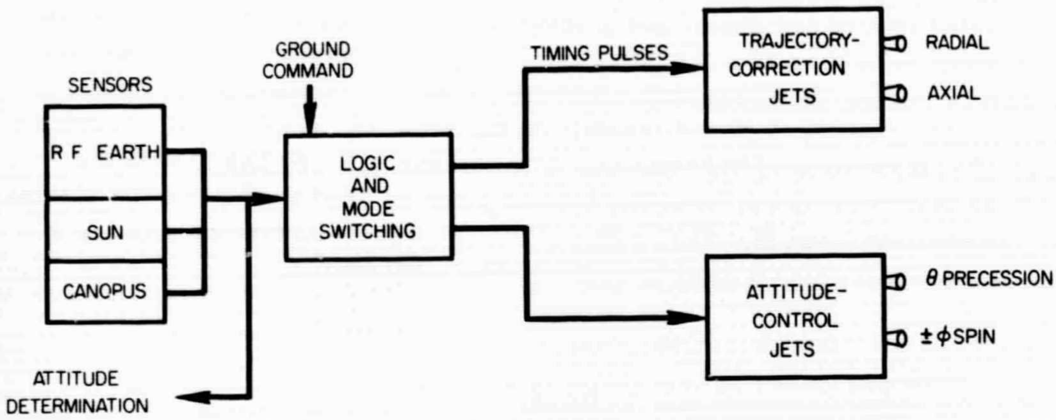

Figure 2. Cruise-Control System (Spin-Stabilized)

In the closed-loop operation, precession will begin when the error exceeds the 0.5-degree deadband and will stop when the error is less than 0.25 degrees. Two different RF earth sensors will be used. The first will use the medium-gain horns in interferometer fashion to develop antenna axis pointing error. The second sensor will develop the error information by processing the electronically offset signal received from the high-gain antenna. The high-gain acquisition angle is approximately 2 degrees.

The sun and Canopus system will provide both a backup earth-tracking system and the emergency search mode if communications with the spacecraft are lost for a specified period (days-weeks). Time-delaying the precession jet firings 90, 180, 270, and 360 degrees of the spin cycle as referenced to a sun sighting can offset the antenna axis from the sunline and can move it inertially about the sunline. The antenna axis will point to earth when the angle between it and the sunline, as well as the angle determined from the time between sun and Canopus sightings, is as programmed.

Open-loop precession of the antenna axis will be possible where the ground link controls the precession jet firings. A ground-generated train of timing commands will be sent to the spacecraft by processing telemetry information from the sensors. To provide the proper phasing, the jet will be fired after a specified sensor pulse. It will also be possible to phase the jet firings from the ground. Ground commands will select and initiate all control modes except for the emergency search mode, which will be discussed later.

Planetary Encounter (Three-Axis Stabilized)

Figure 3 shows the planetary-encounter control system. Figure 4 shows that the apparent motion of the planet, with respect to an inertially fixed spacecraft frame of reference, necessitates two rotations of the scientific platform to keep it pointed toward the planet and to keep the antenna axis pointed toward the earth. Instead of using a two-gimbaled platform, the required rotations will be performed by rolling the spacecraft about the earthline and turning the platform about its single gimbal. A planet tracker on the scientific platform will provide two axes of planet-position information. The tracker will be aligned with the platform so that information from one axis (δ_ϵ) can be used to position the gimbal angle. Information from the other axis (α) will be processed through a cosecant-control law and will then be used to control spacecraft roll.

$$\phi_\epsilon = \alpha \csc \delta$$

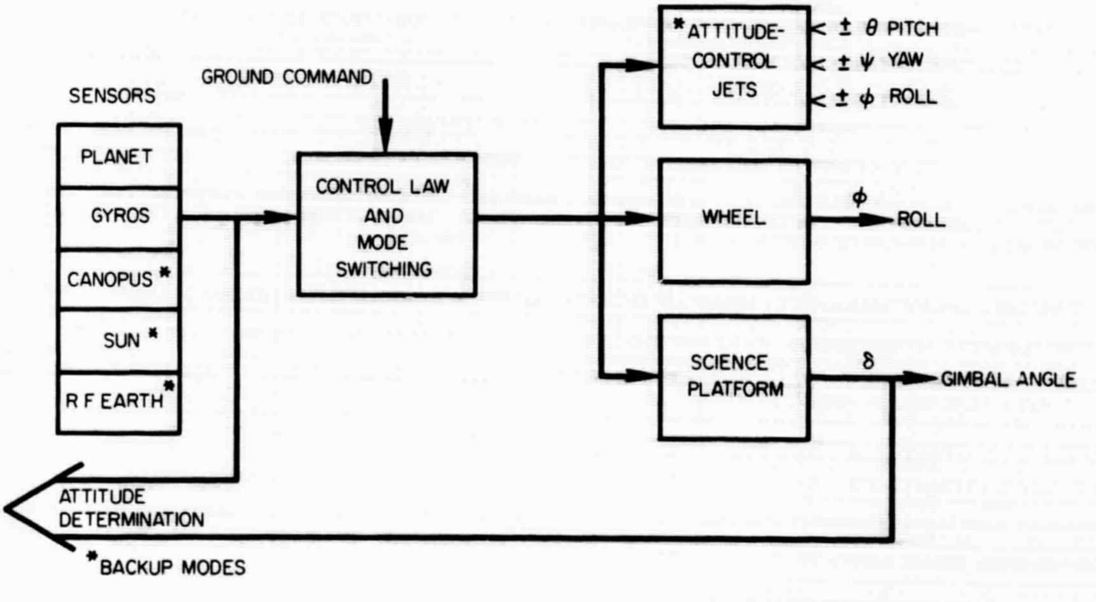

Figure 3. Encounter-Control System (Three-Axis Stabilized)

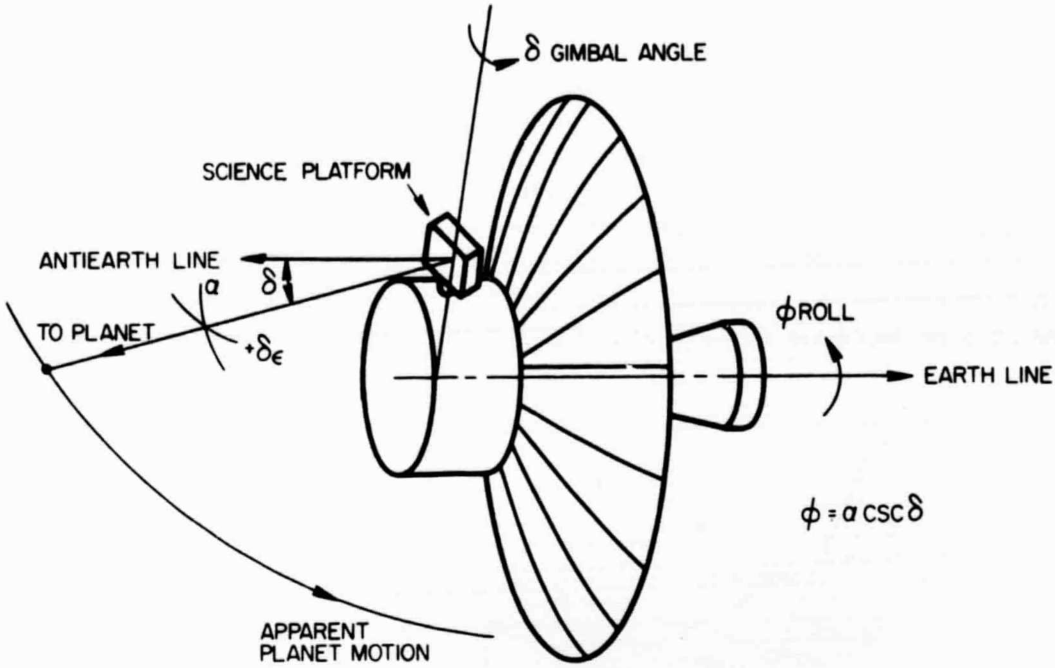

Figure 4. Planetary-Encounter Geometry

Search. Before encounter, the cruise momentum will be transferred from the spacecraft body to the roll momentum wheel, reducing the spacecraft roll rate from 1 rpm to 1/60 rpm. This rate will permit acquisition of either the planet or the navigation star, Canopus. Before this, most of the system components will have been evaluated (except for possibly the planet sensor) and the proper combination of sensors and controllers will have been selected.

Three-Axis Control. In the primary mode of operation, the planet tracker will develop error signals which will permit planet acquisition and tracking. The roll-error signals will torque the gyro and cause the momentum-wheel motor to supply the roll-control torques. The gimbal servomotor will directly position the gimbal angle. If the planet tracker is nonoperative, a backup track mode will be used. An on-board programmer will position the gimbal angle and supply roll-angle torque commands based on an earthline/Canopus reference. Initially, the gyro will be uncaged when referenced to the Canopus sensor and will then be torqued to the vicinity of the planet.

During planetary encounter, the antenna should remain very close to the earthline because of the large wheel momentum. However, if unexpected disturbances near the planets precess the spacecraft more than 0.5 degree off the earthline, the RF interferometer earth sensor will detect the error and the precession-control system will fire the proper pitch or yaw jet and reduce the error to less than 0.25 degree.

Major System Components

Sun Sensor.. The Adcole-type sun sensor measures the angle between the spacecraft-control axis and the sunline and expresses this angle as a digital number. Figure 5 shows the basic principle of operation. A gray-coded pattern on the bottom of a quartz block screens light passing through a slit on the top of the block to either illuminate or not illuminate each of eight photocells. The angle of incidence determines which photocells are illuminated. The outputs from each cell are amplified, and the presence (1) or absense (0) of a signal is stored and processed in a register to provide the desired output. The sun sensor provides an 8-bit word for a 128-degree field-of-view with a 0.5-degree resolution.

