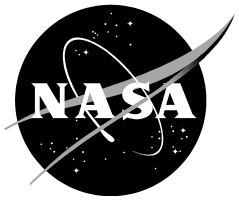
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Pioneer F/G: Spacecraft Operational Characteristics

Charlie F. Hall
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May 1971

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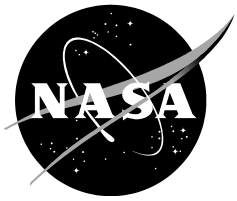
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Acknowledgments

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PIONEER PROGRAM

National Aeronautics and Space Administration
Ames Research Center
Moffett Field, California

PIONEER F/G PROJECT

PIONEER F/G: SPACECRAFT OPERATIONAL CHARACTERISTICS

DOCUMENT NO. PC-202

Richard O. Fimmel

PIONEER PROGRAM

National Aeronautics and Space Administration
Ames Research Center
Moffett Field, California

APPROVAL:

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PC-202.01

PIONEER F/G: SPACECRAFT OPERATIONAL CHARACTERISTICS

DESCRIPTION OF SPACECRAFT

This section was prepared for
Ames Research Center Under
Contract NAS2-6255 by TRW Systems
Group, Redondo Beach, California.

PIONEER F/G SPACECRAFT OPERATIONAL CHARACTERISTICS

1. INTRODUCTION

1.1 STUDY

The Pioneer F/G Spacecraft Operational Characteristics Study was performed by TRW Systems Group for the National Aeronautics and Space Administration, Ames Research Center (NASA/ARC) between November 1970 and April 1971, under contract NAS2-6255. The purpose of this study is to describe the operational characteristics of the Pioneer F and G spacecraft system and its subsystems to the NASA/ARC Pioneer Project personnel having the responsibility for conducting the Pioneer F/G flight mission operations.

This report, "Pioneer F/G Spacecraft Operational Characteristics," is the final report of this study.

1.2 DOCUMENT

PC-202, "Description of Spacecraft and Scientific Instruments," will be a document issued and controlled by the NASA/ARC Pioneer Project Office (PPO). Its purpose will be to embody descriptions of the significant characteristics of the spacecraft and on-board scientific instruments. It is anticipated the body of this report and similar information which relates to the characteristics of the scientific instruments will comprise document PC-202.

This report treats the spacecraft as a system, and it treats its subsystems and assemblies. While the radioisotope thermoelectric generator (RTG) power sources are government furnished equipment to the spacecraft contractor, they are treated herein as a component subsystem of the spacecraft.

This report describes the interactions between the spacecraft and the ground data system (command source for uplink information transmission; telemetry terminal for downlink information transmission); among the various subsystems of the spacecraft; and between the subsystems of the spacecraft and the scientific instruments. The characteristics of the scientific instruments themselves are not treated in this report.

1.3 ORGANIZATION

Section 2 is a summary description of the spacecraft at the system level. It includes descriptions of mission phases, system interfaces, and the capabilities and limitations of performance of system functions.

Section 3 presents descriptions of spacecraft operation at the subsystem and assembly level. The level of detail of these descriptions varies somewhat from subsystem to subsystem; it is guided generally by the criterion that, for operational purposes, the significant factors are what can be learned of the state of the spacecraft from the data which are telemetered from the spacecraft and what influences can be imposed on the state of the

spacecraft by commands transmitted to it. In electronic subsystems and assemblies, this level usually includes fairly detailed logic diagrams of the functions performed within the assemblies (units), but stops short of schematic circuit diagrams.

Section 4 describes how the spacecraft performs in operating modes which correspond to the performance of specific functions at the time of specific events in the mission. These operations generally require the coordinated performance of more than one subsystem. Discussion is restricted to the normal performance of these operations.

Section 5 presents the principal back-up means of performing the spacecraft operating modes of Section 4.

Section 6 presents the anticipated telemetered data expected from the spacecraft as normal response during certain predictable mission events. Emphasis is on the events from prelaunch through the initial orientation of the spacecraft following launch.

Portions of this report may be incomplete or subject to later revision, particularly where subsequent tests, e.g., calibrations of flight sensors, will provide pertinent data.

1.4 APPLICABLE DOCUMENTS

NASA/ARC Specifications

- PC-210.03 Spacecraft Requirements
- PC-220 Spacecraft/Scientific Instrument
- PC-221 Pioneer F Spacecraft/Launch Vehicle Interface Specification
- PC-222 Pioneer F and G Spacecraft/DSIF Interface Specification
- PC-223 Spacecraft/RTG Interface Requirements
- PC-224 ARC/DSN Ground Data System Interface Specification
- PC-230 Mission Operations Plan
- PC-263 Pioneer F Computer Program for Midcourse Maneuver--Functional Requirements
- PC-292 Pioneer F Trajectory Characteristics

2. SPACECRAFT SYSTEM DESCRIPTION

2.1 GENERAL DESCRIPTION

The Pioneer F/G Project has the following mission objectives:

a) Primary. To conduct, during the 1972 and 1973 Jovian opportunities, exploratory investigations beyond the orbit of Mars of the interplanetary medium, the nature of the Asteroid Belt and the environmental and atmospheric characteristics of the planet Jupiter.

b) Secondary. To advance the technology and operational capability for long duration flights to the outer planets.

The Pioneer F and G Spacecraft are launched by the launch vehicles, and at the times shown:

Spacecraft	Launch Vehicles	Launch Period
F	Atlas SLV-3C/Centaur D/TE-364-4	2/28/72 to 3/16/72
G	Atlas SLV/3C/Centaur D-1A/TE-364-4	4/03/73 to 4/20/73

Trip times from launch to encounter with Jupiter will be in the range of 600 to 900 days.

Each spacecraft will weigh approximately 550 lb at launch, which includes 63 lb for the 11 scientific instruments on board, 118 lb for the four radioisotope thermoelectric generator (RTG) power sources, and 60 lb of propellant which is expendable. The spacecraft is spin stabilized, spinning in the cruise configuration at a nominal 4.8 rpm about an axis directed toward the earth.

2.1.1 Mission Description

The Pioneer F and G missions will provide the initial direct exploration of a vast region of the solar system from 1.5 to 6 astronomical units (AU) from the sun, including the vicinity of the planet Jupiter. Within this region is the asteroid belt, lying principally within 10 deg of the plane of the ecliptic and between 2 and 3.6 AU from the sun. Pioneer F will be the first spacecraft to expolre this region beyond 1.5 AU. (Pioneers 7 and 8 have explored out to 1.13 AU. Mariners 4, 6, and 7 (Mars flybys) returned data from 1.5 AU. Mariner '71 will also traverse interplanetary space to 1.5 AU from the sun; however, like Mariners 6 and 7, its scientific payload is primarily directed at exploring Mars.)

Nearly all the time from launch to encounter with Jupiter and after encounter is spent in the interplanetary, solar-wind environment; the spacecraft will depart from all influence of the earth magnetosphere several hours after launch and will not penetrate the Jovian magnetopause until several days before periapsis (closest approach to Jupiter). A diagram indicating the general relationship of a typical trajectory to the Sun, Earth, Asteroid Belt and Jupiter is presented in Figure 2-1.

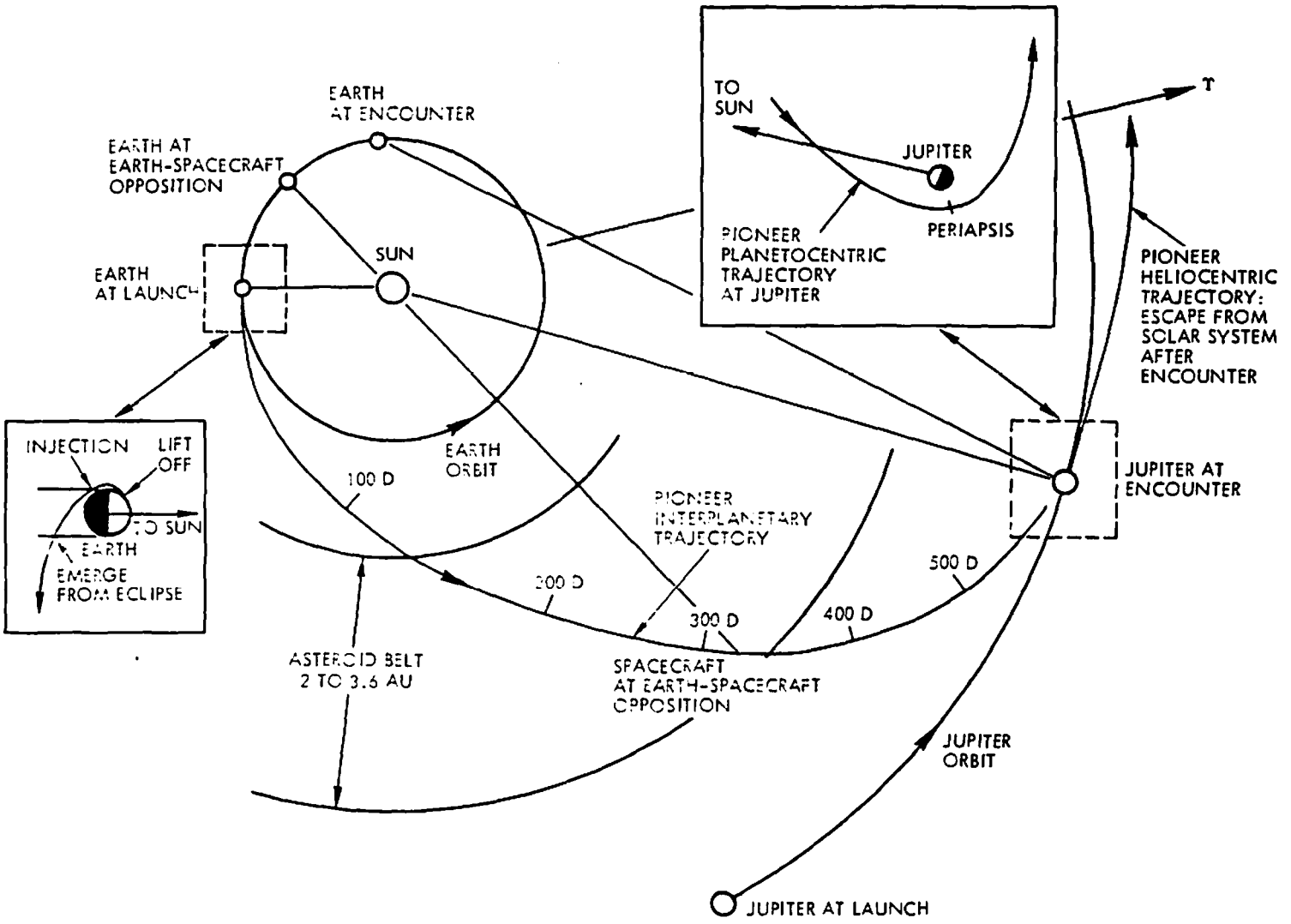


Figure 2-1

PIONEER F MISSION (1972 Launch)

The spacecraft will be directed to perform one or more velocity corrections to compensate for launch vehicle injection errors and to place the spacecraft on a trajectory to the desired target point near Jupiter. It is planned that the first of these maneuvers will be performed within the first 10 days and the second, if necessary, within 30 days after launch.

Precession maneuvers will also be conducted throughout the mission as required to maintain the earth-pointing cruise attitude. These maneuvers, homing on the uplink RF signal received by the spacecraft, will take place every 2 or 3 days early in the mission, and decrease in frequency to one in one or two weeks by the time of Jupiter encounter.

During the long interplanetary cruise phase, the characteristics of particles, fields and zodiacal light will be measured by the scientific instruments and transmitted to Earth.

About 130 days after launch, the spacecraft will be 2 AU from the Sun, and for the next 200 days will pass through the asteroid belt. Here measurements will be conducted which will increase knowledge of the small particle population of the region; here also, may exist a danger of unknown severity to the spacecraft from possible impact by larger size particles.

Approximately 315 days after launch, the motions of the spacecraft and the earth will have placed them in opposite directions from the sun. Because the spacecraft is somewhat out of the ecliptic plane, the spacecraft-earth line will not intersect the sun, but it will come within a few solar radii. This is close enough that radio propagation between the earth and the spacecraft will be influenced significantly by the plasma of the solar corona. It is also close enough that the large receiving antennas on the earth will pick up solar rf noise which will interfere with reception of spacecraft signals for a few days.

A detailed review of the scientific objectives of the mission, as they are to be obtained by the selected complement of scientific instruments has led to a tentative selection of the objectives of the encounter trajectory for Pioneer F and, therefore, also placed limitations on the post-encounter trajectory. These objectives for the encounter trajectory for Pioneer F are as follows:

a) To penetrate the Jupiter radiation-belts

b) To provide good viewing conditions of Jupiter before periapsis

c) To obtain a short occultation of the spacecraft by Jupiter (less than one hour)

d) To provide a radius of closest approach to the center of Jupiter of between 2 and 3 Jupiter radii.

Such a trajectory will approach Jupiter in a counterclockwise direction (as viewed from the north pole) from slightly south of the ecliptic, pass around the the planet in a counterclockwise direction at an altitude

between 1 and 2 Jupiter radii above the surface and exit slightly north of the planet. The post encounter trajectory will have an inclination to the ecliptic of less than 5° and the spacecraft will slowly escape the solar system. An additional 2.5 to 3 years are required to reach a distance from the sun equivalent to Saturn's orbit.

For Pioneer G, the trajectory and sequence of events up to near Jupiter encounter will very similar to that for Pioneer F. At present, the encounter trajectory for Pioneer G has not be selected. However, the prominent options at this time are:

a) One similar to that for Pioneer F

b) A polar passage about Jupiter

2.1.2 Mission Phases

The mission phases of the spacecraft starts with completion of the integration of all subsystems and the scientific instruments. Spacecraft Specification PC-210.00, Paragraph 3.1.2 defines these phases in detail; they are summarized here.

a) Tests. The period of testing of the spacecraft system, from integration of the spacecraft until shipment from the spacecraft contractor's facility to CKAFS.

b) Prelaunch. The period of testing between spacecraft arrival at CKAFS and launch. It includes tests before and after mating to the launch vehicle.

c) Powered Flight. The period from liftoff of the launch vehicle to injection (burnout of its third stage). The duration of this is 13 minutes, 44 seconds.

   At injection, the spacecraft is spinning at 60^{+23}_{-9} rpm, its appendages are still stowed, and it is still attached to the spent third stage by the third stage/spacecraft interstage structure.

d) Post Injection. Includes the period from injection to separation of the spacecraft from the launch vehicle (not over 120 seconds) and the period encompassing the occurrence of events initiated automatically by the spacecraft sequencer. These events include despin, RTG deployment, magnetometer deployment, and initial orientation, the last one ending no later than 164 minutes after liftoff.

   At the end of this phase, the spacecraft will have its appendages deployed and should be spinning at about 4.8 rpm. It should be oriented so that the spin axis (spacecraft +Z axis) is pointing within 10 deg of the direction that the earth will be from the spacecraft about two to four hours after liftoff.

e) Flight. Encompasses almost all the time from launch to encounter. It ends (logically) several days before closest approach to Jupiter, when the first influence of the planet's proximity may be detected by on-board instruments. It includes passage through the Asteroid Belt.

During this phase, two types of propulsive maneuvers will be necessary; periodic earth-tracking maneuvers to keep the spacecraft spin axis directed toward the earth, and several (probably two) midcourse trajectory correction maneuvers. Two other propulsive maneuvers are possible: spin control, to adjust the spin rate; and a late trajectory retargeting, more likely with Pioneer G than with F.

f) Encounter. The period from several days before to 30 days after closest approach to Jupiter. While the spacecraft may undergo one or more earth tracking maneuvers and a spin control maneuver during this period, the major operational functions undertaken are associated with the scientific instruments and the programming of their modes to obtain the maximum information about Jupiter.

g) Post Encounter. The period from 30 days after planetary passage until the spacecraft can no longer perform and telemeter information to the earth.

2.1.3 Spacecraft Physical Description

The prominent physical features of the spacecraft are shown in Figure 2-2.

The spacecraft is spin stabilized in flight, with the spin axis parallel to the high-gain antenna axis (and to the launch vehicle longitudinal axis during the launch phase). A 60 rpm spin rate is attained to stabilize the launch vehicle's third stage firing. After separation from the launch vehicle, the spacecraft's spin rate is reduced partially by firing the despin thruster. Subsequent deployment of the RTGs and magnetometer reduces the spin rate to about 4.8 rpm, the nominal spin rate during the entire cruise phase of the mission. The cruise attitude is with the spin axis directed toward the earth, except for about two hours after launch and for several hour periods associated with the two trajectory correction maneuvers expected within the first month of flight.

The equipment compartment is the central box beneath the large antenna reflector. It contains the electronic units of the spacecraft subsystems and scientific instruments, most of the scientific sensors, and the propellant tank. The equipment compartment is constructed of aluminum honeycomb floor and side wall panels, and is covered on the outside by multilayer insulation, except for necessary apertures and for thermal control louver packets under the floor. The compartment provides temperature control and meteoroid protection for the components located within.

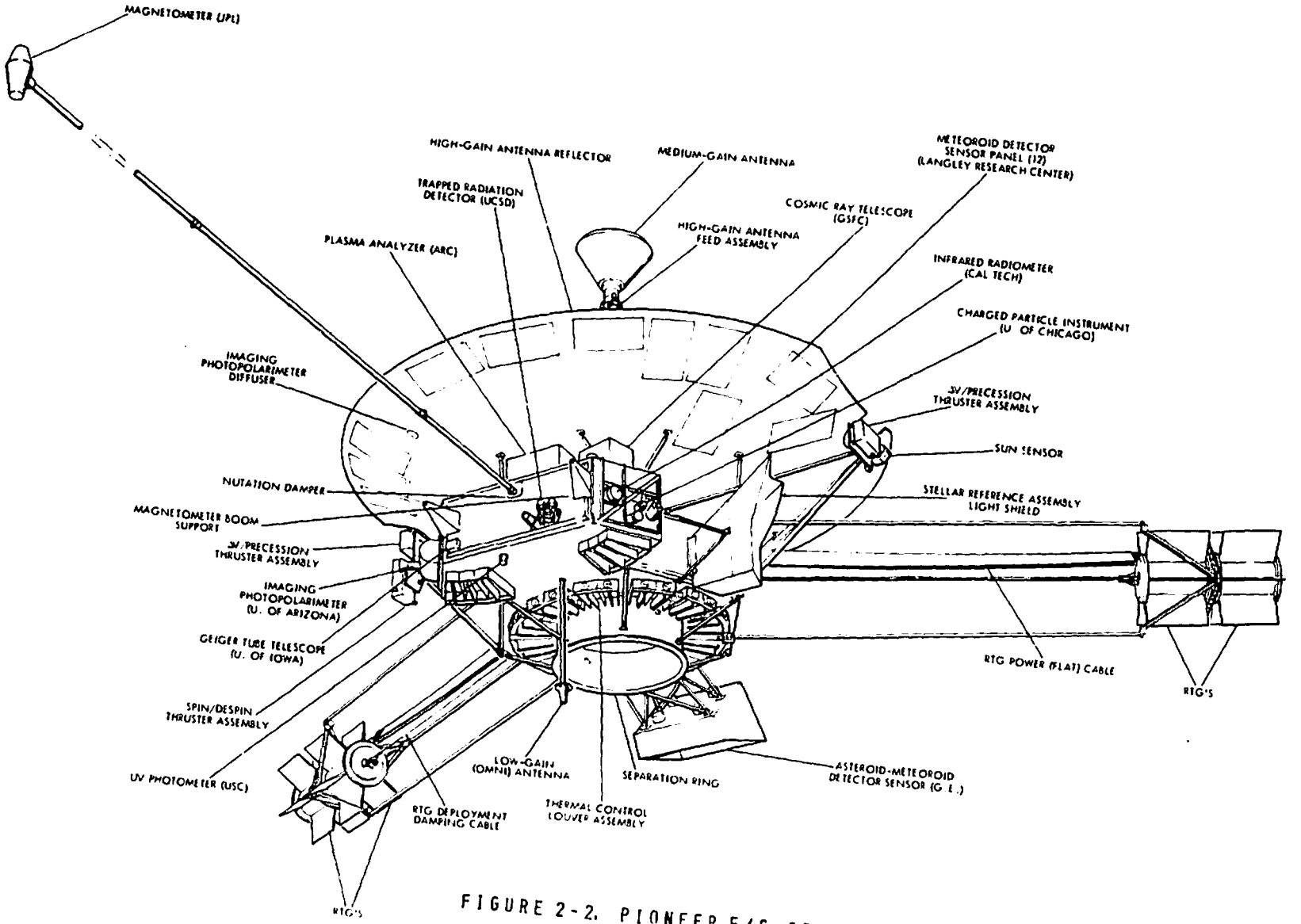


FIGURE 2-2. PIONEER F/G SPACECRAFT

Three communications antennas are visible. The large 9-foot dish is the reflector of the high-gain antenna. The axis of this dish is the spacecraft Z-axis, and is parallel to the spacecraft spin axis within close tolerances throughout the mission. The tripod supports the high-gain antenna feed, as well as a medium-gain horn antenna whose attitude and beam center are canted 9.3 deg from the +Z-axis. The third antenna, a log-conical spiral mounted on a mast in the opposite direction, gives hemispherical coverage in the -Z direction.

Six one-lb hydrazine thrusters are located in three boxes (Thruster Cluster Assemblies) near the perimeter of the 9-ft dish. They are supported by trusses from the equipment compartment. Four of the thrusters thrust parallel to the spin axis for precession and velocity increment maneuvers, and two thrust tangentially for control of the spin rate. The Sun Sensor Assembly is also located at the perimeter of the dish, where it achieves a narrow segment field of view from 1 to 170 deg from the +Z-axis; it is supported by the truss of one TCA. One member of each of these trusses carries a propellant line for the thrusters, an electrical propellant line heater, and electrical cabling to carry valve-driving signals to the thrusters and telemetry data from the TCAs and SSA.

Three appendages are launched in a stowed position close to the equipment compartment, but soon after injection and partial despin are deployed to an extended position. The four RTGs are mounted in two pairs 120 deg apart. Each pair is extended radially (after pyrotechnic bolt cutters are fired) some six feet, under the action of centrifugal force against the restraint of a cable-connected viscous damper. The path and extent of deployment is controlled by three parallel guide rods for each RTG pair. The magnetometer sensor is deployed after the RTGs by unfolding a 17 ft, four-segment boom into the third 120 deg position. This deployment is initiated by a bolt cutter, forced centrifugally and by boom hinge springs, restrained by another viscous damper, and terminated in an accurately held position by hinge latches.

Since the four RTGs are more than 10 times heavier than the magnetometer sensor and boom, the spacecraft center of gravity moves away from the magnetometer (i.e., toward the -X axis) as a result of these deployments. However, the spin axis (principal inertia axis) of the spacecraft remains parallel to the Z-axis, by observing two principles: all the deployments take place in a single plane perpendicular to the Z-axis; and this plane of deployment contains the spacecraft center of gravity. (The deployed magnetometer, whose position is parallel to the XY plane is held accurately by the boom latch springs, is nevertheless freely articulated in the XZ plane. This articulation is coupled to a viscous damper located just within the equipment compartment, and provides the primary damping of any spacecraft wobble motions.)

The purpose in deploying RTGs is to minimize the effects of nuclear radiation--neutrons which could damage spacecraft components (primarily electronic) and gamma ray photons which would interfere with instrument measurements at closer proximity. A consequence is that thermal interaction

between the RTGs and other parts of the spacecraft is reduced to a negligible value. The purpose in deploying the magnetometer is to place it where the magnetic field due to the spacecraft is predictably extremely small.

In addition to the magnetometer, several other instrument sensors are located external to the equipment compartment. Twelve penetrable meteoroid detection panels are attached to the back side of the 9 ft dish, where they have the best exposure to the expected direction of maximum meteoroid flux. An array of four optical asteroid/meteoroid detectors is held below the equipment compartment and generally between the stowed RTG pairs; the viewing direction is 45 deg from the -Z axis. Two instruments, to achieve fovorable fields of view, are located between the equipment compartment and the 9 ft dish. The cosmic-ray instrument views in three different directions, all in a plane perpendicular to the spin axis. The plasma sensor has a fan-shaped field of view whose axis of symmetry is in the +Z direction; it views through an aperature in the dish.

A truss-stabilized cylinder beneath the equipment compartment provides the structural tie to the launch vehicle until two minutes after third stage burnout, when the spacecraft is separated from the final launch vehicle stage.

2.1.4 Launch Vehicle Description

The space booster vehicle for the Pioneer F mission is the three stage Altas/Centaur/TE-364-4. The long tank Atlas is a space booster version of the Series D intercontinential ballistic missile, and when used with the second stage Centaur vehicle is designated the "Atlas SLV-3C." The Centaur is an operational high energy space vehicle using liquid hydrogen for fuel. The Atlas/Centaur system requires the TE-364-4 third stage for Pioneer Jupiter injection velocities.

The General Dynamics/Convair Atlas/Centaur program is managed by the Centaur Project Office, NASA, Lewis Research Center, while the McDonnell/Douglas TE-364-4 program is under the cognizance of the Delta Project Office, NASA Goddard Space Flight Center. Overall launch vehicle management for the Pioneer F mission is the responsibility of Lewis Research Center.

For the Pioneer F launch, a typical performance of the launch vehicle has these characteristics:

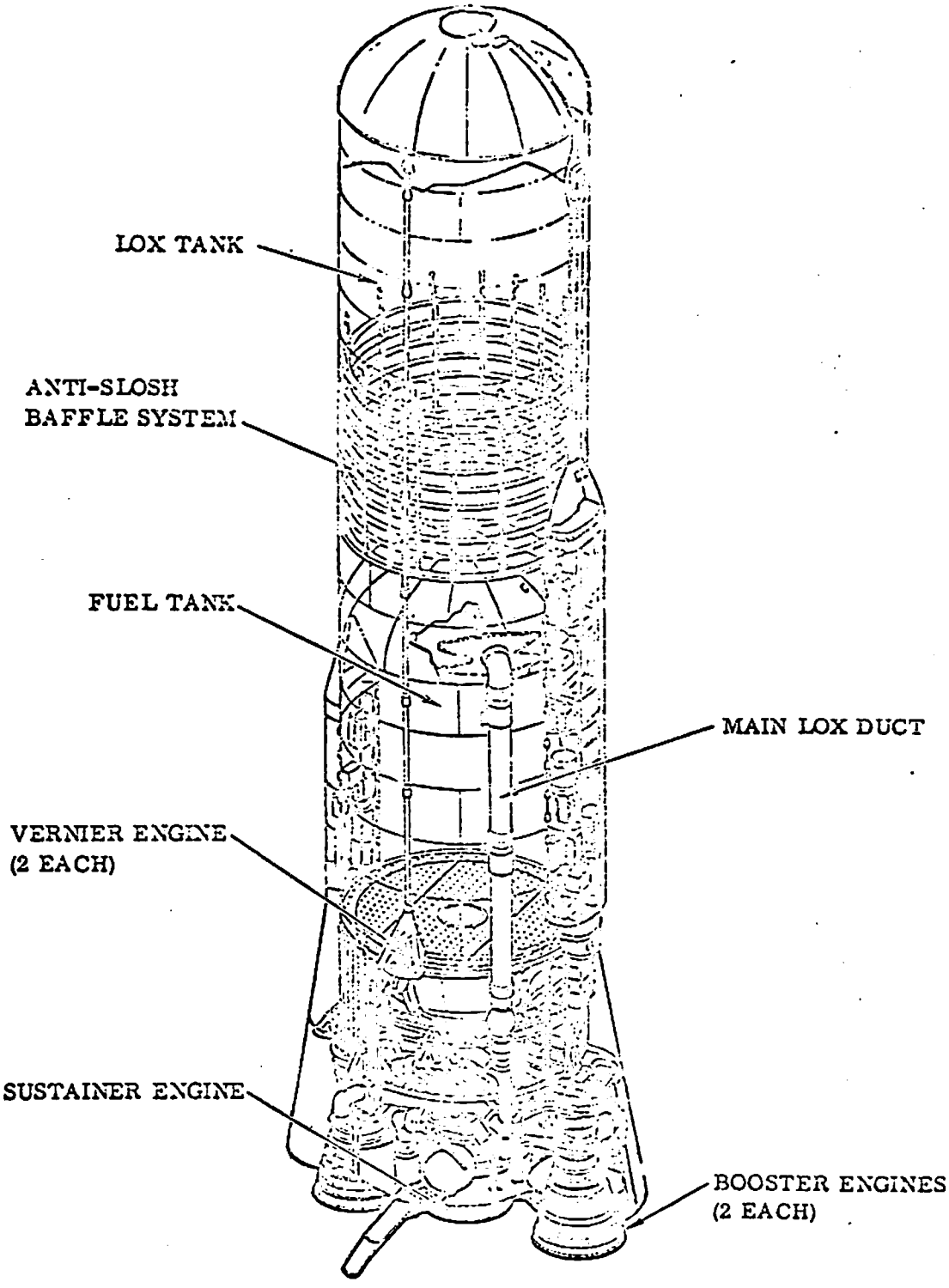
Launch azimuth	108 degrees
Injection energy (square of departure hyperbolic asymptotic velocity)	84 km^2/sec^2
Injection flight path angle	4 degrees
Injection altitude	117 n. mi.

Actual performance will vary in some regards from launch day to launch day, or even at different launch times of the same launch day.

2.1.4.1 Standard Atlas. The Atlas, as shown in Figure 2-3 is powered by two booster engines which are jettisoned during flight and a sustainer engine which burns continuously until propellant depletion. Two small vernier engines are provided for roll control during the sustainer phase of flight.

The Atlas tank structure consists of a pressurized stainless steel cylinder 10 feet in diameter which is divided by an intermediate bulkhead into separate fuel and oxidizer tanks. The jettisonable booster section which contains the two booster engines and power pack is secured to a thrust ring at the aft end of the tank assembly. Separation of the booster section from the tank is accomplished by operating pneumatic separation fittings severing the connection to the tank. Guide rails maintain booster section orientation during separation to prevent collision.

A cylindrical interstage adapter of aluminum skin and stringer construction mates the Centaur vehicle to the Atlas. Separation of the



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	ATLAS	DOC. NO.
		FIG. 2-3
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Centaur is achieved by igniting a flexible linear shaped charge located at the forward attachment of the interstage adapter. The charge severs the connection to the Centaur and eight solid propellant retro-motors located on the Atlas are fired to back the Atlas away from the Centaur.

The Atlas is controlled during the booster phase of flight by its own flight programmer. After booster engine staging, however, the Centaur guidance system assumes control of the vehicle. The airborne hydraulic system provides power for gimballing all engine thrust chambers as required by the flight control system.

2.1.4.2 Standard Centaur. The Centaur vehicle shown in Figure 2-4 has a nominal diameter of 10 feet and is 30 feet in length. It employs Liquid Hydrogen (LH_2) and Liquid Oxygen (LO_2) as propellants. The primary thrust is provided by two Pratt and Witney engines developing a total of 30,000 pounds.

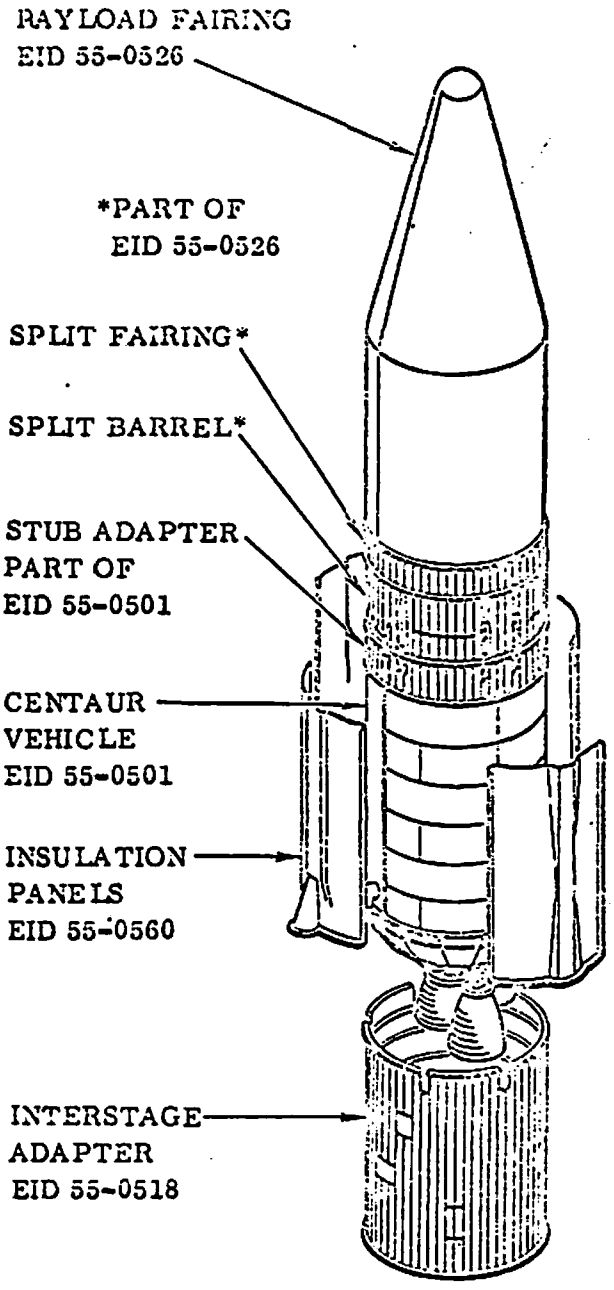
The engines may be started, shutdown, and restarted in a space environment to accomplish desired orbital changes. The Centaur is capable of coast periods up to 25 minutes between engine firings but for Pioneer F and G it will be fired in a single burn period with no restart. Vehicle orientation during coast periods or after shutdown is provided by reaction control engines powered by Hydrogen Peroxide (H_2O_2) monopropellant.

The Centaur tank structure is fabricated from stainless steel and is pressurized to support environmental loads. The propellants are separated by a double-walled intermediate bulkhead, which is insulated and evacuated. Fiberglas insulation panels protect the LH_2 from aerodynamic heating; and a Fiberglas nose fairing protects the third stage, payload and Centaur equipment from aerodynamic loads during ascent through the atmosphere. These are both jettisoned after Atlas booster engine jettison and Centaur main engine start, respectively.