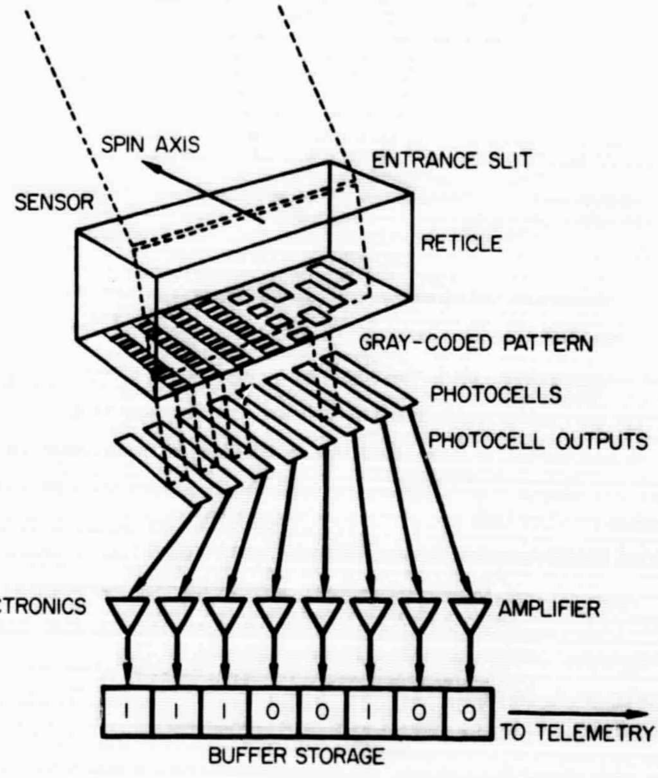

Figure 5. Sun Sensor

Although many of these solar-aspect systems have been used on board both spinning and nonspinning spacecraft, they have been used for near-earth applications only. The large dynamic range in sun size and intensity and the desire for improved resolution will necessitate some modifications to the present systems. Some specifications of the present Adcole system are:

- Power—Requires approximately 1 watt from the regulated supply
- Weight — 2 pounds (electronics and two sensor heads)
- Sensor field-of-view — Each sensor head has ± 65-degree field-of-view with a 0.5-degree resolution.

Additional nonspinning sensors located on the subreflector compartment will supply sunline information for attitude determination during the three-axis stabilized encounter mode.

Canopus Sensor. The Canopus sensor is a solid-state star scanner designed to measure the attitude of the spinning spacecraft during the cruise phase of the mission and to reference the integrating gyro during the encounter phase of the mission.

Basically, the sensor consists of an optical system which images Canopus on an array of silicon photodetectors (Figure 6). This array contains two side-by-side rows of 40 sensors separated by a 0.25-degree angle. The electronics interrogate each sensor to determine the position of the image on the detector array.

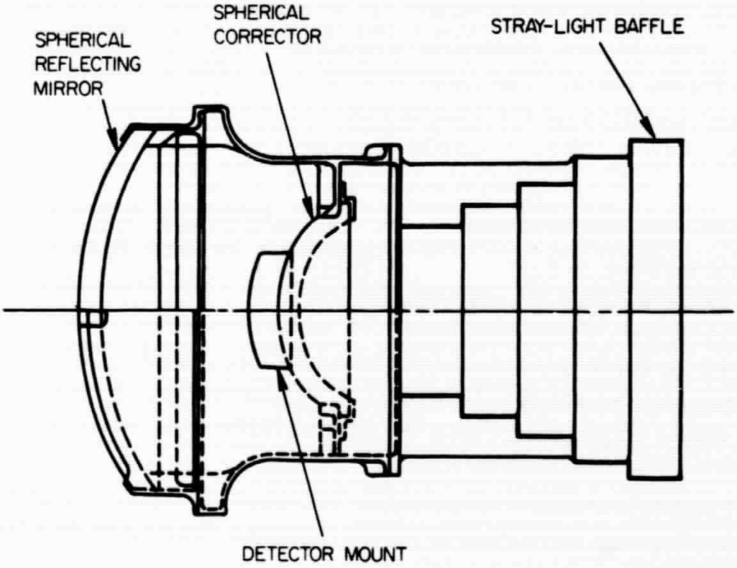

Figure 6. Canopus Sensor

The sensor uses a catadioptic optical system with a corrective lens made of Corning 7940 quartz. It has a 40- by 1-degree field-of-view. The sensor axis is normal to the spin axis with the 40-degree dimension along the spin axis. The sensor scans a star belt 40 degrees wide about the celestial sphere. When the spin axis lies nearly in the ecliptic plane, the scanning pattern is centered about the ecliptic poles and will therefore always include Canopus. For large spin-axis orientations out of the ecliptic plane, the pattern will include other stars near the celestial equator which can be used in the control-system scheme.

Ground command can control the threshold level to permit proper operation if sensitivity changes. In the cruise phase, time-grating based on the spacecraft spin rate also serves to prevent confusion with Sirius or planets. Silicon photodiodes serve as the detector elements because of the vast reliability of, and radiation-effects data available for, silicon. These devices also exhibit rapid and reversible recovery from exposure to intense light.

Planet Tracker. The planet tracker consists of an optical system, a detector array, and electronics mounted on the gimbaled platform. It measures the visible light reflected from a planet and thus determines the center of the brightness. It will measure the angles α and δ_{ϵ}.

The reflective optical system images the planet on the detector array, a cruciform with a vertical axis of α and a horizontal axis of δ. These axes consist of small solid-state detectors 2 degrees wide by 0.15 degrees high. The electronics sample each detector, beginning at one end and sweeping through to the opposite end. Ideally, the detectors will read zero down to the point where the planet image lights them. The illuminated detectors will then have a finite output, showing the position of the image (correct to one detector element) along the detector array.

Figure 7 shows one form of logic. At the beginning of a cycle, the tracker sampling electronics puts one count into a register as each unlighted detector is sampled. The counting stops as lighted detectors are sampled and resumes with a negative sign when unlighted detectors are again reached. At the end of the sweep, the number in the register is twice the error signal. This will be true, even when the image is small and intersects only one axis. Several safety features can be added to prevent false counts or individual detector failures.

This type of digital sensor is desirable for planets like Jupiter which may vary greatly in brightness in different bands. These detectors are used only in the on or off mode.

Simplified calculations show that the sensor could be used with good safety margin on Jupiter and could be expected to work well even on Pluto. Because the proposed reflective optical system would result in a large obscuration, the overall optic diameter being considered is 4 inches. The required field-of-view is approximately 38 degrees (i.e., somewhat larger than Jupiter at RCA).

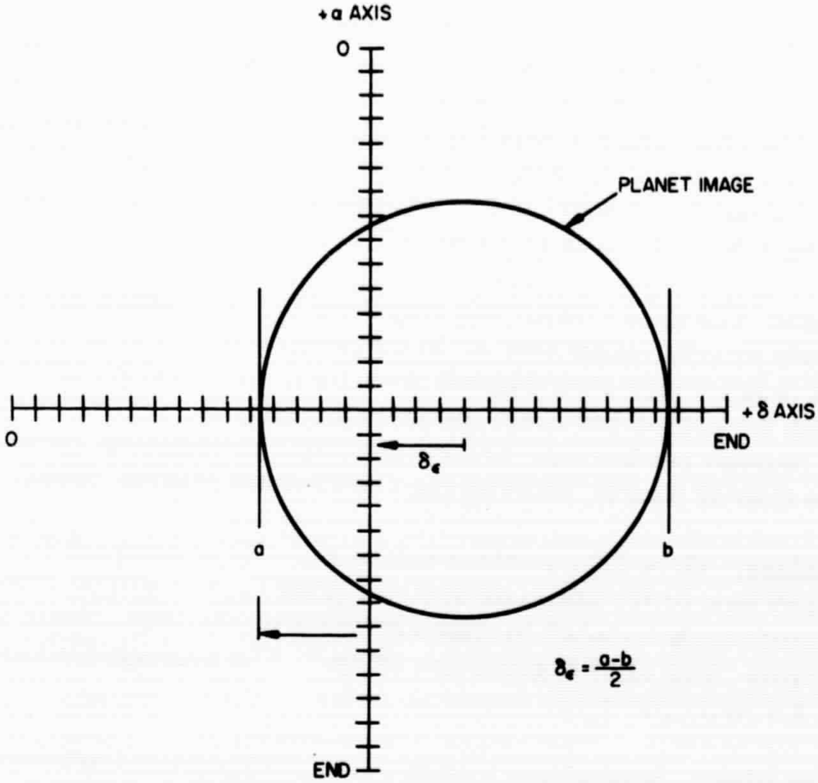

Figure 7. Planet Tracker

It is possible to use this instrument to offset the platform by command. In this mode, the electronics would call for the planet image to illuminate certain specific detector elements. Because the electronics sweep and read the elements individually, they can produce error signals until particular elements are lighted while others are unlighted. Actually, plans are to use offset pointing only near Jupiter and only on the roll axis. Although this action will not give an exact north-south sweep, it will approximate north-south and will give the actual positions.

Digital Timing Unit. The digital timing unit performs the following functions:

- Initiates the thrust pulses at the appropriate times to obtain the proper direction of spin-axis precession during the cruise and encounter phases of the mission
- Controls the pulse duration to obtain the proper firing angle and the desired rate of precession
- Delays the firing pulses for a specified time during the encounter and emergency search modes

Coning Dampers. A fluid-ring damper will be used for small cone angles, and a ball-impact damper will be used for large cone angles. Because it is temperature sensitive, the fluid-ring damper will be located in the electronics compartment. The ball-impact damper is not sensitive to temperature effects and will be located on one of the experiment booms. The dampers will have a time constant of about 3 to 4 hours and an estimated weight of about 3.25 pounds. Both dampers have been flown on previous GSFC satellites.

Gyro System. The gyro system is a single-axis system which can operate in either the rate or integrating modes. In the integrating mode, it will produce a linear output over a range of ±5 degrees from null. The total residual drift will be less than 0.1 degree per hour. In the rate mode, the system will measure rates up to 20 degrees per second. The system will be completely redundant, will weigh less than 10 pounds, and will require 12 watts average power.

For redundancy, three subminiature rate gyros will be incorporated into the rate feedback loop or the roll-axis system. These gyros will be grouped in a package approximately 2 inches in diameter and 3 inches long. Their operating range will be from 0.01 to 50 degrees per second. The average excitation power will be 3.5 watts per gyro.