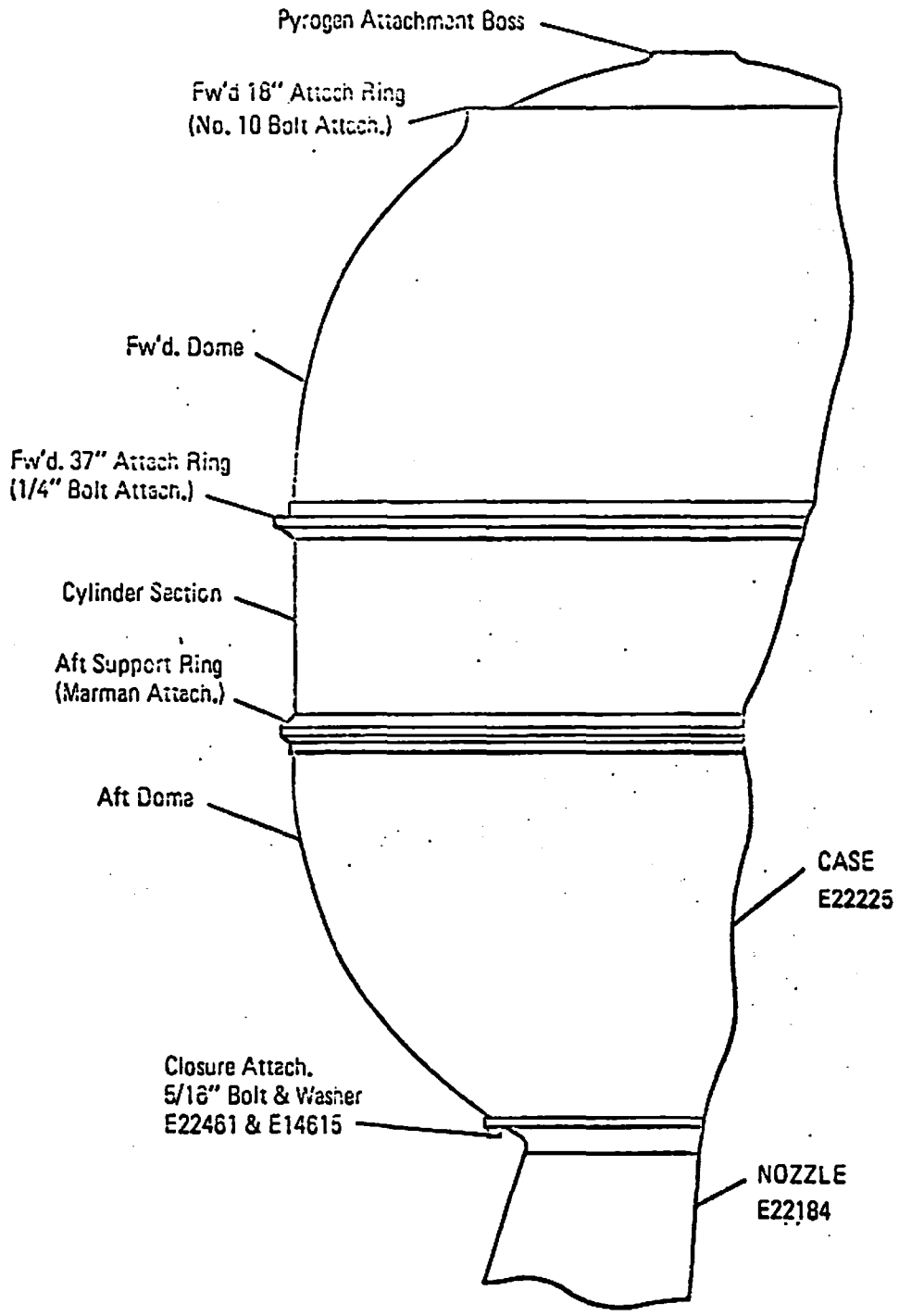
Centaur guidance is performed by a Honeywell all inertial system capable of accommodating a wide variety of steering programs for orbital, lunar, and interplanetary trajectories. During flight the guidance system determines vehicle position and velocity and initiates steering commands. The autopilot acts on these commands and keeps the vehicle oriented in the direction ordered by the guidance system. Commands from the autopilot are fed to separate hydraulic systems which gimbal each main engine. Attitude is controlled during coast periods by commands issued to the H_2O_2 control engines.

2.1.4.3 TE-364-4. The TE-364-4 motor case shown in Figure 2-5 is basically an extended spherical pressure vessel consisting of a cylinder section 15 inches in length, welded at each end to 37 inch diameter hemispherical domes. The TE-364-4 employs the solid TP-H-3062 as propellant developing a total thrust of 14,808 pounds.

The motor case is fabricated from forged 6Al-4V titanium heat treated to 165 k psi minimum ultimate tensile strength. Port openings are machined in the aft and forward hemispheres to admit nozzle and igniter assemblies,



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	TE-M-364-4 ROCKET MOTOR SKETCH	NASA AMES RESEARCH CENTER MOFFETT FIELD, CALIFORNIA
		DOC. NO
		FIG. 2-5
	REV. NO. / DATE	SHEET 1 OF 1

respectively. Interface support and attachment structures consist of three full circumferential rings; bolt attachment ring with the forward portion of the cylinder; and 37-inch diameter Marman support ring integral with the aft portion of the cylinder, allowing for attachment of the third stage spin table and two integral flanges for attachment of the spacecraft attach fittings.

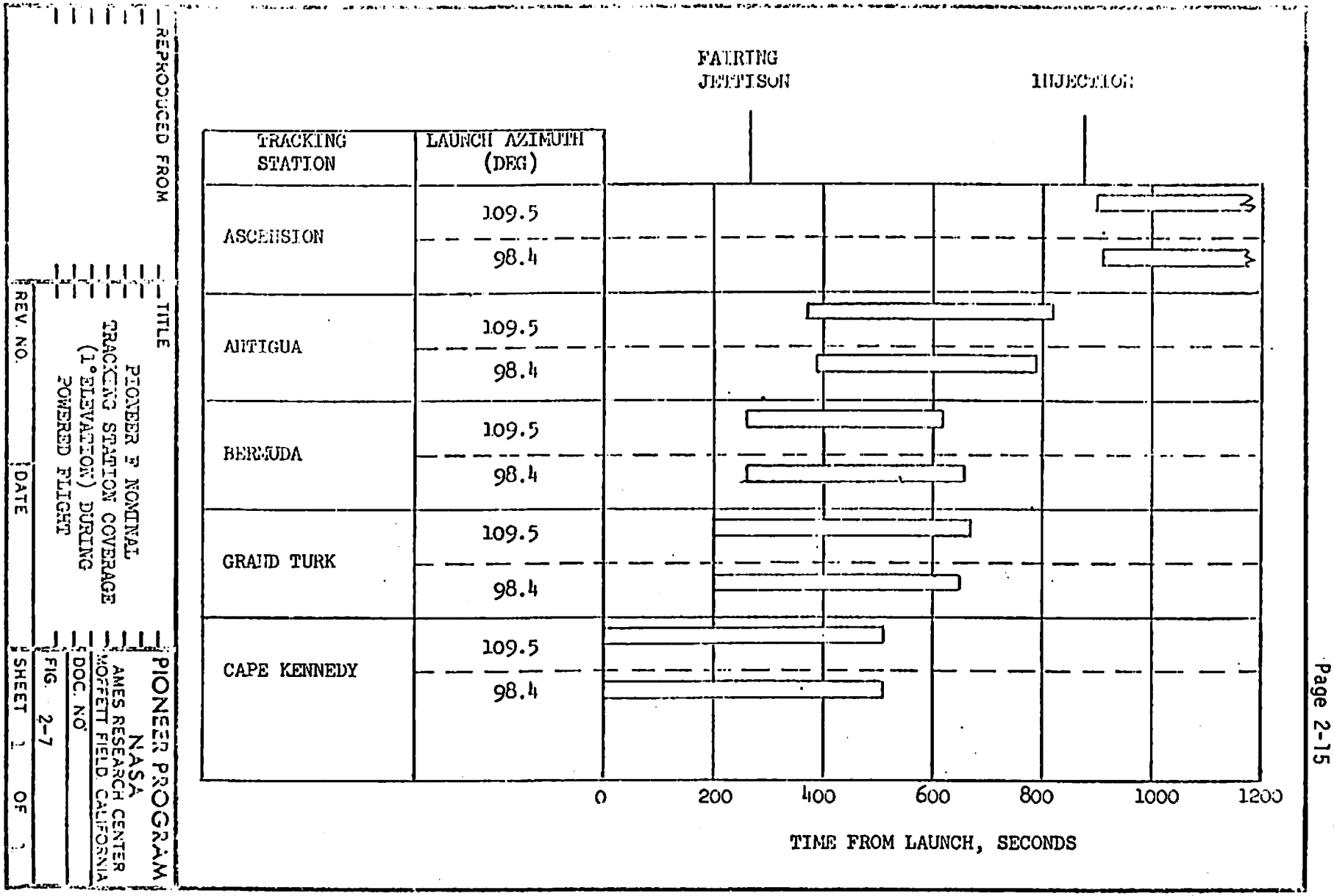
The third stage has no active guidance control system and therefore, it and the spacecraft system must be spin stabilized throughout the burn period in order to minimize guidance errors.

2.1.5 Powered Flight Sequence

Events from liftoff to injection of a typical Pioneer F powered flight sequence, Figure 2-6, will be covered by tracking stations at Cape Kennedy, Grand Turk, Bermuda, Antigua, and Ascension (see Figure 2-7). Figure 2-8 shows the typical Pioneer F earth tracks between the launch azimuths of 98.4 degrees and 109.5 degrees.

Event	Time from Liftoff (sec)
1. Liftoff	0
2. Booster Engine Cutoff (BECO)	153.4
3. Booster Jettison	156.5
4. Insulation Panel Jettison	188
5. Sustainer Engine Cutoff (SECO)	243.3
6. Sustainer Jettison	245.3
7. Centaur Main Engine Start (MES)	254.2
8. Fairing Jettison	266.8
9. Centaur Main Engine Cutoff (MECO)	705.8
10. TE-364-4 Spin Up	775.8
11. Centaur Jettison	777.8
12. TE-364-4 Ignition	790.8
13. TE-364-4 Burnout	834.7
14. TE-364-4 Spacecraft Separation	954.7

Figure 2-6. Typical Powered Flight Sequence of Events for Atlas/Centaur/TE-364-4



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PIONEER F NOMINAL TRACKING STATION COVERAGE (1° ELEVATION) DURING POWERED FLIGHT

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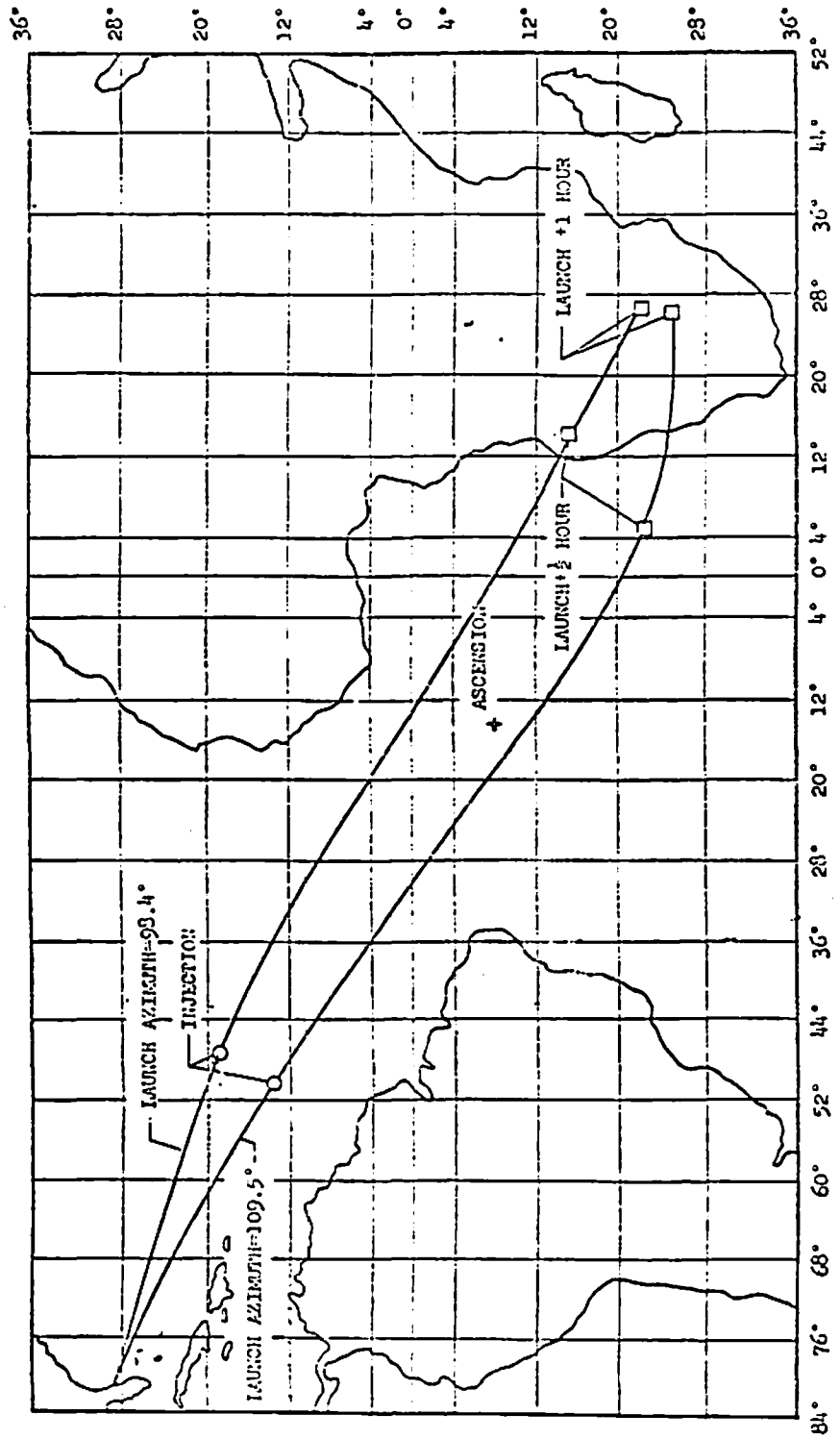
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FIG. 2-7

SHEET 1 OF 1



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TYPICAL PIONEER F
EARTH TRACKS

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FIG. 2-8

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2.1.6 Ground Data System Interface Description

The Pioneer F/G ground data system receives, processes, records, analyses and transmits command and telemetry data to and from the Pioneer Spacecraft. It is formed by the combination of the Jet Propulsion Laboratory deep space tracking stations (DSS), JPL/Space Flight Operations Facilty (SFOF), and the NASA/Ames Research Center (ARC) Pioneer Mission Analysis Area.

A block diagram of the ground data system is shown in Figure 2.9. The system operates in three modes; normal, hot backup, and manual entry. In normal mode, a single Telemetry Command Processor (TCP) and a Command Modu-Assembly (CMA) are used. The redundant Telemetry Command Processor and Command Modulation Assembly are used for the hot backup mode. The Telemetry Command Processor accepts decoded data from the sequential decoder, formats the data, writes the digital original data record (ODR), and processes and transmits command messages. The ground data system at the Space Flight Operations Facility is connected to the Deep Space Stations by a high speed data link (HSDL). The telemetry and tracking data received are processed and displayed and sent to the remote computer at ARC.

All on-line real time functions for flight exist in the IBM 360-75 system at Space Flight Operations Facility. Some of the off-line functions are in the Univac 1108, with others in the Pioneer off-line Data Processing System (POLDPS). The ARC-XDS Sigma V and the IBM 360-75 have full command and telemetry capabilities including remote control of mission dependent software. The system produces displays of alphanumerical and graphic data on TV (IPP data) or CRT displays and also produces hard copy outputs. The system also performs detailed analysis of the subsystems including power, communications, data handling, command, attitude control, propulsion, and thermal, and perform overall system analysis including navigation, and processing IPP data for display.

The characteristics of the command signals for transmission to the spacecraft are next described. The uplink signal is a frequency shift key, pulse code modulation (FSK/PCM) system having two tones of 128 Hz and 204.8 Hz to represent zero and one respectively. The command bit rate is 1 bps. A normal command is 22 bits, but may be as few as 20 bits for successive commands. A command word consists of a four bit preamble (zeros) a sync bit, a two bit decoder address, a three bit routing address, an eight bit command message, and four parity bits. The preamble may be only two zero bits for successive back to back commands in a low noise environment. Detailed characteristics of the command subsystem are presented in Section 3.9.

The telemetry data may be either coded or uncoded as selected by ground command. In the coded mode, the encoder replaces the data bits with two parity bits based on the values of the bits previously generated. Coded data can be reconstructed to the original data without sequential decoding. If the encoders fail, uncoded (NRZ-L) data may be sent. The data formats for Pioneer F/G are 192 bits in length and include Formats A, B, C, C1, C2, C3, C4, D1, D2, and D3. Formats A and B are scientific formats and differ only in the format I.D. and the assignment of experiment

data within the frame. Fixed words are included in all formats except D1 through D3 which have no fixed words. D format frames are for special scientific data, and each follows a frame of Format A or B. Format C is the Engineering format which consists of cycling through formats C1 to C4 although each individual format may be transmitted independently. Telemetry operation can occur in any of three selected modes, real time, TLM store, and memory readout. Detailed characteristics of the various formats are given in Section 3.5.

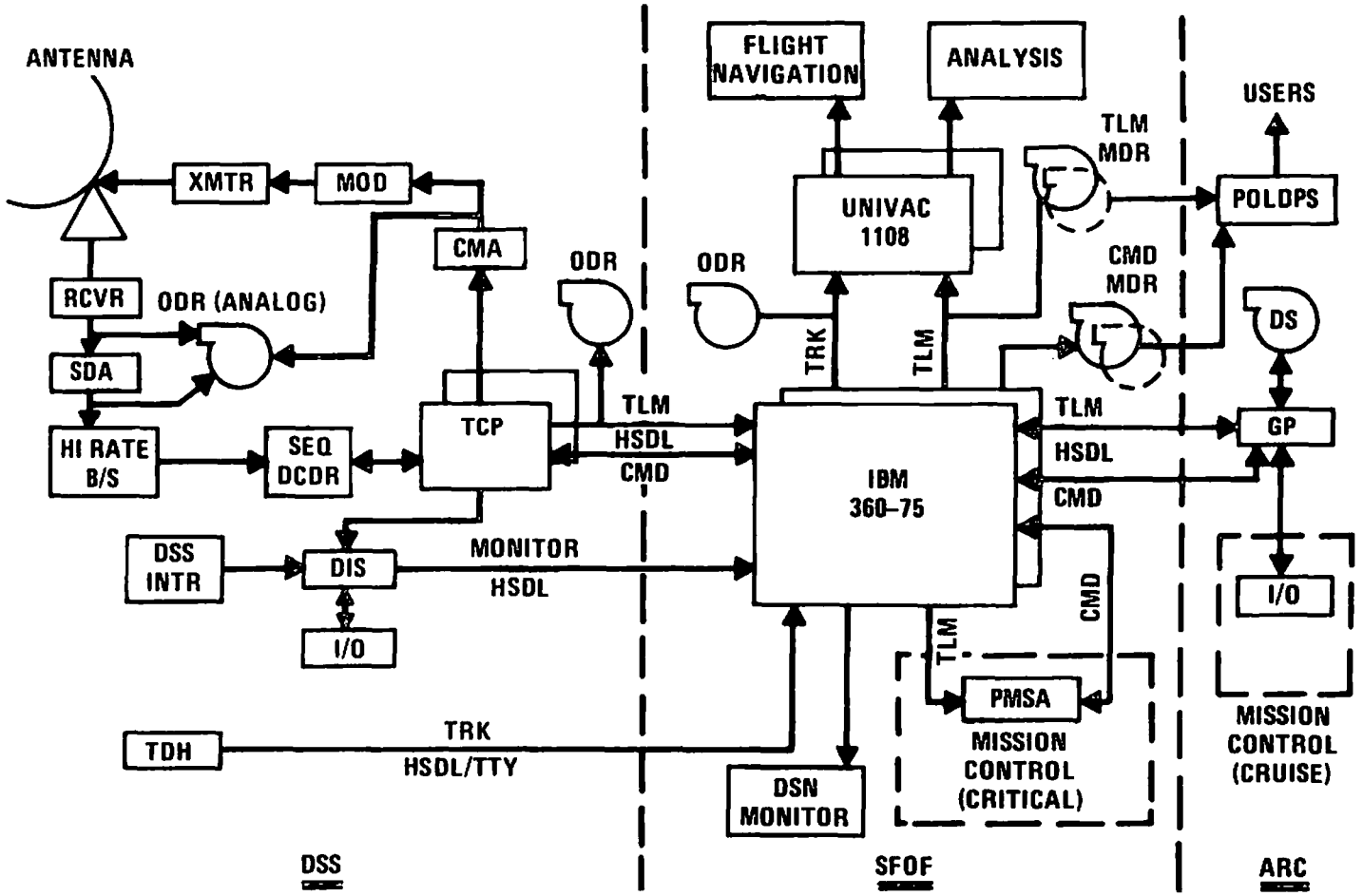


Figure 2-9. Pioneer Ground Data System

2.2 INTERFACES

The spacecraft interfaces can be classified as external and internal. External interfaces are those with the launch vehicle and communications interfaces while internal include the interfaces with the scientific instruments, and with the subsystems of the spacecraft. In this section, the operationally significant interfaces will be defined and briefly summarized.

2.2.1 Launch Vehicle Interface

The primary launch vehicle interfaces are mechanical and electrical and are defined in Pioneer Document PC-221.

The mechanical interface consists of the 25 in x 12 in conical, one piece aluminum interstage structure which in turn is mated to the launch vehicle third stage. The Centaur coordinate system has its -y axis in the downrange direction with the positive z axis pointing aft. The spacecraft/TE-364-4 is mounted with the spacecraft +x axis aligned in the down-range direction with the negative z axis aft.

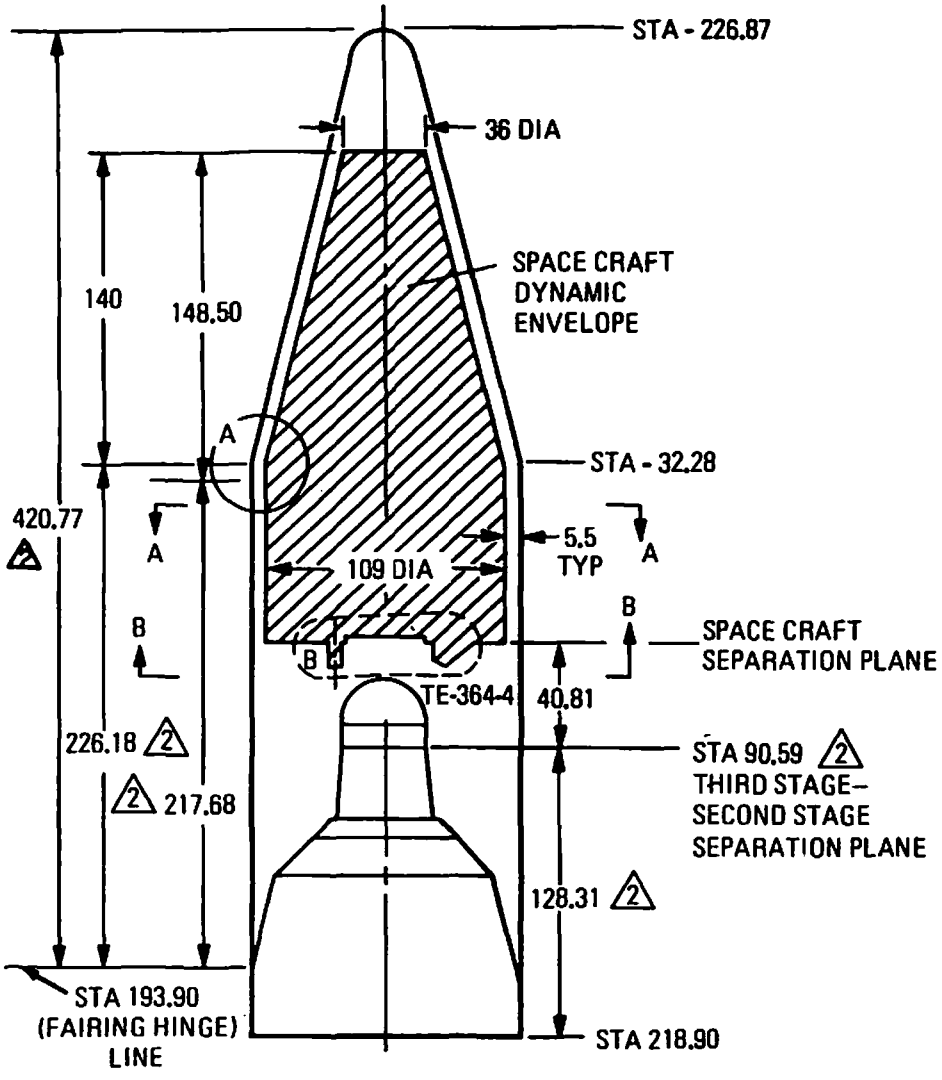
The fairing is a 120 inch diameter cylindrical afterbody 226 inches long with a 194 inch conical forebody as shown in Figure 2-10. This Figure also shows the spacecraft dynamic envelope and the separation planes between the second and third stages and between the third stage and the spacecraft. There are two 30 inch diameter access doors in the fairing; one to accommodate manned access, the other for RTG installation. At about 267 seconds after liftoff, the fairing is jettisoned. After Centaur main engine cutoff the balanced spacecraft/TE-364-4 is spun up to 60 rpm ± 6 rpm, the Centaur is jettisoned and the TE-364-4 is ignited for injection.

The electrical interface is primarily an RF interface. No umbilical wiring or spacecraft power is provided on the launch vehicles. The spacecraft communication RF signal is at 2295 $\pm$ 5 MHz, 8 watts maximum divided between the medium gain and omni antennas. The signal is phase modulated at 32,768 Hz. The receivers are sensitive to radiation levels in excess of -55 dbm in a frequency range of 2110 $\pm$ 5 MHz. A complete description of the communication subsystem, including its capabilities and limitations, is given in Section 3.6. The communications interfaces are defined in PC-222.

2.2.2 Interfaces with Scientific Instruments

Eleven scientific instruments have been selected for inclusion in the Pioneer F and G Spacecraft. They are as follows:

	Type	Experimenter
1.	JPL Helium Vector Magnetometer	Smith
2.	ARC Plasma Analyzer	Wolfe
3.	U/Chicago Charged Particle Instrument	Simpson
4.	U/Iowa Geiger Tube Telescope	Van Allen
5.	GSFC Cosmic Ray Telescope	McDonald



NOTE: ALL STATIONS ARE CENTAUR STATIONS

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		DOC NO
		FIG 2-10
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	Type	Experimenter
6.	UCSD Trapped Radiation Detector	Fillius
7.	UCS Ultraviolet Photometer	Judge
8.	U/Arizona Imaging Photopolarimeter	Gehrels
9.	CIT Jovian Infrared Radiometer	Münch
10.	GE Asteroid/Meteoroid Detector	Soberman
11.	LaRC Meteoroid Detector	Kinard

The locations of these instruments are shown in Figures 2-2, 3.1-1, 3.1-2, and 3.1-3. In addition to the experimenters above, two experimenters have been designated for investigations which do not involve onboard instrumentation other than the normal communications system. These are:

Type	Experimenter
S-Band Occultation	Kliore
Celestial Mechanics	Anderson

The collection of instruments carried weighs about 60 lb. Each instrument receivers power through an individual fused branch circuit. The total power used by the scientific instruments is about 24 watts nominal with a peak load of about 34 watts. In the event of an undervoltage condition on the spacecraft, the scientific instrument group is the first to be turned off.

In addition to the power interface, the instruments are provided with a roll index pulse attitude reference. The roll index pulse accuracy is within 0.16° if the Stellar Reference Assembly is used as the reference signal. The Star Referenced pulse can be delayed to any orientation (such as the ecliptic plane) with a quantization of 0.112°. With the sun sensor as roll reference, the accuracy is nominally about 0.3° but can be as great as 1° for the spin axis pointing within 2-3 degrees from the sun. The sun sensor detects the plane containing the spin axis and the sun, and cannot be delayed or controlled.

The telemetry formats A, B, D1, D2, D3, E1, and E2 are all primarily for scientific instrument data. The D and E formats are special formats as follows.

- D-1. 192 bits, U/Arizona Imaging Photopolarimeter
- D-2. Bits 1-24, CIT Infrared Radiometer
  Bits 25-192, U/Arizona IPP
- D-3. ARC Plasma Analyzer

- E-1, E-2. These special formats are for subcommutation of scientific information in two words of each format A and B or one words of each format C1, C2, C3, and C4 when telemetered as the main frame.

The data handling subsystem has the capability of storing up to 49,152 bits of data in the data storage unit (DSU). Analog signals from the scientific instruments are converted to 6 bit digital words and can be telemetered in real time, can be stored in the DSU, and telemetered simultaneously, or can be read out from previous storage. Storage and memory readout bit rates are selectable by ground command.

A complete definition of the interfaces between the spacecraft and the scientific instruments can be found in PC-220.

2.2.3 Interactions Between Subsystems

The subsystems comprising the primary spacecraft operational characteristics are as follows:

1) Mechanical/Structural Subsystem

2) Thermal Control Subsystem

3) Attitude Control Subsystem

4) Propulsion Subsystem

5) Data Handling Subsystem

6) Communications Subsystem

7) Power Control Subsystem

8) RTG/RHU

9) Electrical Distribution Subsystem

The interfaces between the subsystems are classified as mechanical, thermal, and electrical. The operationally significant interfaces are shown in the system block diagram, Figure 2-11.

The Mechanical/Structural Subsystem provides mechanical support for the instruments and other subsystems and includes deployment mechanisms, deployment and wobble dampers and the feed movement mechanism.

The Thermal Control Subsystem is also without commands since all thermal control is passive via the aft mounted louver system, except for off-on control by thermal power dissipation of the various subsystems and scientific instruments. Thermal control performance is indicated by six temperature measurements from various locations in the equipment compartment.

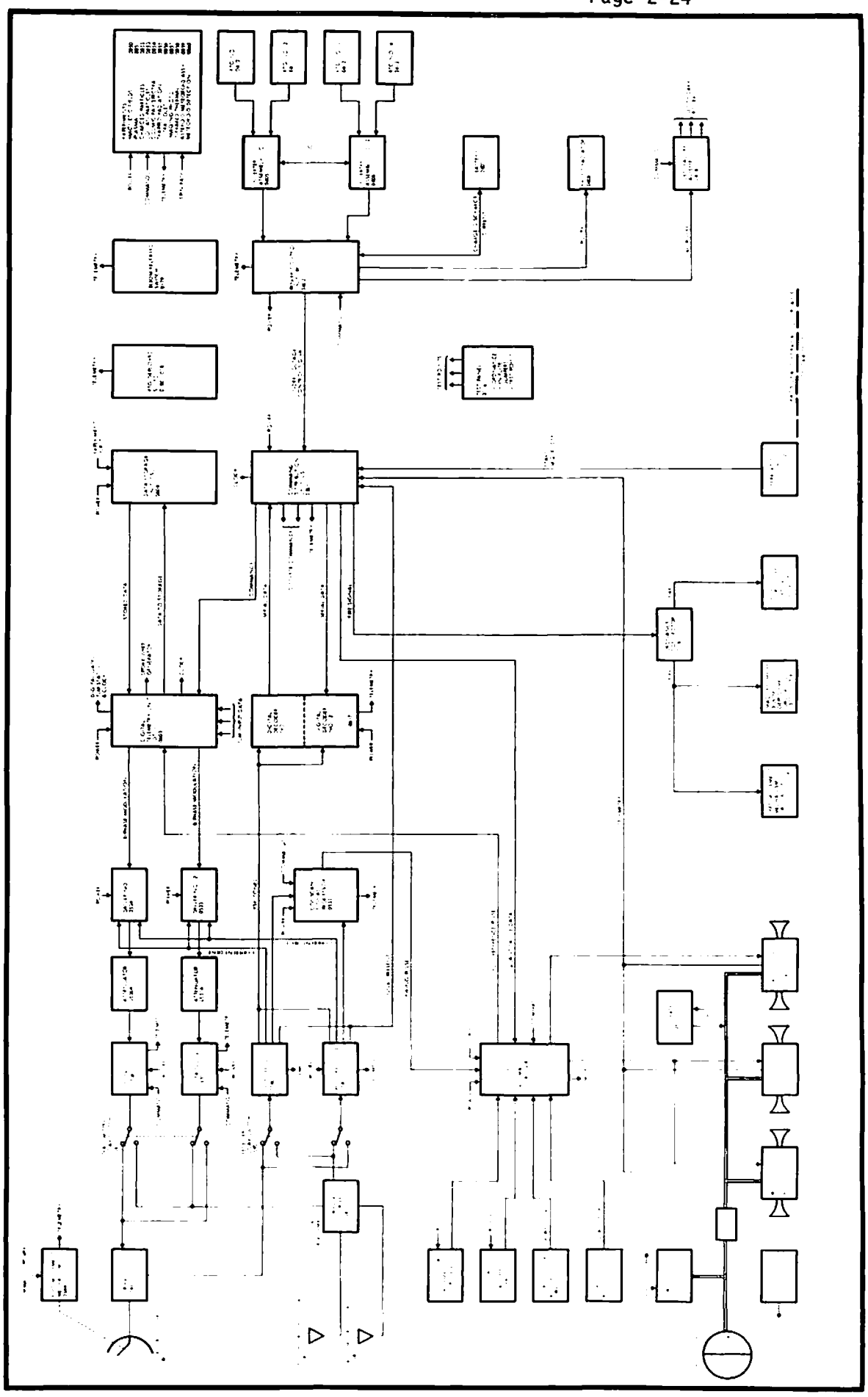


FIGURE 2-11

PIONEER F/G SPACECRAFT

SYSTEM BLOCK DIAGRAM

Power to the spacecraft is normally controlled by shunt regulation of the RTG output. The RTGs are deployed at about 120 inches from the spacecraft c.g. to minimize thermal, magnetic, and nuclear radiation interface interactions. The RTG electrical interface consists of the power input of 156 watts to the spacecraft, and the telemetry indication of RTG temperature. The RTG load and the control of power for all units in the spacecraft is provided by the Power Control Subsystem.