Momentum Wheel. Several momentum-wheel configurations are under study. Wheel and motor characteristics of one of these configurations are:

Wheel	
Size	4-foot diameter
Weight	15 pounds
Moment-of-inertia	1.7 slug-ft^2
Motor speed	500 rpm
Angular momentum	100 ft-lb-sec
Motor	Dc, brushless (electronically commutated)
Torque	20 in-oz
Power	8 w at nominal rpm, 24 w at maximum torque

CRUISE SYSTEM OPERATION (Spin-Stabilized)

Precession Technique

Figures 8 and 9 show the conventional spin-axis precession technique used to precess the spin axis toward the sun. The sun is fixed in the plane of the paper and, as the sensor field-of-view passes the sun, a short jet burst is triggered, producing a quadrature change in momentum (ΔH). The original

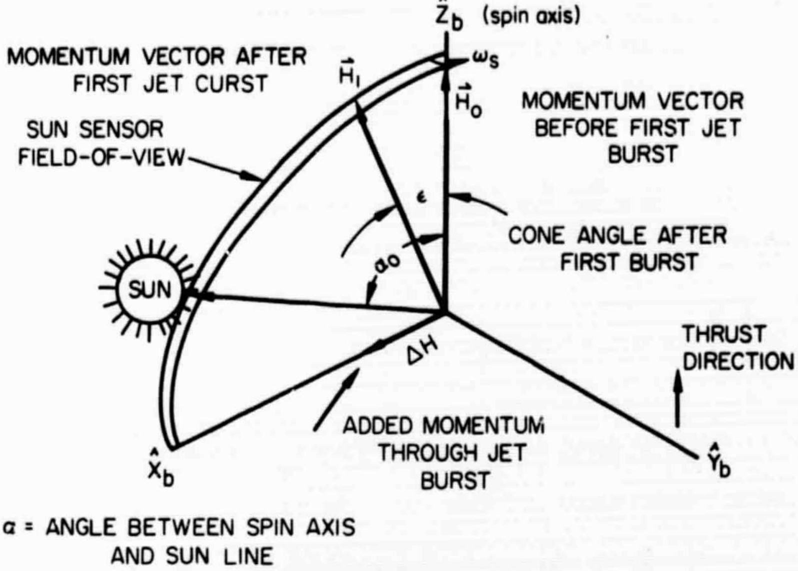

Figure 8. Body Frame

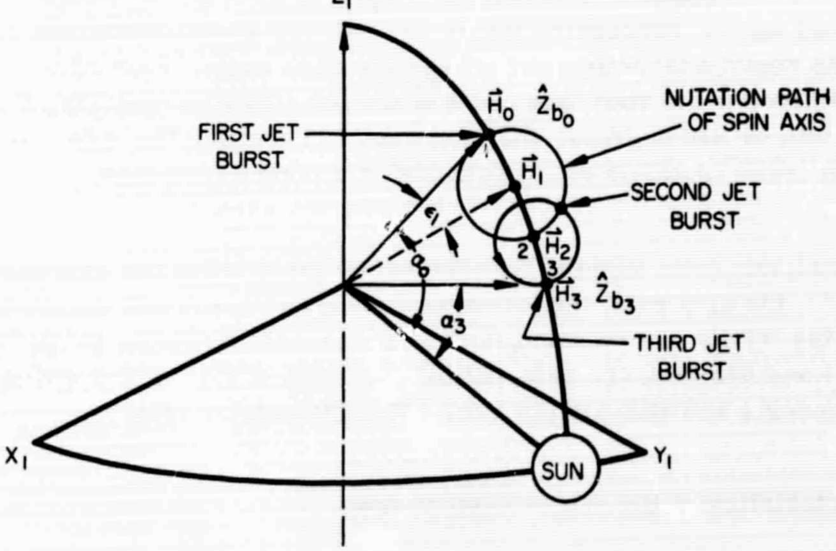

Figure 9. Inertial Frame

momentum vector (H_0) is thereby altered so that it is no longer aligned with the spin axis (Z_b); therefore, during the quiescent portion of the spin cycle, the spin axis nutates about the inertially fixed momentum vector (H_1). The period of nutation is

$$N = \frac{2\pi A}{H_0} = \frac{2\pi A}{C\omega_z} \approx 37 \text{ seconds}$$

where

A = transverse moment-of-inertia

C = spin moment-of-inertia

ω_z = spin rate

The angle between the spin axis and the momentum vector is called the cone angle (ϵ). With the next sun sighting and jet burst, the new momentum vector (H_2) will be positioned at point 2 (closer to the sunline), resulting in a different cone angle. Successive jet firings will increment the momentum vector in the desired direction and change the cone angle. Repeating this sequence will precess the spin axis into the sunline. By time-delaying the jet bursts 90, 180, or 270 degrees, the single jet can precess the spin axis clockwise around, away from, or counterclockwise around the sunline.

The maximum cone angle is determined by the ratio of the moment-of-inertia (C/A), the spin rate, the jet-thrust level, and the thrust duration. The position of the RTG's are adjusted to create a favorable inertia ratio. The jet torque level and duration are selected to keep cone-angle buildup within the sensor thresholds and still yield a reasonable precession rate.

Characteristics of the cruise system are:

Thrust level	0.5 lb
Moment arm	3.5 ft
Spin-axis inertia (fully loaded propellant)	954 slug-ft^2
Average transverse inertia	638 slug-ft^2
Spin-axis angular momentum at 1 rpm	100 ft-lb-sec
Precession step size per pulse	0.15 degrees
Precession rate (one pulse per spin cycle)	9 deg/hr
Precession requirement (degrees)	840 (total)
Earth tracking	285
Emergency search	130
External torques	145
Contingency	280
Spin trim requirement	3 rpm 2 rpm (redundancy) 5 rpm
Momentum requirement	1960 ft-lb-sec (total)
Precession	1460 ft-lb-sec
Spin change	500 ft-lb-sec
Propellant	Freon-14
Specific impulse	45 lb_f sec/lb_m
Propellant weight	12.4 lb

Primary Earth-Track Control Mode

The RF spin-axis earth-pointing system uses medium- and high-gain antennas for sensing the angular error (θ_E) between the spin axis and the earthline. The interferometer medium-gain antennas are mounted near the focal point of the dish and their axes are aligned parallel to the spin axis (Figure 10). Their radial separation is two wavelengths of the 2.1-gHz beacon signal radiated by the earth stations. The error between the spin axis and earthline causes the uplink carrier signal to impinge on the horn antennas with a different signal phase. The phase difference is processed into a voltage proportional to the angle between the spin axis and the projection of the earthline in the interferometer reference plane which contains both horns. As the antenna horns turn with respect to the earthline because of the spinning motion of the spacecraft, the error voltage is modulated at the spin frequency When the horns turn 90 degrees from the position shown in Figure 10, the error signal is zero because the earthline/spin-axis plane is then normal to the reference plane.

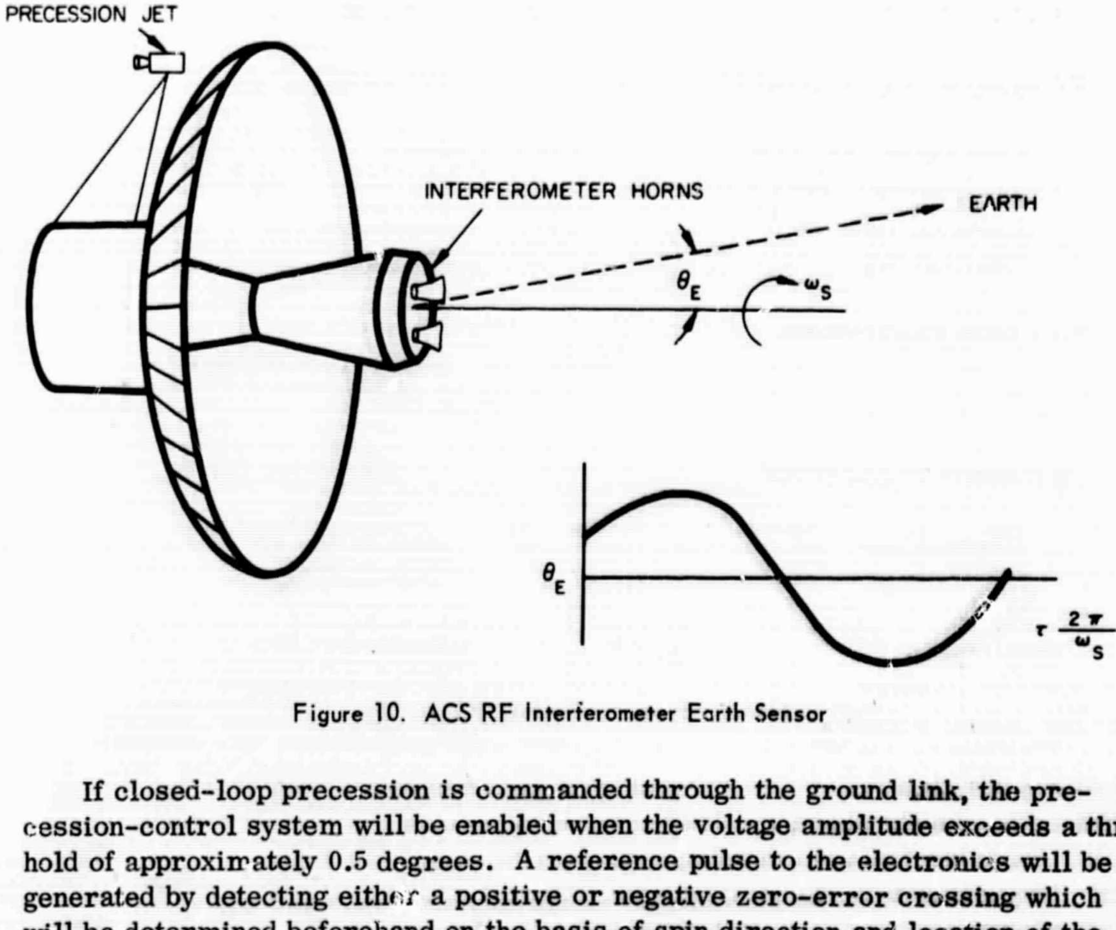

Figure 10. ACS RF Interferometer Earth Sensor

If closed-loop precession is commanded through the ground link, the precession-control system will be enabled when the voltage amplitude exceeds a threshold of approximately 0.5 degrees. A reference pulse to the electronics will be generated by detecting either a positive or negative zero-error crossing which will be determined beforehand on the basis of spin direction and location of the precession jet. The jet will be pulsed once per spin cycle until the error is reduced below 0.25 degree. The 24-degree beamwidth of the medium-gain horn antennas permits earth acquisition when the earthline is within ±12 degrees of the spin axis.