The remaining subsystems, ACS, propulsion, data handling, communications, and electrical (command) distribution have numerous interfaces, primarily electrical. These interfaces are defined in the interface matrix, Figure 2-12. The numbers in this matrix refer to the following electrical interface items.

1) Power = nominal cruise power

2) Commands = number of commands

3) Telemetry = number of telemetered quantities

4) Data

5) Timing signals

6) Real time clock frequency reference

7) Roll reference signal

8) RF

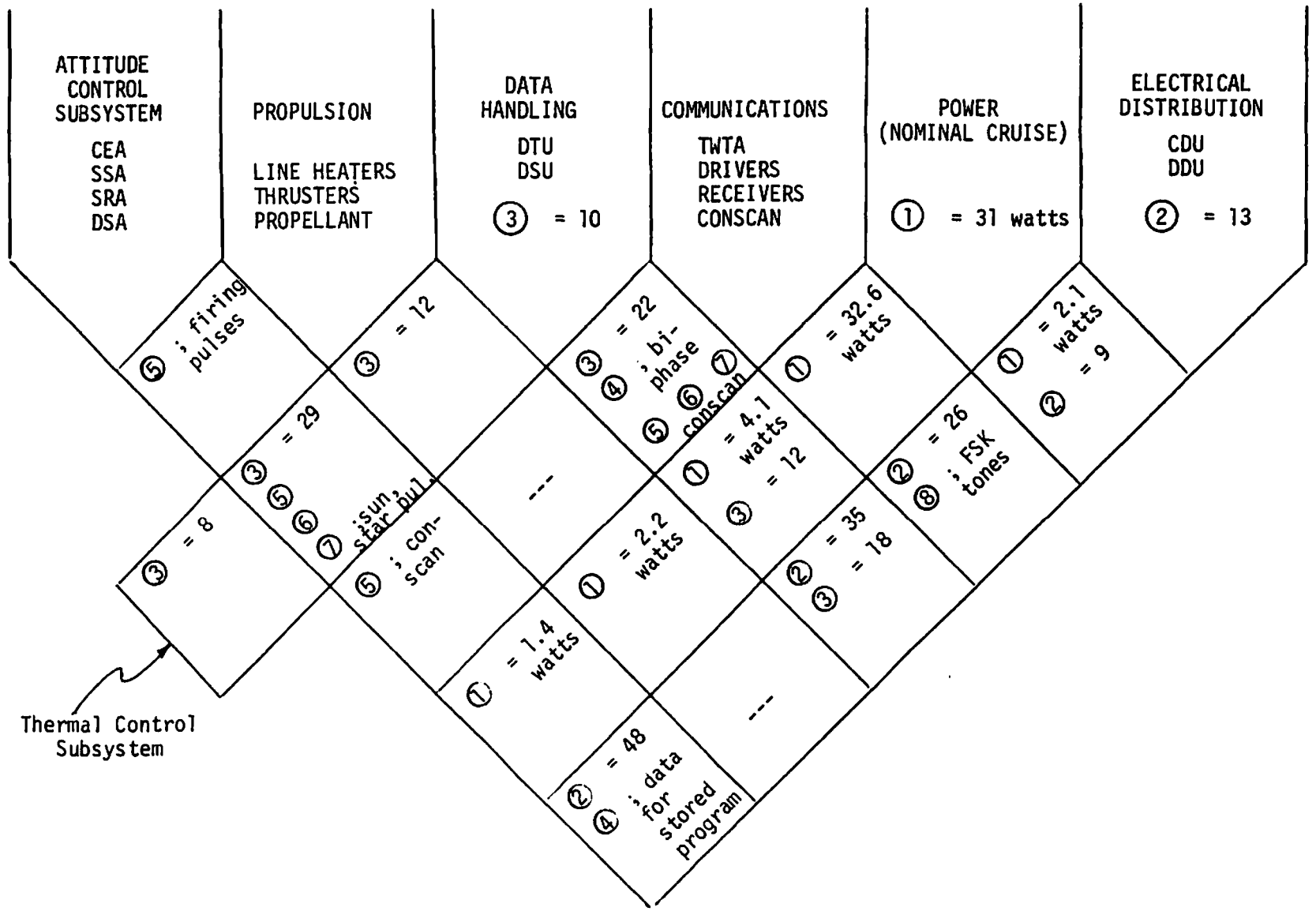


Figure 2-12. Subsystem Interaction Matrix

2.3 SPACECRAFT CAPABILITIES AND LIMITATIONS

In this section, bare summaries are presented of the spacecraft capabilities and limitations. Only the operationally significant features are considered in order to give a general overview of the material presented in Section 3 where detailed discussions of the capabilities and limitations of each subsystem are presented. The subsystems of the spacecraft are:

Subsystem	Section
Mechanical/Structural	3.1
Thermal	3.2
Attitude Control	3.3
Propulsion	3.4
Data Handling	3.5
Communications	3.6
Power	3.7
RTG/RHU	3.8
Electrical Distribution	3.9

In the five subsections to follow, the summaries of the communication link performance, the propulsion maneuver allocations, overall power capability, and data and command capabilities are presented. The operating modes with these capabilities are presented in Figure 2-13, a matrix of subsystem versus normal mode operation for the primary mission phases.

2.3.1 Communication Link Performance

The communications subsystem provides a two way RF link with the spacecraft using three antennas, an omni-directional low-gain antenna, a medium gain horn antenna, or the 9 foot diameter high gain dish. These antenna radiate a non-coherent RF signal without an uplink signal present to permit acquisition by the deep space network.

The system receives from the ground and demodulates commands and data in a PCM/FSK/PM format at a frequency of about 2110 MHz. The system transmits using a PCM/PSK/PM format at a frequency of about 2292 MHz using a single subcarrier at 32 KHz. A phase coherent mode available for doppler measurements has a frequency translation ratio of 240/221. Downlink transmission is via the Travelling Wave Tube Amplifiers which are interchangeable by command between the high gain antenna and the medium gain-omni combination. On the medium gain-omni combination, there is equal power division of the 8 watts maximum output of the TWTs.

The antenna systems have the following characteristics.

High Gain Antenna

Effective radiated power (ERP)	70 dbm
Beamwidth	$\approx$3.5° up 3.3° down
Polorization	RCP

Aligned to +z, can be offset 0.8°

SUBSYSTEM	MODE				
	INITIAL FLIGHT SEQUENCE	MIDCOURSE CORRECTION	CRUISE	ENCOUNTER	POST ENCOUNTER
Mechanical	1. Deployment Ordinance and Mechanisms RTG Magnetometer 2. Wobble Damping	1. Feed Movement for Conscan 2. Wobble Damping	1. Feed Movement for Conscan 2. Wobble Damping	Same as For Cruise	Same
Attitude Control and Propulsion	1. Despin 2. Earth Acquisition Precession Maneuver Calibration of Thrusters 3. Roll Reference Signals 4. Closed Loop Conscan	1. Stored Program for Precession ΔV Return Precession 2. Control for Conscan 3. Spin Control 4. Roll Reference Signals	1. Star Roll Reference 2. Conscan Control for Earth Tracking 3. Spin Control 4. ΔV/SCT	Same as For Cruise	Same as For Cruise
Data Handling	1. Telemetry--BR 2048 Formats C-Engineering accelerated in main frame 2. Timing Signals 3. Roll Index Pulse	1. Telemetry C-4 Engineering accelerated in main frame 2. Timing Signals 3. Roll Index Pulse	1. Telemetry Formats A & B D-Special Science E-Science Subframe	1. Telemetry Store 2. Memory Readout 3. Telemetry D-Special Science Format E-Science Subframe A & B 4. Timing Signals 5. Roll Index Pulse	Same as For Cruise and Encounter
Communications	1. Downlink Only Initially, PCM/PSK/PM format uplink, when established is PCM/FSK/PM format	1. Omni and Medium Gain, Uplink and Downlink Communication 2. Coherent Mode Doppler Measurement 3. Conscan	1. Omni, Medium Gain, and High Gain Antenna Communication--Bit Rate Depends on Signal Strength 2. Coherent Mode Doppler Measurement 3. Conscan on Medium Gain to 3 AU	1. High Gain Antenna Communications, 210' Dish Gives 2048 Bit Rate 2. Coherent Mode Doppler 3. Conscan 4. S-Band Occultation	1. High Gain Antenna Communication, Bit Rate Drops with Signal Strength 2. Coherent Mode Doppler 3. Conscan
Command and Electrical Distribution	1. Sequencer Control • Initiate Despin • Thrust Verification • Initiate RTG Deploy • Initiate Magnitude Deploy (fires ordinance) • Start Earth Acquisition Maneuver 2. Real Time Command Processing 3. Command Memory 4. Thruster Firing Count 5. TLM Conditioning 6. Overvoltage Sensing	1. Real Time Command Processing 2. Command Memory 3. Thruster Firing Count 4. TLM Conditioning 5. Overvoltage Sensing	1. Real Time Command Processing 2. Command Memory 3. TLM Conditioning 4. Overvoltage Sensing 5. Signal Present Detection for Receiver Switch: One Receiver on Omni-Medium Gain, Other on High Gain Antenna	Same as For Cruise	Same as For Cruise
Power	1. Supply Power 2. Regulate Voltage AC 28 volts DC CTRF voltages 3. Undervoltage Provisions	Same As Initial Flight Sequence	Same	Same	Same, RTG Degradation Determines Life of Spacecraft

FIGURE 2-15. MATRIX OF SUBSYSTEMS VS NORMAL OPERATING MODES

Medium Gain Horn
- ERP: 47 dbm
- Beamwidth: ≈32° up, 29° down
- Polarization: RCP
- Aligned 9.3° from +z axis for conscan

Omni-Low Gain Antenna
- ERP: 35 dbm
- Beamwidth: 120°
- Looks in -z direction

The bit rate capabilities of the system range from 16 to 2048 bps and depend on communication distances and the signal strength.

The communications system contains the conical scan equipment used to obtain earth tracking. With the medium gain horn, the spacecraft can be aligned to the earth line within 1.3° out to about 3 AU from the earth. The high gain antenna conscan will align to within about 0.3°.

During operation, each antenna always has one receiver connected to it. To protect against receiver failures where the failed receiver is connected to the only antenna system through which the spacecraft can be commanded, an automatic receiver reverse is incorporated into the system. If 36 hours pass without a signal presence indication from either receiver, the automatic reverse interchanges the receiver connections.

2.3.2 Propulsion Maneuver Allocations

The spacecraft propellant tank is expected to contain 59.9 lb of hydrazine monopropellant and 0.9 lb of GN_2 pressurant at launch. A nominal allocation of the hydrazine to the various mission phases is presented in Table 2-1. Gross trades for the allocations are also shown.

Table 2-1. Nominal Propulsion Maneuver Allocations

Maneuver	Nominal Allocation (lb_m)	Assumed I_{sp} (sec)	Present Capability	Gross Trade (performance/lb_m)
Despin	0.64	215	59 rpm	uncontrollable
Spin Control	1.10	140	14 rpm	12.8 rpm
Precession	8.08	140	1250°	155°
ΔV	49.08	215	200 m/sec	4.1 m/sec
TOTAL	58.90			

In addition to that shown in the table, the tank will contain allocations for leakage of 0.2 lb_m-N_2H_4, 0.6 lbs for calibration, and an unusable residual of 0.8 lb_m N_2H_4. The total contents of the tank will be about 61.4 lb_m launch.

The capability shown is based on a nominal spacecraft mass pf 550 lb including a full tank. Depending on the performance characteristics of the thrusters and on various propulsion system parameters, the I_{sp} actually achieved may differ from that given above. The tradeoff can, however, be expected to be approximately 1 m/sec ΔV = 3.2 rpm spin control = 38.0° precession, or 1 rpm spin control = 12° precession = 0.31 m/sec ΔV.

The effects that will be achieved for a given maneuver depend strongly on tank pressure and the operational mode, pulsed or steady state. The extremes in performance are shown in Tables 2-2 and 2-3.

Table 2-2. Range of Propulsion Performance

Characteristic	Maximum	Minimum
Tank Pressure	550 psia	140 psia
Thrust	1.23 lb_f	0.49 lb_f
Specific Impulse		
Steady State Firing	218.4 sec	213.4 sec
Pulsed Firing	160 sec	130 sec

Table 2-3. Range of Performance Characteristics

Mode		Full Tank	Empty Tank
ΔV, Steady State Propellant mass,	$\frac{lb_m}{m/sec}$	0.258	0.236
Firing time,	$\frac{lb_m}{m/sec}$	22.88	51.3
Despin Propellant mass,	$\frac{lb_m}{rpm}$	0.011	--
Firing time,	$\frac{sec}{rpm}$	1.95	--

(continued)

Table 2-3. Range of Performance Characteristics (continued)

Mode	Full Tank	Empty Tank
Spin Control, rpm/pulse		
Pulse Width: 2 sec	0.212	1.90
1 sec	0.106	0.95
1/2 sec	0.052	0.45
1/8 sec	0.012	0.10
1/32 sec	0.0025	0.02
Precession, degrees/pulse pair		
Pulse Width: 2 sec	4.87°	1.90°
1 sec	2.52°	0.95°
1/2 sec	1.24°	0.45°
1/8 sec	0.29°	0.10°
1/32 sec	0.06°	0.02°

These results were obtained from preliminary test data and indicate the wide range of performance variations that must be characterized. The characterization is on an individual thruster, with tank pressure and the influence coefficients for usage, propellant supply temperature, valve supply voltage, and other factors taken into account. Further discussion of the propulsion subsystem performance is in Section 3.4.

2.3.3 Power Subsystem Capability

The power subsystem has three primary functions; control and regulate the RTG power output, support the spacecraft load, and perform battery load sharing. The regulation used is shunt regulation which controls the 28 VDC bus to within $\pm$ 1%, although long term drift and component aging can increase this to 2%. The system also supplies power at 5.3V, 12V, and 16 VDC to the subsystems as required from the transformer-rectifier assembly.

At launch, the 4 RTGs supply about 39.2 watts each to the spacecraft. This will degrade to about 35 watts each at encounter. The loss of one RTG will therefore not reduce power availability below that required in the commanding mode, about 105 watts.

The battery consists of 9 cells of 5 ampere hours capacity each. The battery would perform load sharing at times of peak requirement when the power availability has degraded.

2.3.4 Data Capability

The data handling subsystem accepts digital and analog data from the spacecraft, arranges these data into formats and generates a coded digital data bit stream which modulates the subcarrier. The bit rates are 16, 32, 64, 128, 256, 512, 1024, and 2048 bits per second depending on communication system capability. The basic science formats are "A" and "B," (192 bits) with three special "D" formats which can be alternated with A or B. The five engineering "C" formats are accelerated into the mainframe as selected, each 192 bits.

The data system is capable of storing 49,152 bits of scientific or engineering data for delayed transmission to earth. The memory system is also used to buffer high rate instrument data (16 K BPS) into the prevailing telemetry bit rate.

The data subsystem also supplies timing and control signals to the spacecraft instruments and subsystems. These timing signals include real time frequencies and a roll index pulse which can be used to identify the ecliptic plane to within 0.16 degrees.

A detailed description of all data system capabilities and limitations is presented in Section 3.5.

2.3.5 Command Capability

The command system block diagram is shown in Figure 2-14. The heart of the system is the Command Distribution Unit which accepts serial data at one bit per second. A command word consists of 22 bits composed of a 4 bit preamble, a sync bit, a 2 bit decoder address, 3 routing bits, an 8 bit command message, and 4 parity bits. The preamble can be reduced to 2 bits for back to back (contiguous) commands.

Commands can be stored two ways, in the stored command programmer, or in the control electronics assembly for certain attitude control maneuvers. The stored commands in the CEA result in a sequence of events and constitute state augmentation.

The Electrical Distribution Subsystem, of which the command distribution unit is part, also contains the capability for controlling the initial post launch sequence and provides ordinance firing, despin verification, receiver signal present detection, overload control, and thruster pulse counting. A complete description of this subsystem is given in Section 3.9.