Many factors affect interferometer earth sensor performance: misalignment of the horn axis with the spin axis, error in separation distance between horns, and noise contributions. Based on noise alone, the interferometer system will easily accommodate the 0.25-degree threshold if the proper ground transmitter is used. Figure 11 shows the 3-sigma deviation of pointing error based on Gaussian noise contribution to the signal. As can be seen, the 210-foot 400-kw ground system will produce very little error at Pluto distances (30 AU).

Earth sensing with the high-gain system is obtained by a conical-scan technique where the radiation pattern (beam axis) is electronically offset with respect to the spin axis of the spacecraft on receive only. Beam-axis offset is in a known reference plane. The RF signal is modulated in amplitude by the offset at the spin frequency. As in the medium-gain system, processing of this signal yields a sinusoidal error voltage with amplitude proportional to the angle between the spin axis and the earthline. This voltage is greatest when the earthline lies in the spin-axis/beam-axis plane and is located nearest the beam axis. Threshold detection and phase measurement are incorporated to determine that the error is excessive and to generate a pulse to the control electronics which provide coordinated jet bursts to reduce the error below the 0.25-degree total threshold. This system will operate as a backup to the interferometer system in the cruise mode.

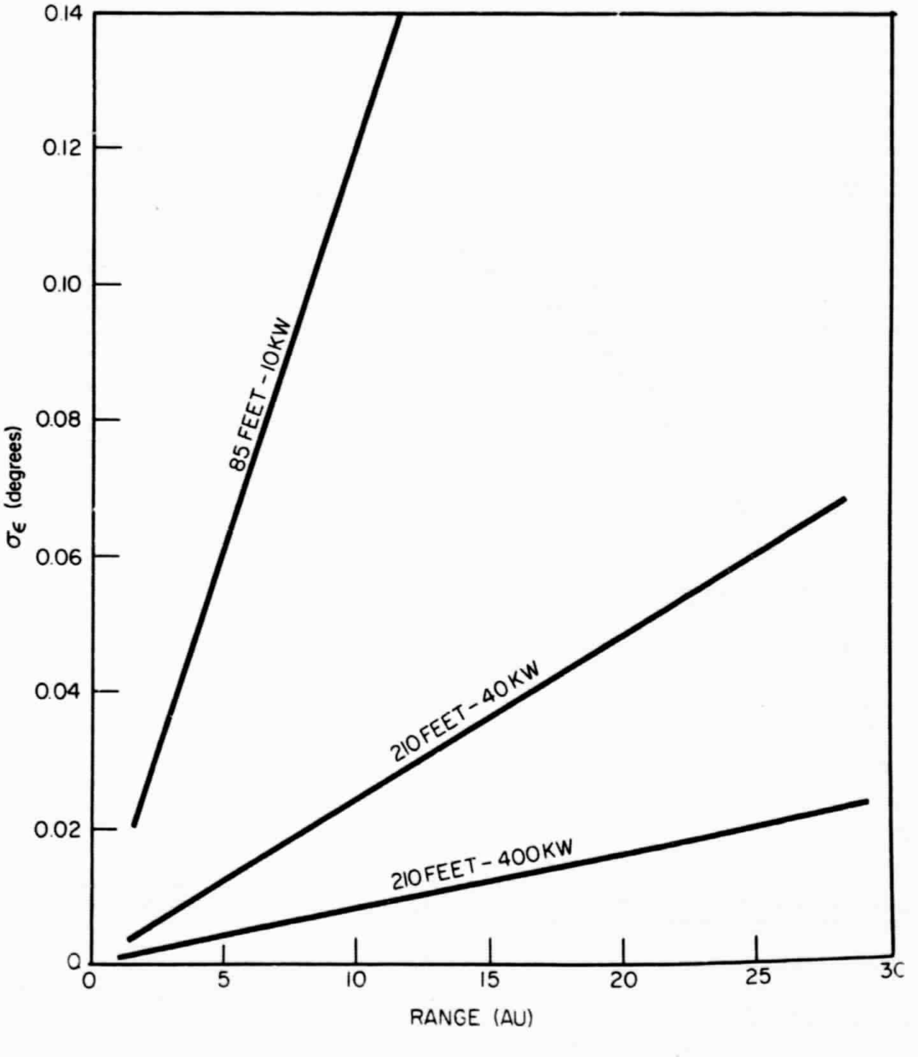

Figure 11. 2λ Interferometer Errors Caused by Gaussian Noise Contributions to the Signal

The spin axis of the spacecraft must be periodically precessed to keep it aligned with the earth. Keeping the spin axis pointing toward the earth for a typical mission (1977 Jupiter/Saturn/Pluto) will require 230 degrees of precession (Figure 12). The average rate of change is:

Time (days)	Rate (deg/wk)	Corrections
0-380	1.6	Once/week
380-620	0.5	Once/2 weeks
620-820	1.3	Once/2 weeks
820-3000	0.26	Once/month

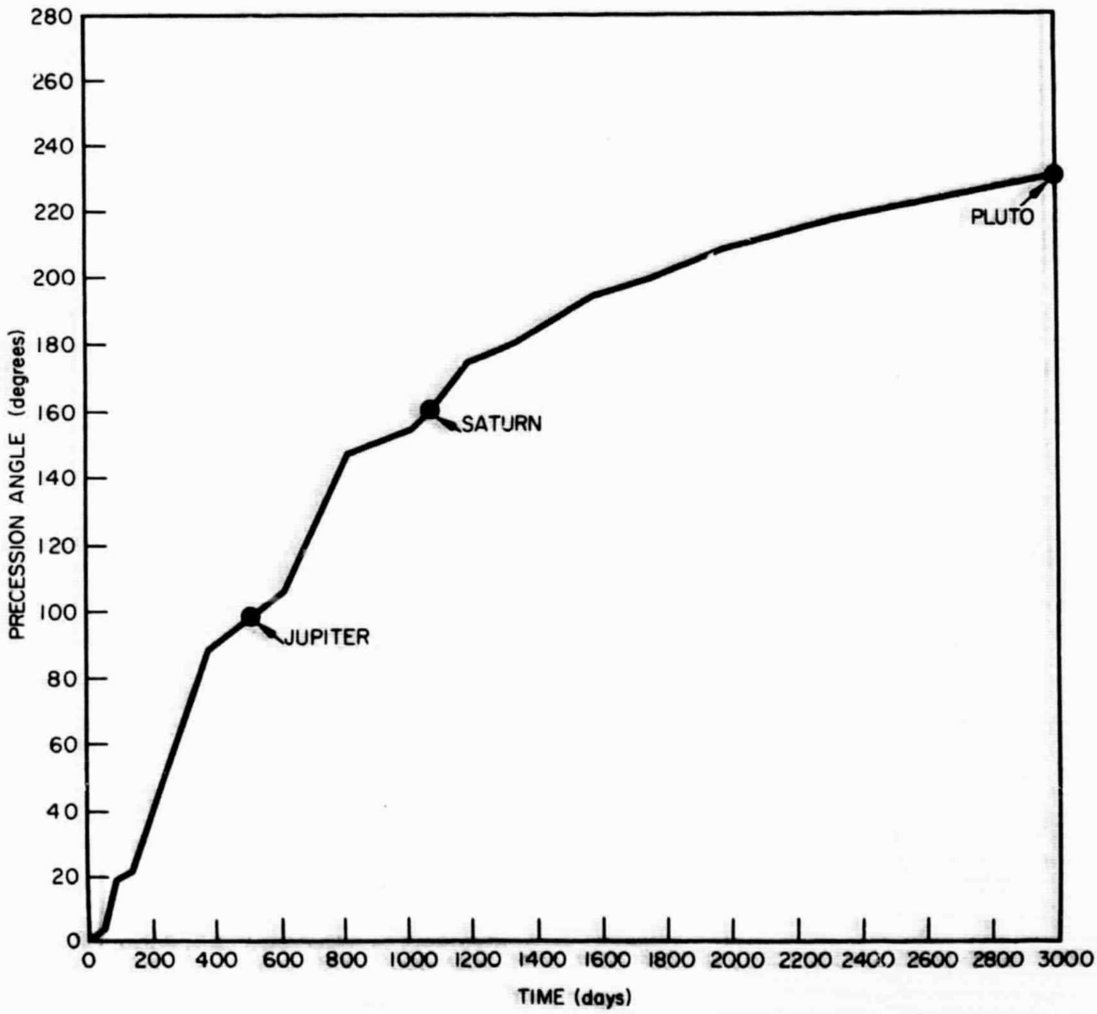

Figure 12. Required Spin-Axis Precession to Track Earth (1977 Earth/Jupiter/Saturn/Pluto)

Backup Earth-Track Control Mode

By using sun and Canopus sensor information, the spin axis can be made to offset the point from the sun and thereby acquire and track the earth. Figure 13 shows precession geometry. In general, two precession maneuvers are required. In order to point toward the earth, the spin axis must be precessed through the angles α_c and β_c. The spin axis is first precessed as previously described along a great circle toward the sunline until $\alpha_s \approx \alpha_c$. The β precession about the sunline is obtained by time-delaying the jet bursts an additional 90 degrees of the spin cycle. This precession is terminated when $\beta \approx \beta_c$.

Figure 14 shows the sun/spacecraft/earth angle (α_c) and the angle between the plane formed by the earthline/sunline and the earthline/Canopus line (β_c) for a 1977 Earth/Jupiter/Saturn/Pluto mission. Command angles α_c and β_c will be periodically computed on the ground and stored on board the spacecraft.

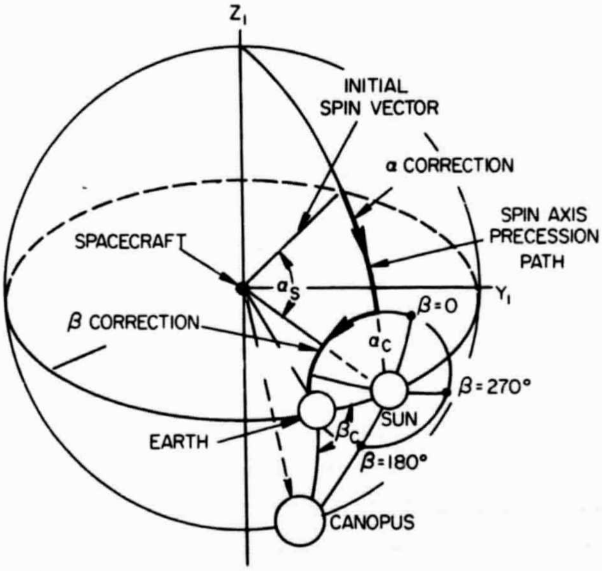

Figure 13. Sun/Canopus System Geometry

The time between the Canopus and sun pulses and the spin rate obtained from successive pulses from the sensors will permit on-board computation of β.