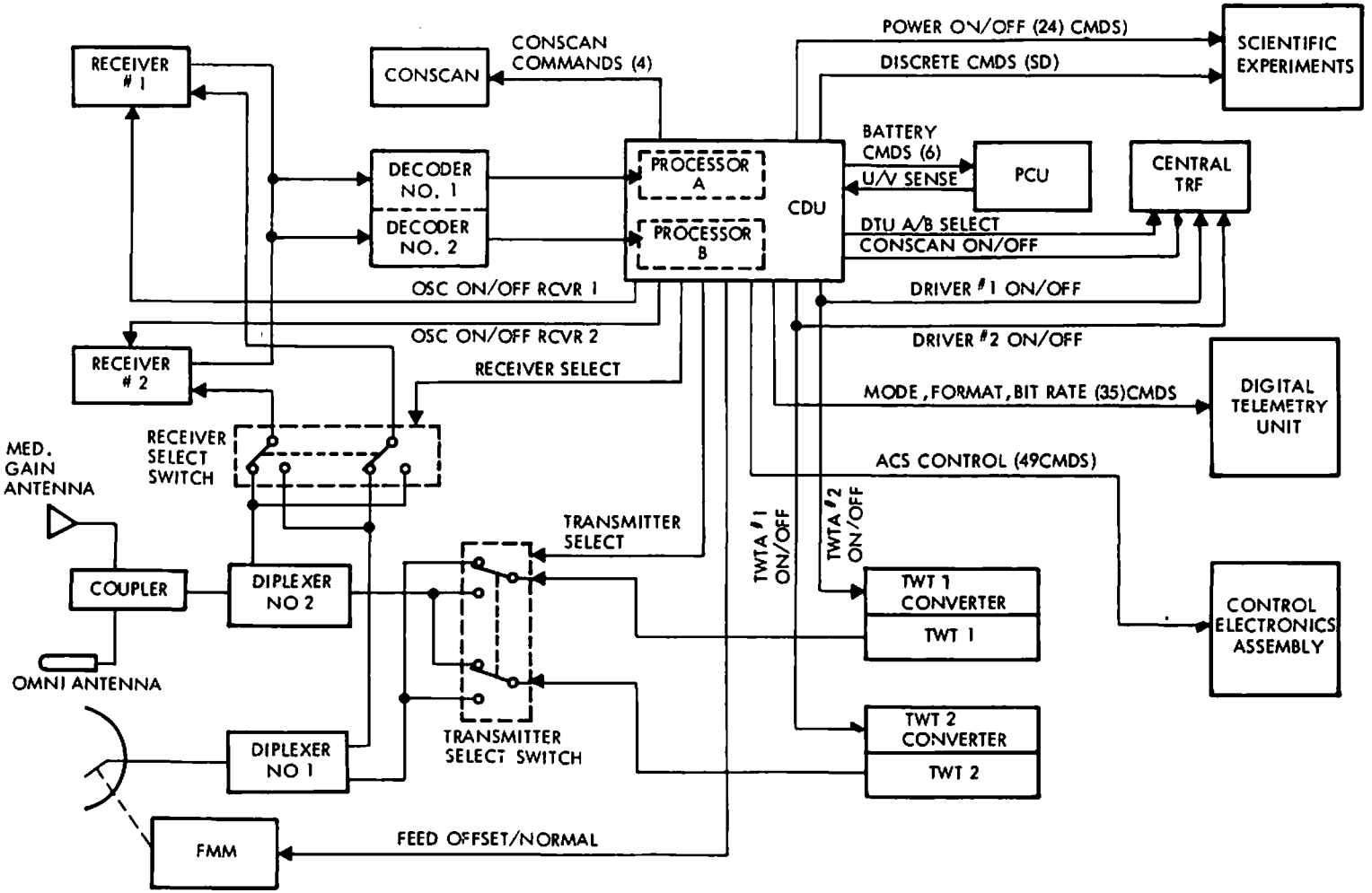


FIGURE 2-14. COMMAND SYSTEM BLOCK DIAGRAM

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3.1 MECHANICAL DESIGN

This section presents spacecraft mechanical design characteristics including the physical arrangement of equipment, functional characteristics of mechanisms and a summary of pertinent mass properties data.

3.1.1 Design Configuration

The spacecraft equipment, for discussion purposes, is divided into internal and external classifications denoting items inside and outside of the equipment compartment. Exceptions to this classification are items attached to the exterior of the compartment vertical panels.

3.1.1.1 Internal Configuration. Figure 3.1-1 presents the equipment housed within the compartment and items attached to the compartment vertical panels. The equipment list in the figure identifies each component and includes the reference designator number for the component. The reference designator number is used to correlate the component with the telemetry and command "word" assignments.

The compartment is divided into two sections; an equipment section, which is the hexagonal section, and an experiments section which is the bay on the +X side. The sections are separated by a partial vertical panel which is also used to mount some of the communications equipment and the meteroid (Kinard) detector electronics.

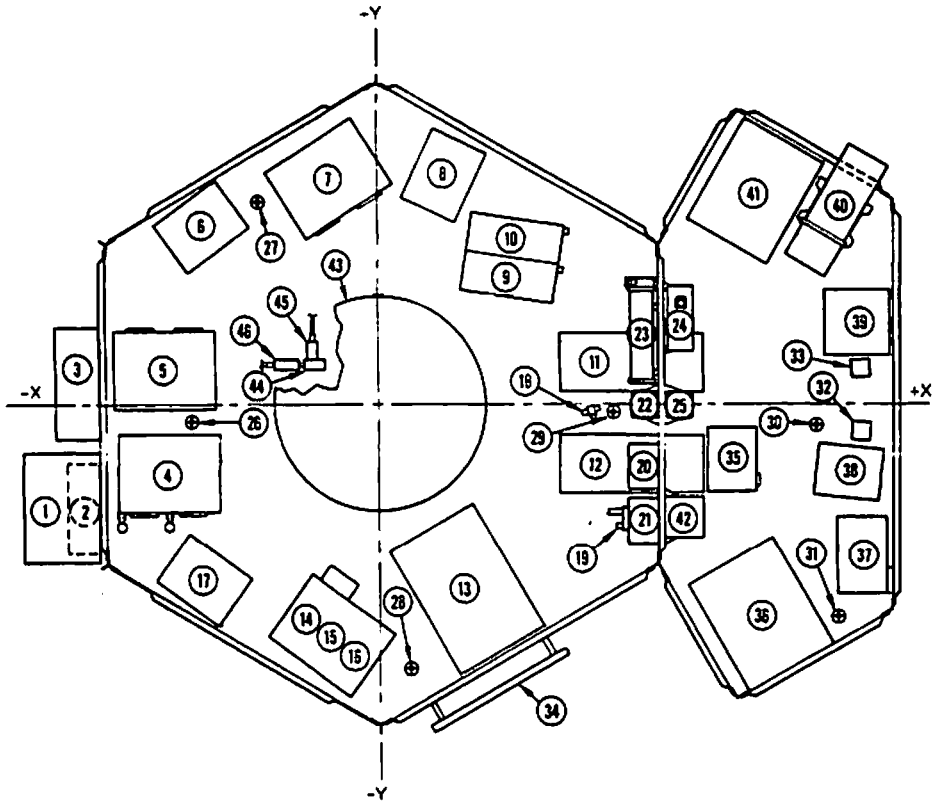
The equipment section components include subsystem electronics plus the propellant supply assembly. Items mounted on the exterior of the compartment vertical panels include the shunt radiator, battery, data storage unit and the asteroid/meteroid experiment (Soberman) electronics assembly. There are six (6) thermistors located on floor of the compartment, four in the equipment section and two in the experiments section.

Six of the eleven scientific instruments are housed in the experiment section. The five external instruments are the magnetometer, the plasma analyzer, the meteroid detectors, the cosmic-ray telescope and the asteroid/meteroid detector. The location of these experiments is shown on the spacecraft external views.

3.1.1.2 External Configuration. Spacecraft external views looking inboard, aft and forward are presented in Figures 3.1-2, 3.1-3, and 3.1-4 respectively. The inboard view gives the relative fore and aft position of components, notes the spacecraft nominal Z axis center of gravity and gives the moment arms of the thrusters. The magnetometer and RTGs are deployed in a plane that contains the spacecraft center of gravity and is parallel to the XY plane to prevent dynamic inbalances in the deployed condition.

The twelve (12) meteroid detector (Kinard) panels are attached to the aft side of the antenna reflector but are omitted on the inboard view for clarity. The detectors are shown in Figure 3.1-4.

The plasma analyzer (Wolfe) instrument and the cosmic-ray (McDonald) telescope are located on the top (+Z) surface of the experiment section



INTERNAL EQUIPMENT ARRANGEMENT

Figure 3.1-1

EQUIPMENT IDENTIFICATION

ITEM NO.	TITLE	REFERENCE SUBSYSTEM	REFERENCE* DESIGNATOR
1	Data Storage Unit (DSU)	Data Handling	0604
2	Asteroid/Meteoroid Detector Electronics (GE/Soberman)	Instruments	0859
3	Battery	Electrical Power	0407
4	Power Control Unit (PCU)	Electrical Power	0402
5	Central TRF Unit	Electrical Power	0414
6	Inverter Assembly No. 2	Electrical Power	0406
7	Command Distribution Unit (CDU)	Electrical Distributor	0301
8	Stellar Reference Assembly (SRA)	Attitude Control	0231
9	Receiver No. 1	Communications	0540
10	Receiver No. 2	Communications	0541
11	TWTA No. 1	Communications	0536
12	TWTA No. 2	Communications	0537
13	Digital Telemetry Unit (DTU)	Data Handling	0603
14	Control Electronics Assembly (CEA)	Attitude Control	0230
15	Conscan Signal Processor	Communications	0533
16	Digital Decoder Unit	Data Handling	0617
17	Inverter Assembly No. 1	Electrical Power	0405
18	Attenuator TWT No. 1	Communications	0536A
19	Attenuator TWT No. 2	Communications	0537A
20	Transmitter Driver No. 1	Communications	0534
21	Transmitter Driver No. 2	Communications	0535
22	Transfer Switch — Receive	Antenna	0590
23	Diplexer No. 2/Coupler	Antenna	0543
24	Diplexer No. 1	Antenna	0542
25	Transfer Switch — Transmit	Antenna	0589
26	Thermistor No. 1	Thermal	0782
27	Thermistor No. 2	Thermal	0783
28	Thermistor No. 3	Thermal	0784
29	Thermistor No. 4	Thermal	0785
30	Thermistor No. 5	Thermal	0786
31	Thermistor No. 6	Thermal	0787
32	Despin Sensor No. 1	Attitude Control	0288
33	Despin Sensor No. 2	Attitude Control	0291
34	Shunt Radiator Assembly	Electrical Power	0408
35	Magnetometer Electronics (JPL/Smith)	Instruments	0850
36	Imaging Photo — Polarimeter (U/Arizona/Gehrels)	Instruments	0857
37	Geiger Tube Telescope (U/Iowa/Van Allen)	Instruments	0853
38	Ultraviolet Photometer (USC/Judge)	Instruments	0856
39	Trapped Radiation Detector (UCSD/Fillius)	Instruments	0855
40	Infrared Radiometer (CIT/Munch)	Instruments	0858
41	Charged Particle Instrument (U/Chicago/Simpson)	Instruments	0852
42	Meteoroid Detector Electronics (LaRC/Kinard)	Instruments	0860
43	Propellant Tank	Propulsion	0929
44	Temperature Transducer	Propulsion	0929
45	Filter - Propellant	Propulsion	0929
46	Pressure Transducer	Propulsion	0929

*The reference designator is used to correlate the unit with telemetry and command "word" assignments.

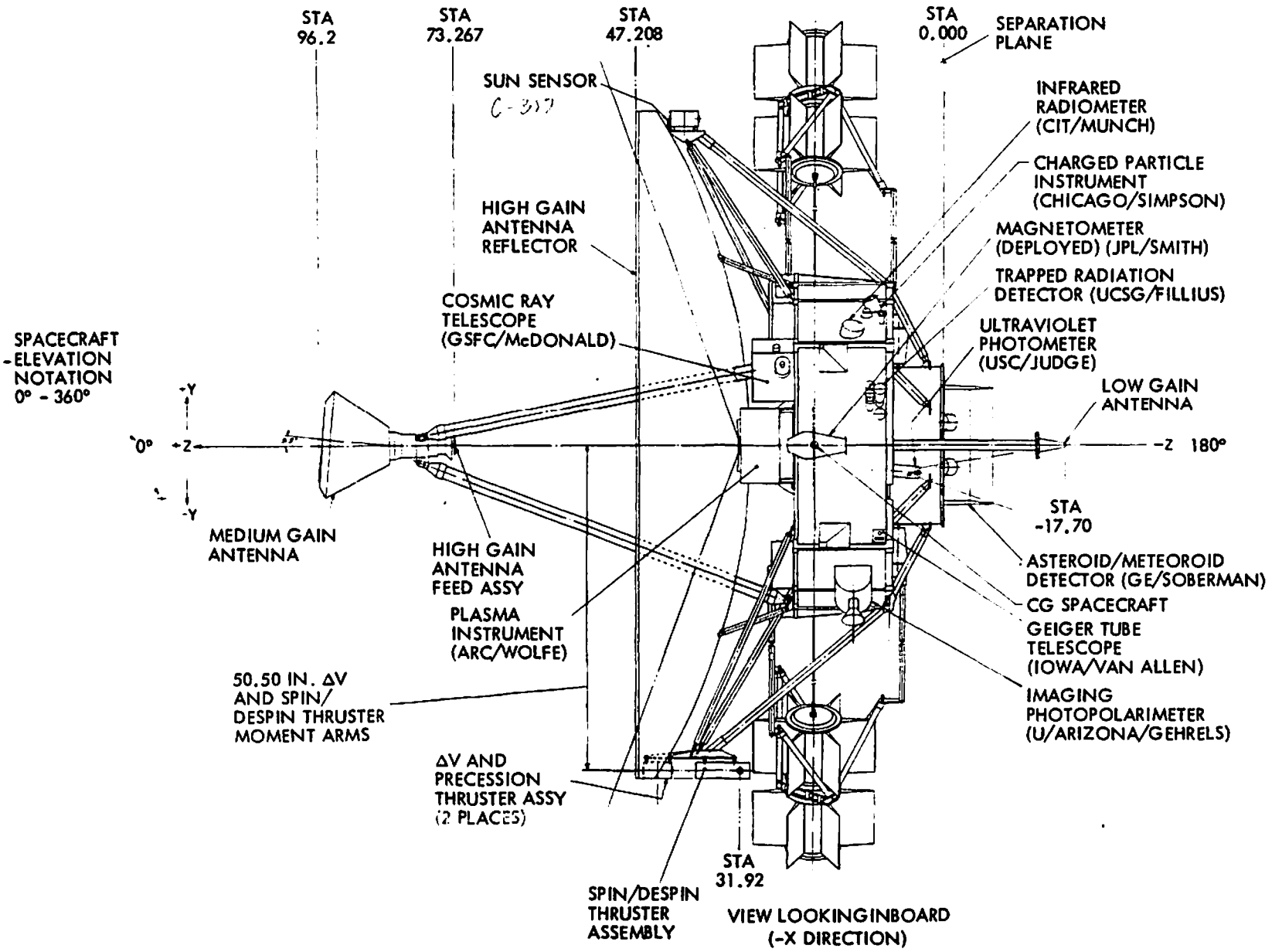
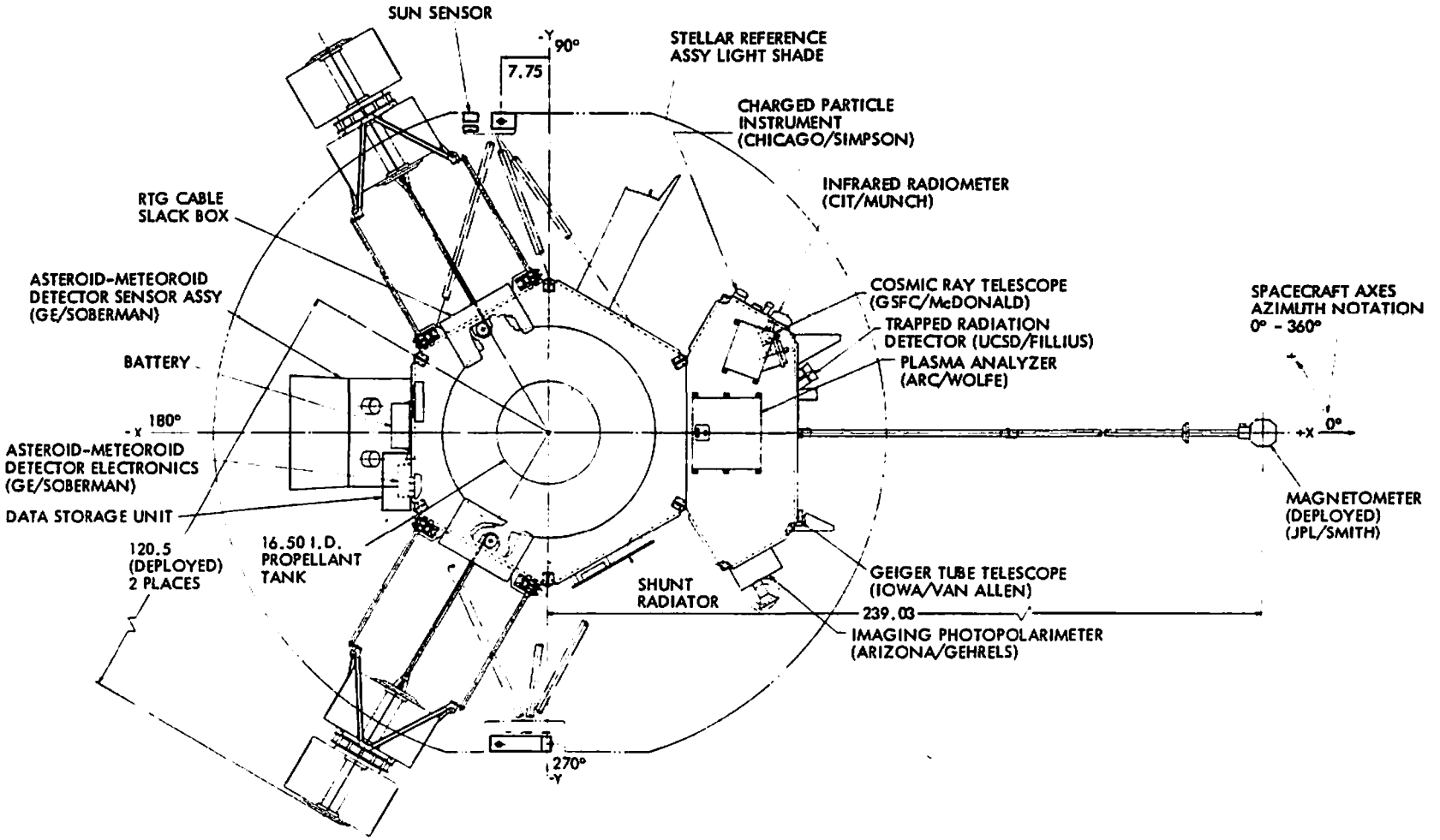
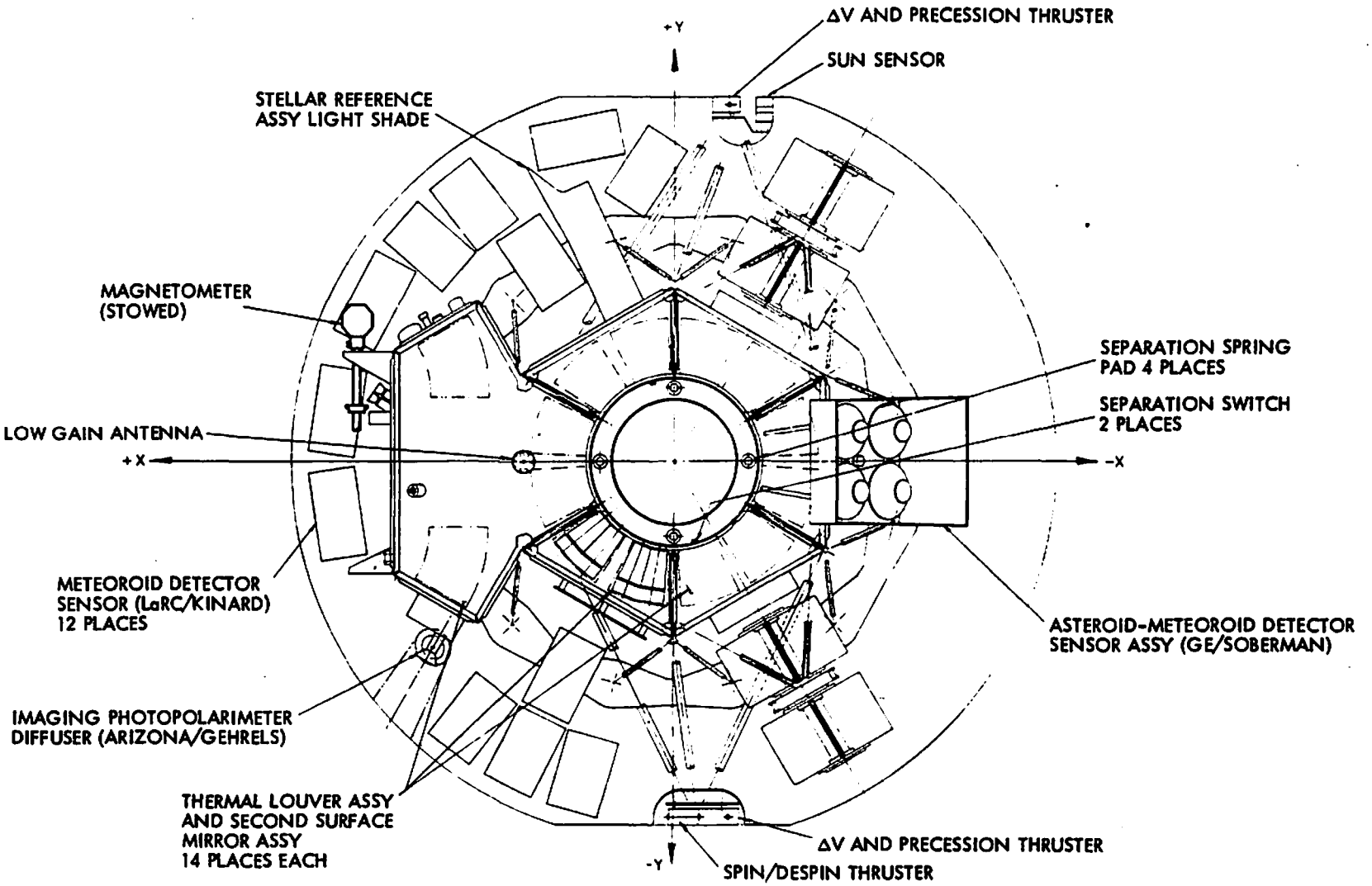