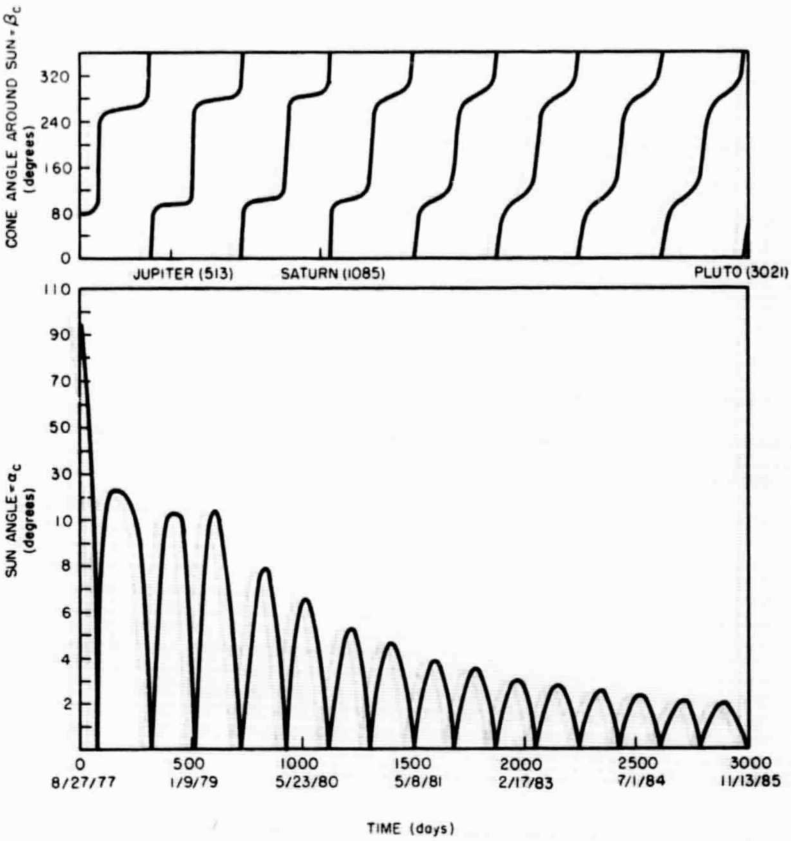

Figure 14. Sun/Canopus System Command Angles vs Time

Figure 15 is a block diagram of one possible implementation of the backup control system. Commanded angles α_c and β_c are stored in complemented form in their respective command registers. The basic function of the control system is to sequentially drive α_s to α_c and β to β_c. When the sun crosses the sun-sensor plane, the sun angle is stored in the α_s register. The α_s (actual) and α_c (commanded) angles are subtracted from each other and stored in the α_s - α_c register. The sign bit signal from this register is fed to either AND gate 2 or AND gate 4. The difference ($\alpha_s - \alpha_c$) is shifted into a comparator circuit. If the error is greater than a commanded or stored $\Delta\alpha$, a signal is sent to AND gates 2 and 4, and additional AND gates 1 and 3 are inhibited to prevent β correction. Thus, the jet is fired at the proper time to reduce the α error.

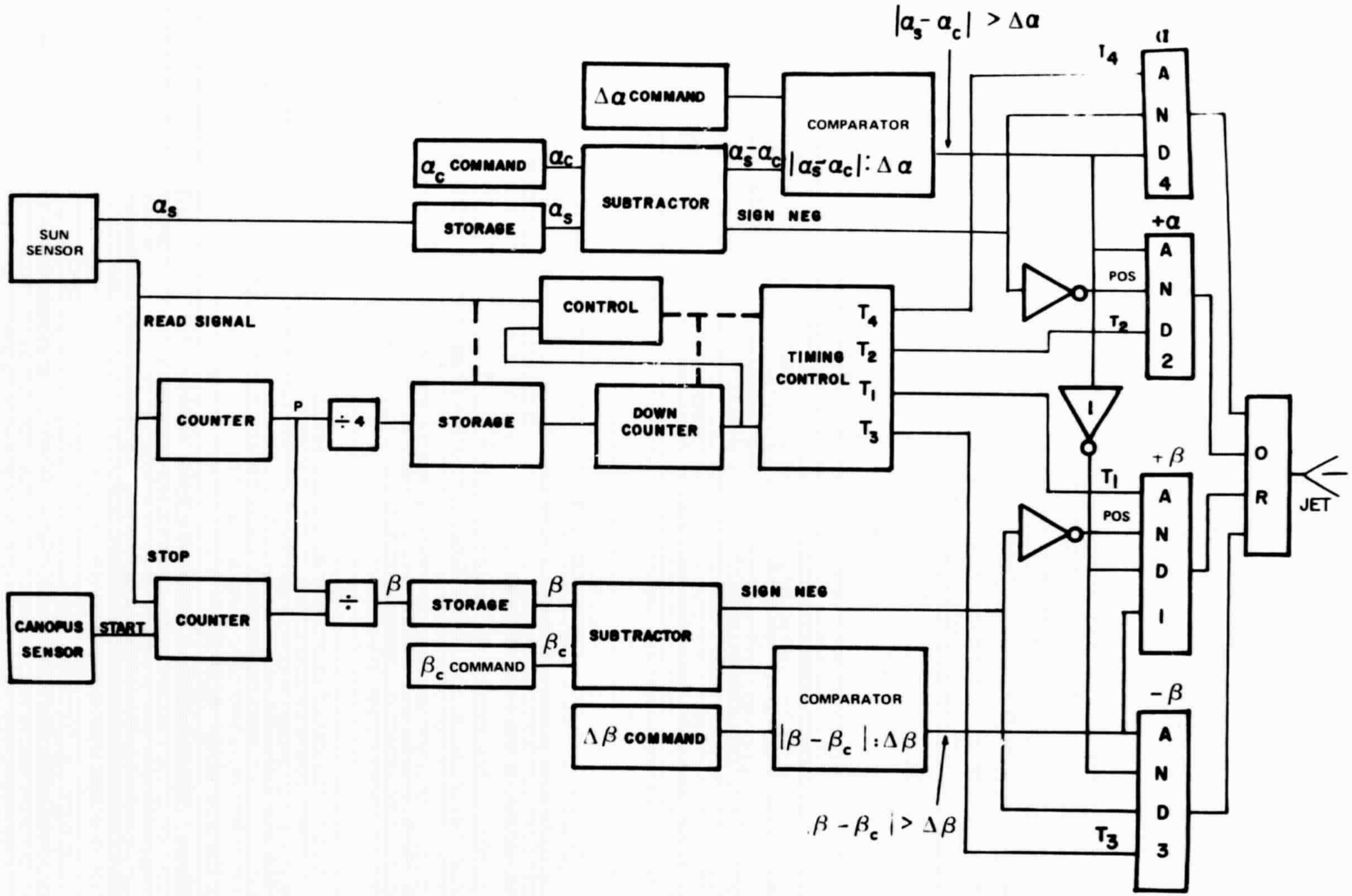

Figure 15. Sun/Canopus Backup Precession-Control System

The momentum vector will be precessed toward the sunline until the magnitude $|\alpha_s - \alpha_c|$ is less than $\Delta\alpha$. When this occurs, the output of the comparator becomes 0, AND gates 2 and 4 are inhibited, and the β gates are enabled through inverter 1. At this time, the probe momentum vector lies on a cone approximately α_c degrees from the sunline. Momentum-vector motion around the periphery of the cone is initiated, using Canopus and sun-sensor information.

The Canopus sensor will initiate a pulse when the Canopus line lies in a specified spacecraft plane and is within the sensor's 40-degree fan field-of-view. The sun sensor will also initiate pulses when the sun is coincident with a given spacecraft plane. These two pulses control the starting and stopping of the β counter. The counter measures the time difference between the sun and Canopus pulses which is proportional to the angle β and generates an error signal equal to $(\beta - \beta_c)$.

The error signal is stored and its sign bit is used to enable AND gates 1 and 3. The error signal is then fed into a comparator where it is compared to dead-space angle $\Delta\beta$. If the error signal is outside the dead space, AND gates 1 and 3 are enabled, permitting spin-axis precession along the cone. This precession pulsing will continue until $|\beta - \beta_c|$ is within dead zone $\Delta\beta$. At this time, the spin axis will be pointed to the commanded initial position, α_c and β_c, to within the accuracy allowed by the control system dead zone.

The preceding paragraphs defined the control law and logic of this system. The following paragraphs show how the precise time of jet firings is obtained. The output of the sun-sensor "read" signal is used to turn the P counter on and off. The value stored in the counter is the period of 1 revolution of the spacecraft. The counter contents are divided by 4 and stored. The stored value is the time of 0.25 revolution. This value is shifted into the down counter and is also retained in the storage register.

When the next sun pulse is received, down counter 1 begins counting down. When the contents are zero, a pulse is sent to AND gate 2, displaced in time T/4 seconds. The counter is immediately reset and begins counting down again. At the end of the next three successive countdowns, AND gates 3, 4, and 1 are pulsed. This operation continues until another sample of P is commanded or programmed. This method of varying the timing with the spin rates reduces the requirements of maintaining a tight spin-rate control.

Emergency Search Mode

Loss of communications with the ground for a programmed period will automatically initiate an emergency search mode. Two spin-axis precession patterns will be programmed as dictated by the spacecraft trajectory with respect to the ecliptic plane and by the current-peak angle between the sunline and the earthline.

Near Ecliptic ($\alpha_{c\epsilon}$ Small or Large). When the spacecraft is near the ecliptic plane, the spin-axis precession pattern will be a continuous precession into the sunline. If communications are not reestablished, a small precession (2-degree) will occur away from the sun in any arbitrary direction. The spin axis will then be precessed about the sunline until the stored $\beta_{c\epsilon}$ is obtained. This maneuver will essentially place the spin axis in the ecliptic plane.