STA 96.2
STA 73.267
STA 47.208
STA 0.000
SEPARATION PLANE
SUN SENSOR
C-317
INFRARED RADIOMETER (CIT/MUNCH)
CHARGED PARTICLE INSTRUMENT (CHICAGO/SIMPSON)
HIGH GAIN ANTENNA REFLECTOR
MAGNETOMETER (DEPLOYED) (JPL/SMITH)
TRAPPED RADIATION DETECTOR (UCSG/FILLIUS)
COSMIC RAY TELESCOPE (GSFC/McDONALD)
ULTRAVIOLET PHOTOMETER (USC/JUDGE)
SPACECRAFT -ELEVATION NOTATION 0° - 360°
LOW GAIN ANTENNA
+Y
0°
+Z
-Z
180°
-Y
STA -17.70
MEDIUM GAIN ANTENNA
HIGH GAIN ANTENNA FEED ASSY
ASTEROID/METEOROID DETECTOR (GE/SOBERMAN)
CG SPACECRAFT
PLASMA INSTRUMENT (ARC/WOLFE)
GEIGER TUBE TELESCOPE (IOWA/VAN ALLEN)
50.50 IN. ΔV AND SPIN/ DESPIN THRUSTER MOMENT ARMS
IMAGING PHOTOPOLARIMETER (U/ARIZONA/GEHRELS)
ΔV AND PRECESSION THRUSTER ASSY (2 PLACES)
STA 31.92
SPIN/DESPIN THRUSTER ASSEMBLY
VIEW LOOKINGINBOARD (-X DIRECTION)

Figure 3.1-2



VIEW LOOKING AFT AT TOP OF COMPARTMENT
SHOWING RTG'S AND MAGNETOMETER DEPLOYED

Figure 3.1-3



VIEW LOOKING FORWARD
SHOWING RTG'S AND MAGNETOMETER STOWED

Figure 3.1-4

of the compartment. The Wolfe instrument views in the +Z (forward) direction through a cutout in the antenna reflector.

Figure 3.1-3 gives a view looking aft with the RTGs and magnetometer deployed. The figure shows the velocity/precession thruster axes displaced 7.75 inches from the spacecraft Y axis. This offset places the thrust axis approximately in line with the spacecraft center of gravity with the RTGs and magnetometer deployed. The spacecraft center of gravity coincides with the geometric centerline for the stowed (launch) configuration and shifts in the -X direction when the appendages are deployed. The effects of this shift are given in the mass properties discussion, Section 3.1.3.

Figure 3.1-4 presents a view looking forward with the RTGs and magnetometer stowed. Shown are the meteroid detectors (Kinard) panels on the antenna reflector, the asteroid/meteroid sensor (Soberman) and the location of the thermal louver assemblies.

The spacecraft side of the separation system is also shown in the figure. The four (4) separation springs are retained with the third stage of the launch vehicle. Upon separation, the separation switches provide a signal to the spacecraft sequencer to initiate a series of events including spacecraft despin, appendages deployment and the initial orientation maneuver.

3.1.2 Mechanisms

Spacecraft mechanisms include the spacecraft wobble damper, deployment dampers, feed movement mechanism, bolt cutters and latches. Since all of these mechanisms, except the wobble damper and feed movement mechanism, function once during the mission they are not characterized in detail. Instead, the effect of appendage deployment on spacecraft dynamics is discussed. The wobble damper characteristics are presented because of its important to overall spacecraft performance and the feed movement mechanism is discussed briefly. Detailed performance characteristics of the feed movement mechanism are presented in the Communications Subsystem Section.

Appendage release sequence and timing information is presented in the system operations section. The mechanisms and the numbers of each required is discussed below.

1) Appendage release requires the bolts that retain the appendages in their stowed position be severed by the bolt cutter assembly. There are fourteen (14) bolt cutters onboard; two for each RTG deployment rod (twelve total), two for the magnetometer boom. The second cutter at each point is redundant.

2) Control of the deployment rate and deployment loads of the RTGs and magnetometer is accomplished by the rotary viscous damper on each assembly (three total).

3) Final lockup of appendages requires that the latches on the RTG supports and magnetometer boom lock. There are ten (10) latches on the appendages; one each for the RTG rods (six total) and one at each hinge on the magnetometer boom (four total). All latches are of the same design; a slotted leaf spring that seats over a protruding catch.

The feed movement mechanism is used to position the high gain antenna feed assembly for the conscan mode. The feed will be located in the conscan position during launch with power being applied to both of the thermal actuators to maintain them under load to better withstand launch vibration and to prevent "bottoming" of the actuator at pin puller release. The second thermal actuator is redundant.

The unit will be released at the same time as the magnetometer boom by firing a redundant ordnance actuated pin puller. The unit is spring loaded so that once released it will move to the boresight position. The thermal actuators are used individually to drive the feed over to the conscan position which involves a displacement of approximately one inch in the spacecraft -Y direction. Series connected microswitches are used for each actuator to shut off heater power when the conscan position is reached. Actuation of either microswitch breaks the circuit. A separate microswitch is used to provide telemetry indication of whether the unit is in the conscan position.

3.1.2.1 RTG and Magnetometer Boom Deployment

3.1.2.1.1 RTG Deployment. The deployment of the two RTG pairs is initiated 360 seconds after spacecraft separation. The deployment has a marked effect on the spacecraft spin rate as shown in Figure 3.1-5. The RTG deployment system is designed to operate at a spacecraft spin rate of 22 $\pm$ 1.5 rpm. The maximum spin rate at which the RTGs can be deployed without structural yielding is therefore 23.5 rpm. Deployment initiation in the range from 23.5 to 28.8 rpm could result in structural loads during deployment and latchup that approach the ultimate structural capability of the system. This might result in structural yielding or possible loss of alignment. Deployment initiation at spin rates greater than 28.8 rpm will probably result in catastrophic structural failure.

The expected time to deploy both RTG pairs ranges from 65 to 300 seconds. For an initiation spin rate in 22 $\pm$ 1.5 rpm the resulting spacecraft spin rate at full deployment will be 5.0 $\pm$ 0.4 rpm. RTG deployment will not significantly increase the spacecraft average wobble angle although there will be an increase in its peak to peak variations due to the increase in spacecraft asymmetry.

3.1.2.1.2 Magnetometer Boom Deployment. The deployment of the magnetometer boom is initiated 1024 seconds after spacecraft separation. The effect of boom deployment on the spacecraft spin rate is shown in Figure 3.1-6. The boom design deployment initiation spin rate is 5.7 $\pm$ 0.4 rpm.which is the resultant spin rate when the RTGs are deployed at

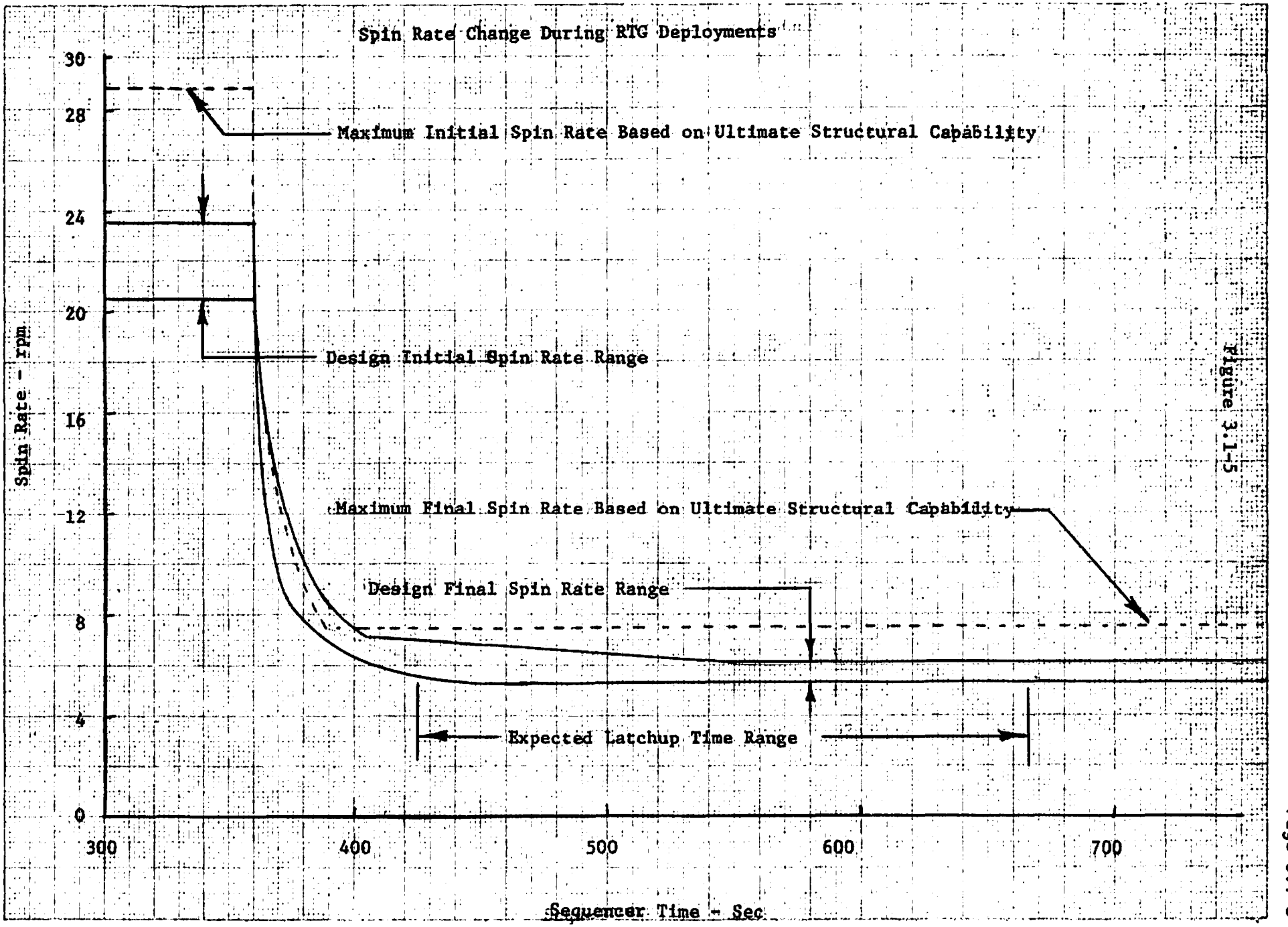

Spin Rate Change During RTG Deployments
Maximum Initial Spin Rate Based on Ultimate Structural Capability
Design Initial Spin Rate Range
Maximum Final Spin Rate Based on Ultimate Structural Capability
Design Final Spin Rate Range
Expected Latchup Time Range
Spin Rate - rpm
Sequencer Time - Sec
30
28
24
20
16
12
8
4
0
300
400
500
600
700