The spin axis will then be precessed away from the sunline in the ecliptic plane in step-dwell-step fashion until the communications link is reestablished or until the angle $\alpha_{c\epsilon}$ is obtained. This angle, corresponding to the peak earth/spacecraft/sun angle shown in Figure 16, will be updated every 6 to 12 months during the early portion of the mission and will be held constant in the latter portion of the mission. Because the spacecraft will be close to the ecliptic plane, the $\beta_{c\epsilon}$ angle (similar to β_c of Figure 14) will be relatively constant for long intervals in the early portion of the mission and can therefore be updated similar to $\alpha_{c\epsilon}$. If the search maneuver is not successful, the spin axis will be precessed into the sun to await earth's intersection of the spacecraft/sunline.

Out-of-Ecliptic ($\alpha_{c\epsilon}$ Small). When $\alpha_{c\epsilon}$ is relatively small and the spacecraft is significantly out of the ecliptic plane so that $\beta_{c\epsilon}$ requires updating too frequently, another search pattern will be used which requires only the sun sensor. The spin axis will again be precessed off the sunline in about 1-degree steps. However, following each $\Delta\alpha$ step, it will be precessed completely about the sunline in a step-dwell-step manner. Although the search pattern is foolproof, it is limited to small $\alpha_{c\epsilon}$ because of fuel consumption considerations.

Spin Rate Trim

Ground command will periodically trim the spin rate to the 1-rpm nominal rate by generating open-loop pulse commands to the spin jets. It can also command a closed-loop trim using the biased-rate gyro-feedback loop of the planet-tracking system to control jet initiation and cutoff.

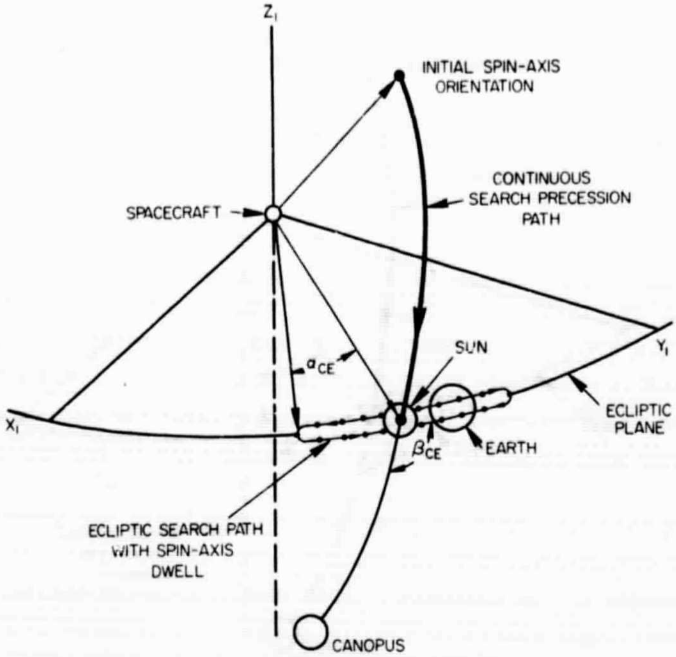

Figure 16. Sun/Canopus System Emergency Search Geometry

PLANETARY-ENCOUNTER CONTROL-SYSTEM OPERATION

Figure 17 shows the planetary-encounter control system. The system contains three normal modes of operation (search, track, and program) and three backup control modes (gyro malfunction, planet-tracker malfunction, and antenna offset from earth).

Normal Modes

Search Mode. Upon ground command, the system will switch from the cruise mode to the planet-search mode. The rate feedback signal will cause the wheel to speed up and absorb most of the spacecraft 100 ft-lb-sec momentum. The search-rate bias will limit the despinning of the spacecraft at 1/60 rpm. The planet tracker will look dead astern and its arms will turn about the earth-line. When the planet enters a specific arm (e.g., the $+\delta$ of Figure 4), a track command will put the system in the track mode.

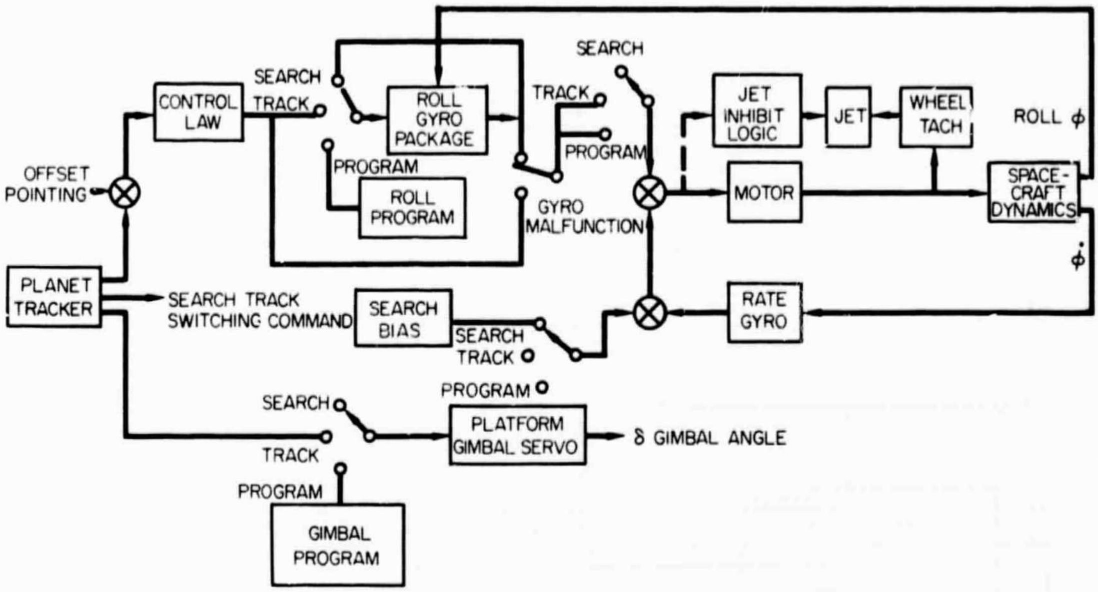

Figure 17. Planetary-Encounter Control System

Track Mode. The presence of a track command will cause three system changes: the rate-gyro bias will switch to zero; the roll gyro will uncage, providing a roll reference; and the scientific-platform servoloop will close, permitting the gimbal to drive and capture the planet in the center of its field-of-view. The roll-control law will modify the planet-tracker roll-error signal, α, to provide a torque-updating signal for the roll-rate integrating gyro. Because α is not equal to roll error but is equal to $\phi_e \sin \delta$ where ϕ_e is roll error and δ is the scientific platform gimbal angle, the α angle from the planet tracker must be divided by $\sin \delta$. This yields ϕ_e, which will be used to update the tracker signal. The output from the roll-control law will then represent the roll-rate signal required to maintain the spacecraft on the planet in roll (i.e., the spacecraft will be rolled so that the plane normal to the scientific-platform gimbal axis will contain the planet. The roll axis will be pointed toward the earth and will be gyroscopically stabilized by the large roll angular momentum stored in the roll-reaction wheel. If offset pointing is required during planetary encounter, program offset-pointing signals can be injected as shown in Figure 18.

The scientific-platform gimbal-axis loop is also a rate loop (Figure 18). The planet-tracker output represents the gimbal tracking error. The control law processes the error to provide the proper rate on the gimbal drive, causing the scientific platform to track the planet.

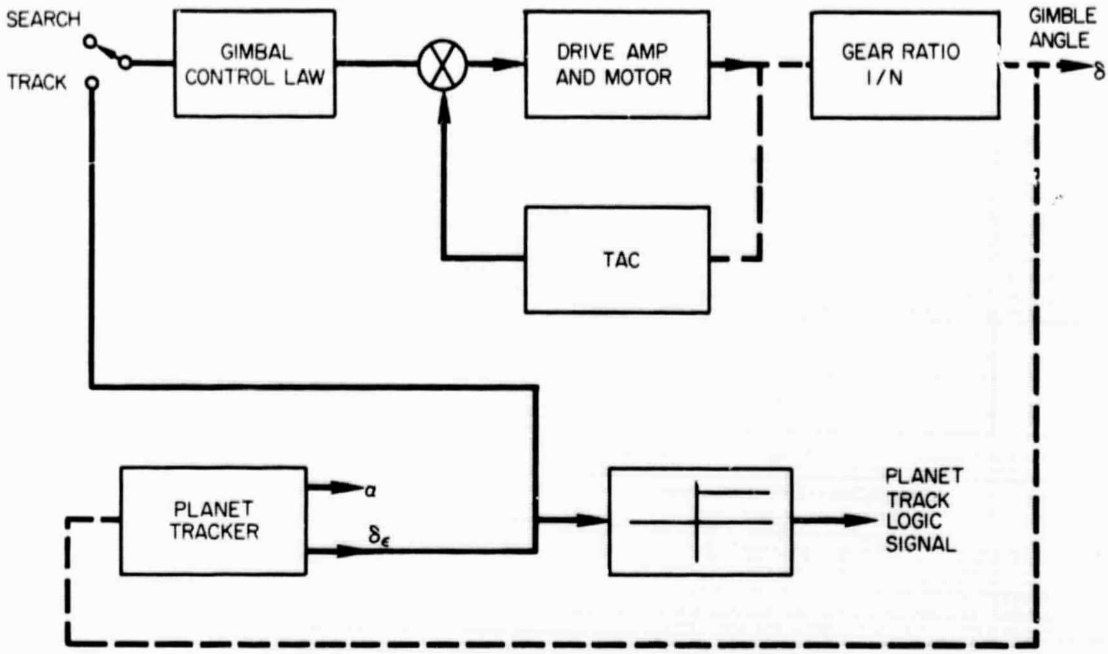

Figure 18. Scientific Platform Gimbal Stabilization Loop

Because the output of the planet tracker is a digital signal, both the roll-control law and the gimbal-control law will consist of digital electronics. Simple counting registers will accumulate the ϕ_ϵ and δ_ϵ error signals at a specified clocking rate, integrating the error signals. The integrated error signals, summed with a signal proportional to error, will generate the appropriate rate signals to drive the roll and gimbal-axis control loop.

In the normal mode of operation, the jet-inhibit logic will provide automatic speed control on the wheel to keep the wheel within an acceptable tolerance about the nominal 500-rpm operating point without unnecessarily expending fuel. For example, during initial speedup, the tachometer-measured speed relative to the desired bias speed will call for plus roll jets. However, because the motor command will also be plus, these jets will be disabled. Conversely, if external torques cause the wheel speed to exceed 650 rpm while tracking the planet, the negative jet will actuate to unload the wheel. Planet track will still be maintained. When the induced negative spacecraft position and rate commands to the motor exceed the negative inhibit threshold, the jets will be disabled. This mechanization will permit the wheel to be incrementally unloaded until its speed is below 560 rpm.

Despinning the spacecraft from 1 rpm to 0.1 degree per second (1 revolution per hour) will take less than 0.5 hour. Planet acquisition may require as much as 1 hour after despin if the spacecraft must make a full revolution before finding the planet. Actual acquisition will take place about 5 minutes after the planet comes into the field-of-view of the planet tracker. Total despin and acquisition will take 1.5 hours or less.

The reason for torquing the gyro with the roll command rather than directly routing it to the motor loop is that it will provide smoother roll-loop operation because of the filtering effect of the gyro. Also, it will provide an immediate roll reference which will permit continuous pointing when the planet tracker loses reference on the dark side of the planet. At this point, the system will be switched to the program mode.

Program Mode. In the program mode, gimbal-angle and gyro-torquer commands from the on-board stored program will permit continuous updating of platform pointing during dark-side encounter even without planet-tracker reference.

Backup Control Modes

Gyro Malfunction. If the two rate-integrating gyros with cross-strapped electronics malfunction, the planet-tracker roll commands will be routed directly to the momentum-wheel motor. In this configuration, planet acquisition will be obtained by programming a gimbal-angle slew maneuver while in the search mode.

Planet-Tracker Malfunction. If both planet trackers (one on each platform) become inoperative, the spacecraft will perform the normal roll search using the Canopus sensor instead of uncaging the gyro through the planet tracker. The Canopus sensor will provide an inertial reference from which an on-board programmer can control the roll and gimbal angles in open-loop fashion. Figure 19 shows the nominal program for a Jupiter encounter during the 1979 Earth/Jupiter/Uranus/Neptune mission.

Antenna Offset from Earth. The trajectory geometry dictates that, without antenna-axis pointing control, the angle between the antenna axis and spacecraft/earthline will change a maximum of 0.2 degree during the Jupiter encounter (up to RCA). This angular change will be much less for the more distant planets. Although the external torques discussed later probably will not cause the pointing error to exceed 0.5 degree, an antenna-axis encounter-precession system (Figure 20) would be desirable from a backup standpoint.

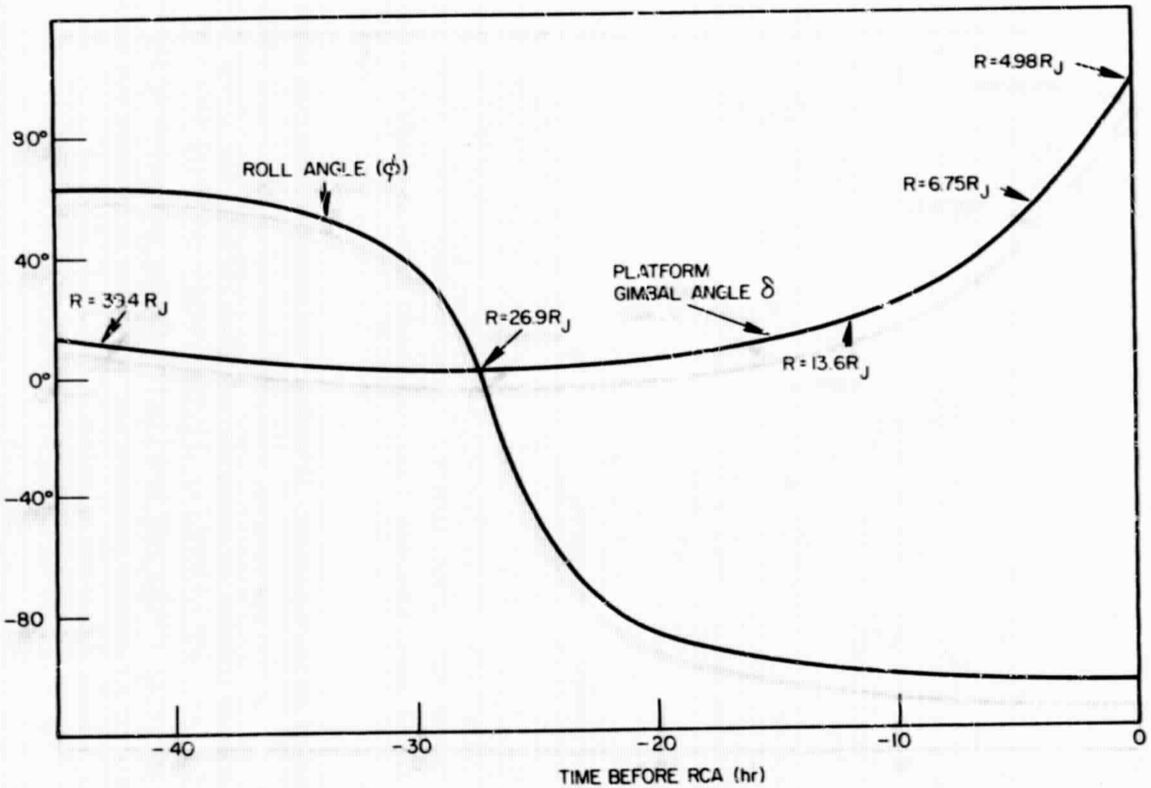

Figure 19. 1979 Jupiter Encounter ACS Control Angles (Jupiter/Uranus/Neptune Mission)

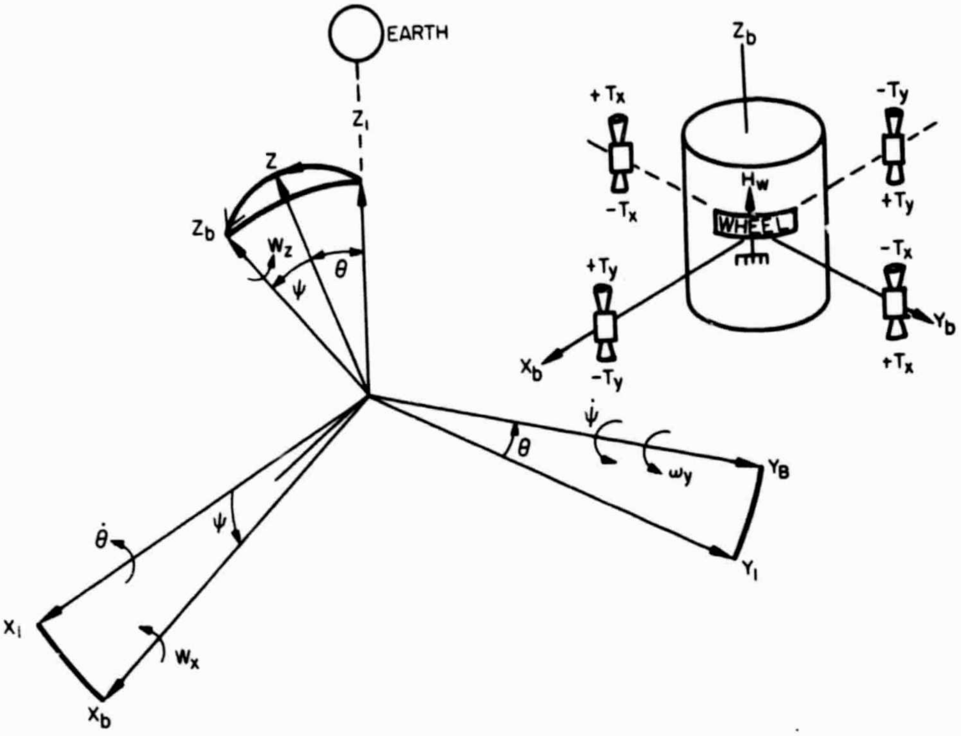

Figure 20. Antenna-Axis Encounter-Precession Geometry

Loss of spacecraft spin motion and the desire to keep the scientific platform locked to the planet increase the complexity of the encounter-precession system relative to the simplified cruise system. Because the RF earth sensors must statically detect two axes of information (pitch (θ) and yaw (ψ) error), another set of interferometer horns will be added with their baseline perpendicular to the baseline of the cruise interferometer horns. Moreover, six additional jets (torque couples) will be required to reduce these errors without rolling the spacecraft about the antenna axis and possibly causing the scientific platform to lose planet track.

Linearized equations of motion relative to a spacecraft fixed-axis system with equal transverse moment-of-inertia are:

$$\overset{\circ}{\omega}_x + \frac{H_w}{A}\,\omega_y = \frac{T_x}{A}$$

$$\overset{\circ}{\omega}_y - \frac{H_w}{A}\,\omega_x = \frac{T_y}{A}$$

$$\overset{\circ}{\omega}_z = 0, \ \omega_z = 0$$

where

H_w = biased wheel momentum × 100 ft-lb-sec

T_x = jet torque about X-body axis

T_y = jet torque about Y-body axis

Position of the antenna axis relative to the earthline can be approximated as:

$$\theta = \int W_x \, dt$$

$$\psi = \int W_y \, dt$$

The solution of these equations for a step torque about the spacecraft X-axis and the initial rate, zero, is:

$$\omega_x = \frac{T_x}{H_w} \sin \frac{H_w}{A} t$$

$$\omega_y = \frac{T_x}{H_w} \left(1 - \cos \frac{H_w}{A} t\right)$$

$$\theta = \frac{T_x^A}{H_w^2} \left(1 - \cos \frac{H_w}{A} t\right)$$

$$\psi = \frac{T_x}{H_w} \left(t - \frac{A}{H_w} \sin \frac{H_w}{A} t\right)$$

In one nutation period ($t = (2\pi A/H_w)$), the antenna axis has precessed in yaw 57.3 ($2\pi T_x \; A/H_w^2$) degrees. However, the coupled pitch motion peaks at 57.3 ($2T_x \, A/H_w^2$) degrees. Therefore, in order to keep the peak pitch motion below 0.5 degrees while the antenna is being inertially precessed in yaw, the jet torque must be less than:

$$T_x = \frac{0.25}{57.3} \frac{H_w^2}{A}$$

Using the cruise system characteristics previously listed and a 100 ft-lb-sec wheel momentum, the maximum thrust level is 0.02 pound, which is well below the present 0.5-pound thrust level used in the cruise precession system. Further study is necessary to determine whether it is practical to reduce this thrust level so that a constant-thrust encounter-precession system is possible.

An alternate approach, similar to the cruise technique, is to apply the torque for a short interval relative to the 37-second nutation period. After the first jet pulse ($T_x \Delta t$), the solution to the equation of motion is:

$$\omega_x(t) = \frac{T_x \Delta t}{A} \cos \frac{H_w}{A} t$$

$$\omega_y(t) = \frac{T_x \Delta t}{A} \sin \frac{H_w}{A} t$$

$$\theta(t) = \frac{T_x \Delta t}{H_w} \sin \frac{H_w}{A} t$$

$$\psi(t) = \frac{T_x \Delta t}{H_w} \left(1 - \cos \frac{H_w}{A} t\right)$$

Owing to the wheel-momentum bias, the initial momentum vector along the spacecraft antenna axis now has an additional component in the inertial direction of the X-axis at the time of the jet pulse. After the impulse, the momentum vector is inertially changed approximately 57.3 ($T\Delta t/H_w$) degrees. To conserve momentum, the antenna axis is caused to nutate about the new momentum vector. This inertial motion is obtained by transferring the initial body-axis rate ($W_x(0) = T\Delta t/A$) between X- and Y-body axes as shown in the equations on motion. When the antenna axis nutates 180 degrees, the body rate is again completely contained about the X-axis, but it has a negative polarity. By again firing the same jet, the transverse rate is eliminated and the momentum vector is again aligned along the antenna axis, which has moved inertially 57.3 ($2T\Delta t/H_w$) degrees.

Figure 21 is a block diagram of the time-delayed thrust-pulse precession technique. When the RF-interferometer earth-sensor pitch (θ_ϵ) and yaw (ψ_ϵ) errors exceed 0.5 degree, the appropriate jets will be pulsed for approximately 40 milliseconds, resulting in a 0.15-degree cone angle. After approximately 18 seconds, a time-delayed jet firing should decrease the error below the 0.25-degree cutoff threshold. If more correction is required, another set of pulses will be generated. Time-delay error, nonrepeatibility of the jet-thrust level, pulsewidth, and uncertainty in the nutation frequency which determines the time delay will cause some residual off-axis momentum or body rates after the delayed jet firing. The passive damper will remove the nutational rates.

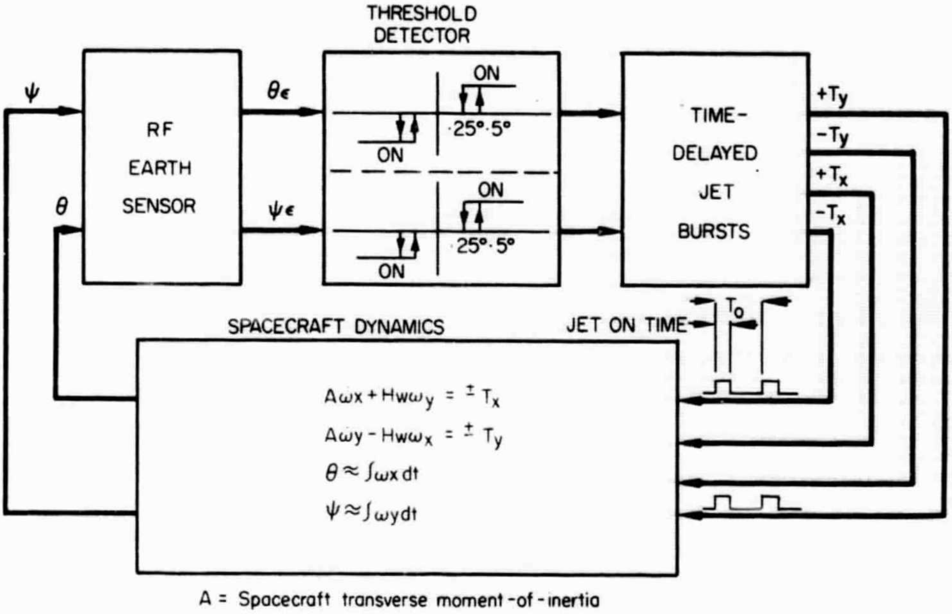

Figure 21. Encounter-Precession System, Block Diagram

EXTERNAL DISTURBANCE TORQUES

To evaluate the efficiency of the 100 ft-lb-sec biased-momentum wheel in maintaining antenna-axis direction, the effects of the disturbance torques during the Jupiter encounter were estimated. The two primary attitude disturbances appear to be magnetic and gravity-gradient torques.

Magnetic torque on the spacecraft is:

$$\bar{T}_m = \bar{M} \times \bar{B}$$

where

M = spacecraft dipole moment

B = ambient field

The worst-case condition of momentum-vector precession exists when the torque is normal to the momentum vector. Assuming a worst-case dipole along the roll axis and the B vector orthogonal to it, a computer check produced the roll-axis precession angle shown in Figure 22. The assumed dipole and momentum vectors were 500 pole-cm and 100 ft-lb-sec, respectively. For a 1-gauss field at three Jupiter radii, the offset at RCA is about 0.13 for an estimate of the effect of gravity-gradient torques. During encounter, only a single-axis torque was assumed (i.e., the roll axis was assumed to lie in the orbit plane) which may be represented by

$$T_g = \frac{3U}{R^3} \ I \sin \theta \cos \theta$$

where

U = gravitational constant of Jupiter

θ = orbit true anomaly

R = radius to the spacecraft

I = difference in inertia between the roll axis and the transverse axis

The effect on this torque, which is orthogonal to the momentum vector (Figure 22), is small compared to the magnetic effect.

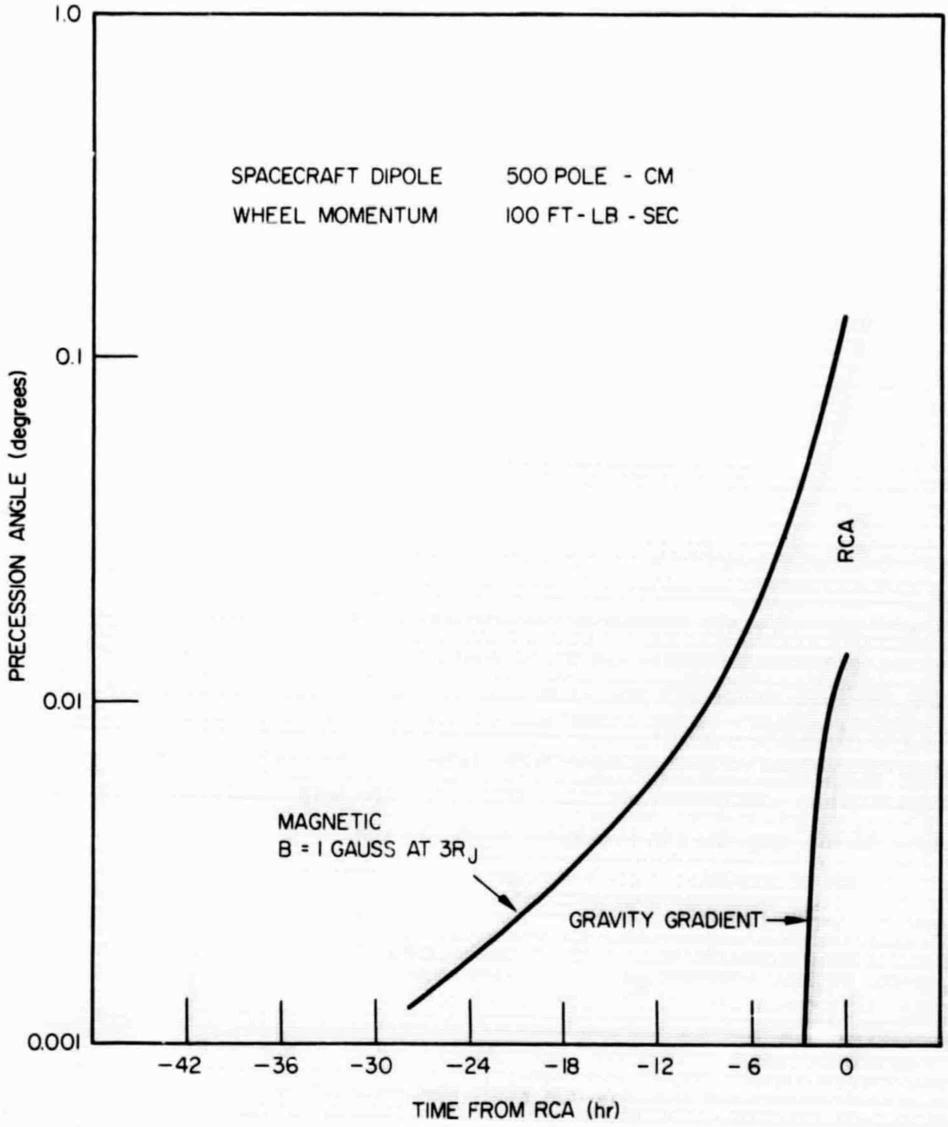

Figure 22. Precession Angle Caused by External Torque During Jupiter Encounter

To check the effect of solar-pressure torques, an average torque equal to the magnetic torque was assumed to act on the spacecraft. Using the solar constant at Jupiter and a solid 15-foot antenna, the distance between the center-of-mass and center-of-pressure necessary to produce that torque is about 3 feet, a long distance. Therefore, the effect of solar torques during encounter should also be small compared to the magnetic torques.

The wheel should be able to absorb the effects of any roll torques. For example, if the 500 pole-cm dipole moment were normal to the roll axis, the wheel would have to absorb the resulting ΔH if the spacecraft is anchored to the planet. For the foregoing numbers, ΔH would be about 2 ft-lb-sec, or about 2 percent of the bias momentum.

FUTURE DEVELOPMENT

The abnormal environment (e.g., the high radiation at Jupiter) and the long mission life necessitate that some control-system components be better than present state-of-the-art. The Honeywell Corporation is developing a solid-state spinning Canopus sensor under contract NAS5-11238. Hopefully, the same design principle will apply to the development of the planet tracker. Also, the Stabilization and Control Branch, GSFC, is analyzing the scientific-platform roll and gimbal drive servoloops on the basis of a digitized planet-tracker error signal. Other components which require analyses and hardware tests are:

- Coning damper
- Momentum wheel
- RF earth sensors
- Sun sensors
- System dynamics (including flexible appendages)