Figure 3.1-5

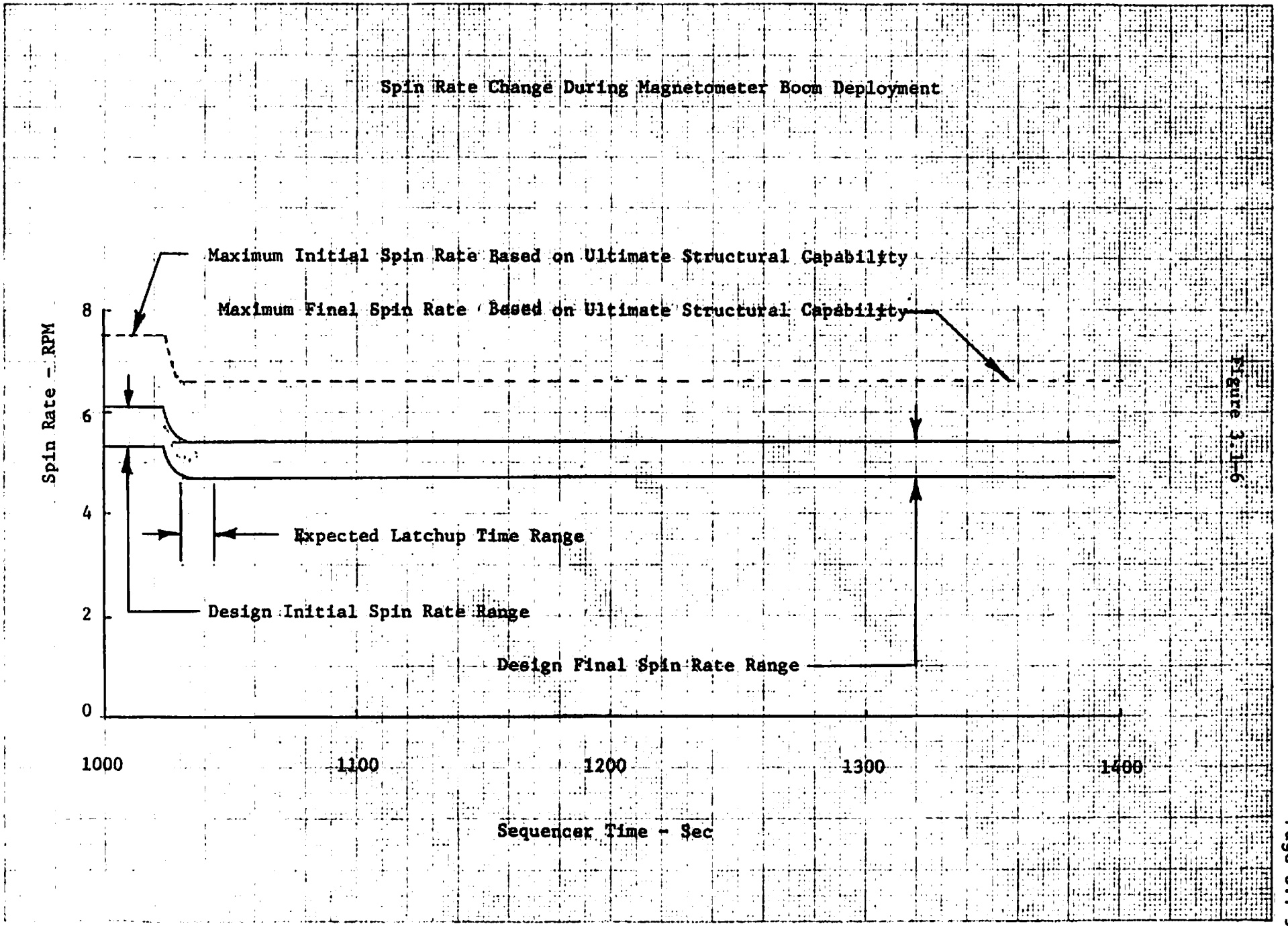

Spin Rate Change During Magnetometer Boom Deployment
Maximum Initial Spin Rate Based on Ultimate Structural Capability
Maximum Final Spin Rate Based on Ultimate Structural Capability
Expected Latchup Time Range
Design Initial Spin Rate Range
Design Final Spin Rate Range
Spin Rate - RPM
Sequencer Time - Sec
0
2
4
6
8
1000
1100
1200
1300
1400

Figure 3.1-6

22 ± 1.5 rpm. The maximum spin rate at which the boom can be deployed without structural yielding is therefore 6.1 rpm. Deployment initiation in the range from 6.1 to 7.5 rpm could result in structural loads during deployment and latchup that approach the ultimate structural capability of the boom. This might result in structural yielding or possible loss of alignment. Deployment initiation at spin rates greater than 7.5 rpm will probably result in catastrophic structural failure.

The expected time to deploy the boom ranges from 6 to 60 seconds. After boom deployment the spacecraft spin rate should then be trimmed to the final nominal rate of 4.8 rpm. Deployment of the magnetometer boom will reduce the spacecraft wobble angle since the boom also activates the spacecraft wobble damper.

The maximum spacecraft spin rate allowable after the magnetometer boom and RTGs are deployed is 13.0 rpm, based upon a no structural yield criterion.

3.1.2.2 Wobble Damper. The Pioneer F/G Wobble Damper consists of a pair of fluid-filled bellows which are attached to both the spacecraft and to the base of the magnetometer boom. Figure 3.1-7 shows a perspective view of the damper and Figure 3.1-8 shows how the damper attaches to the spacecraft and magnetometer boom. The magnetometer boom is free to move relative to the damper (in the spacecraft X-Z plane) but is restrained from rotating about any other axis. When the boom moves about its pivot due to spacecraft wobble motion, the center plate containing the damper manifold is displaced from its nominal position such that the volume of one bellows is reduced while the volume of the other is increased correspondingly; hence, fluid flows through the damping tubes from one side of the assembly to the other. Wobble motion is damped, and the spacecraft returned to the pure spin condition, by virtue of the energy dissipated by the fluid during viscous flow from one bellows to the other.

Figure 3.1-9 depicts damper performance in the form of time constant as functions of spin rate variations and changes in temperature for nominal spacecraft mass properties. Each of the three curves are labeled to indicate the appropriate spin rate; 1) 50% of nominal, 2) nominal spin rate of 4.8 rpm, and 3) 150% of nominal. The temperature uncertainty range (55° - 35°) for Jupiter encounter is shown on the graph and the damper performance over that range is bounded for each of the three spin rates by the two vertical lines drawn on the figure. Figure 3.1-10 is a plot of time constant vs spin rate for nominal mass properties and nominal temperature at Jupiter encounter.

The time constant associated with damper performance corresponds to the time required to reduce the wobble angle by a factor of 1/3. The wobble angle θ at any time t is given by

$$\theta = \theta_0 e^{-t/\tau}$$

where θ_0 is the initial wobble angle and τ is the time constant.

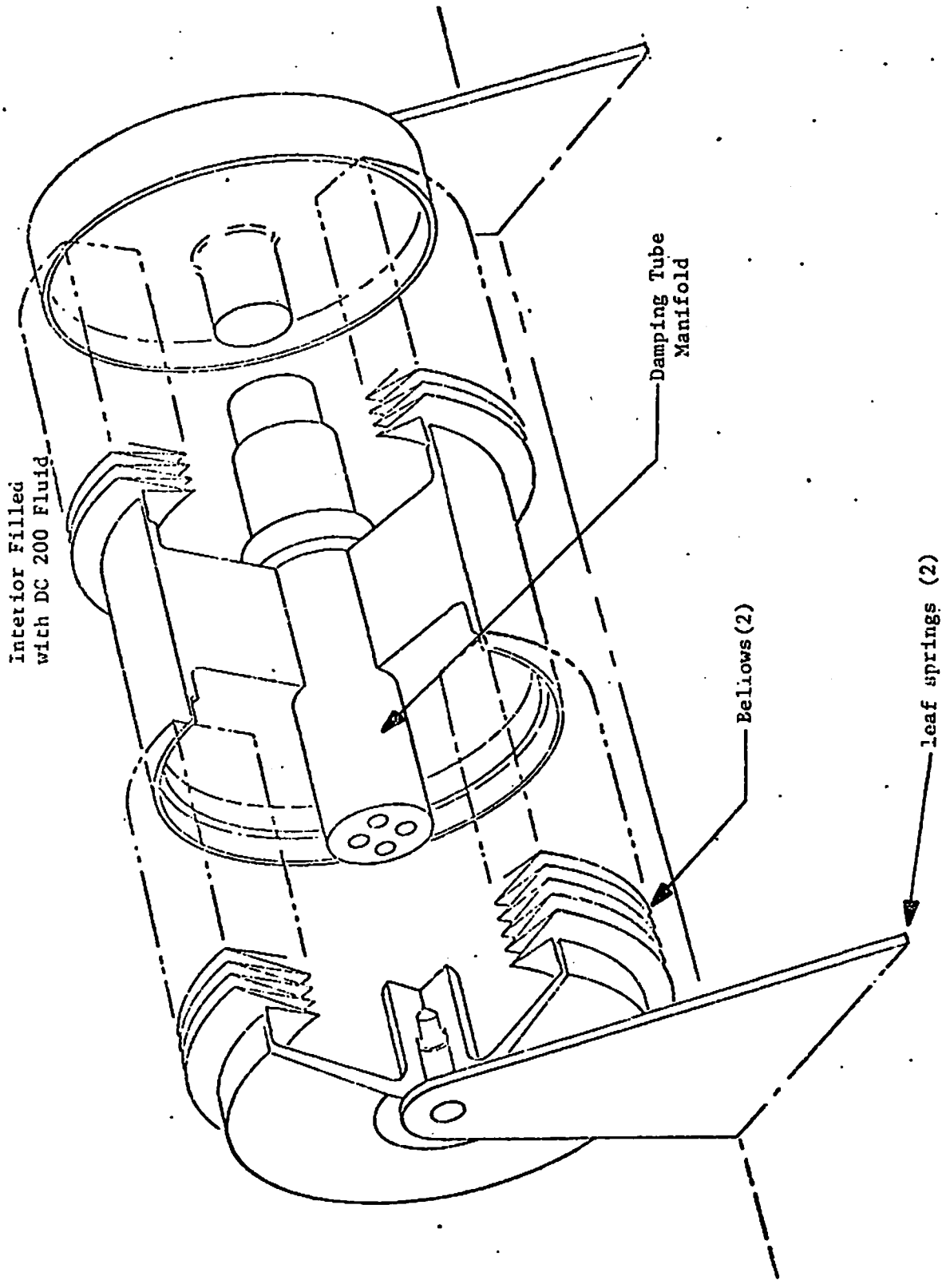


Figure 3.1-7 Perspective View of Wobble Damper

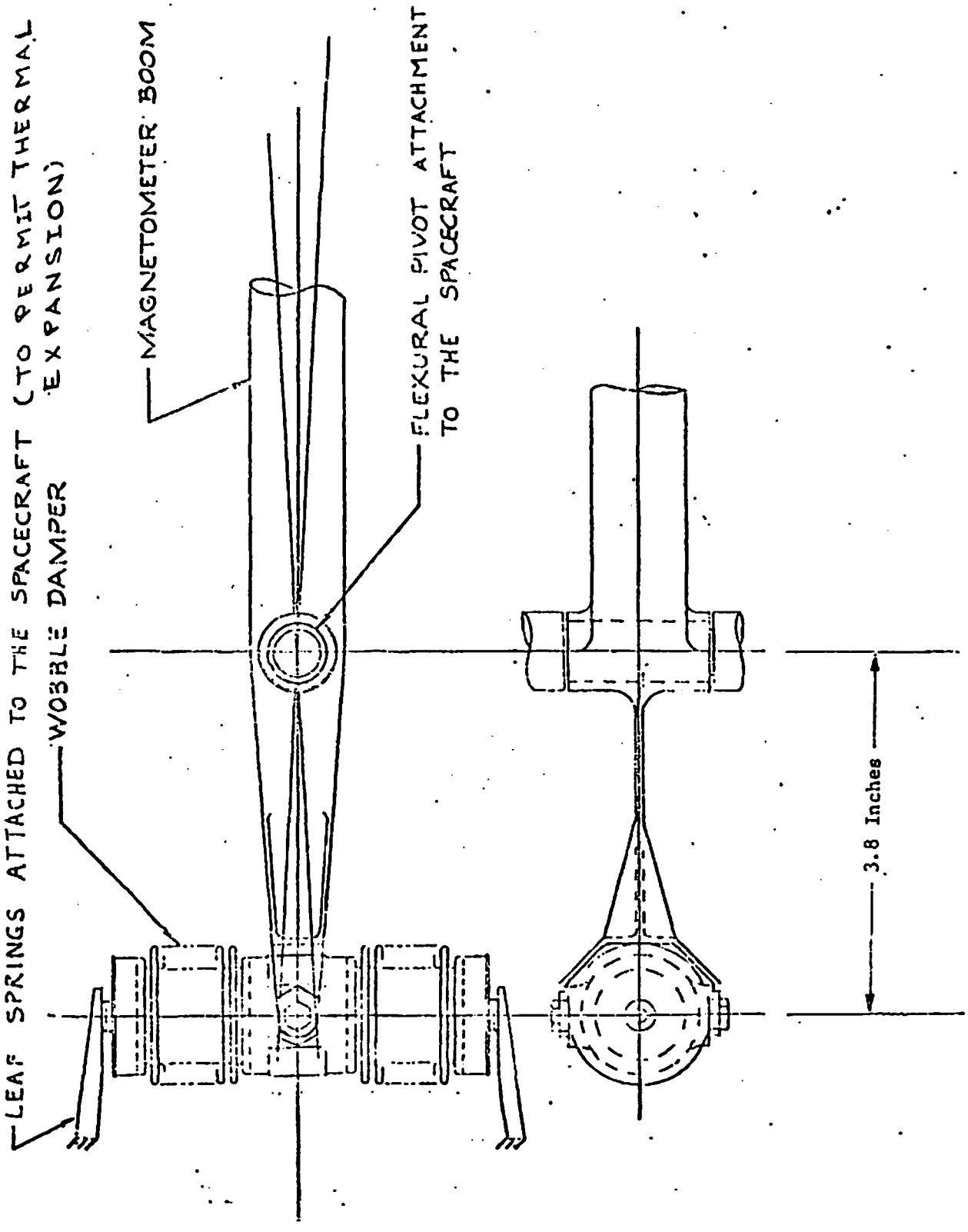


Figure 3.1-8 Operational Schematic of Wobble Damper

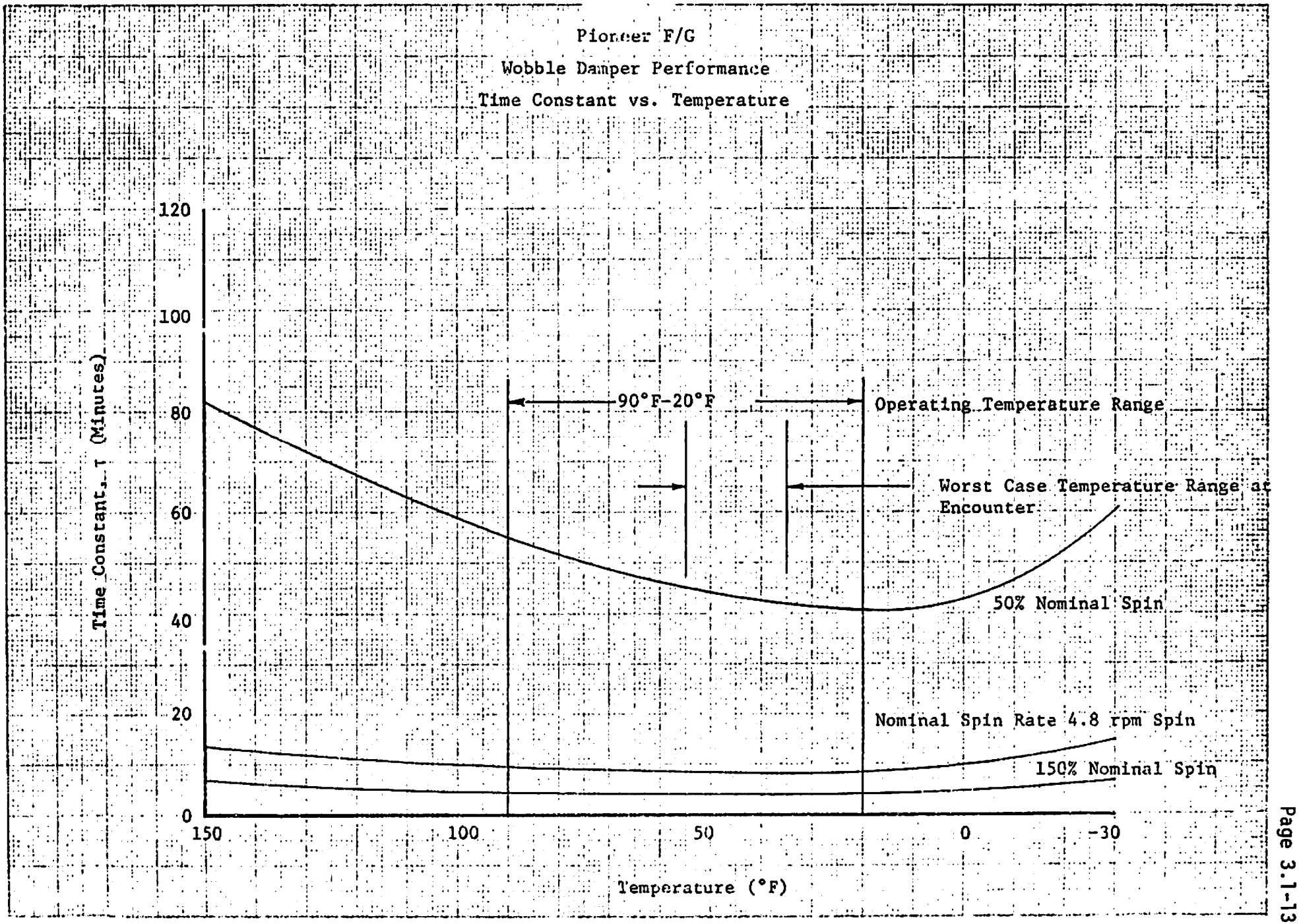

Pioneer F/G
Wobble Damper Performance
Time Constant vs. Temperature
Time Constant, τ (Minutes)
120
100
80
60
40
20
0
150
100
50
0
-30
Temperature (°F)
90°F-20°F
Operating Temperature Range
Worst Case Temperature Range at Encounter
50% Nominal Spin
Nominal Spin Rate 4.8 rpm Spin
150% Nominal Spin

Figure 3.1-9

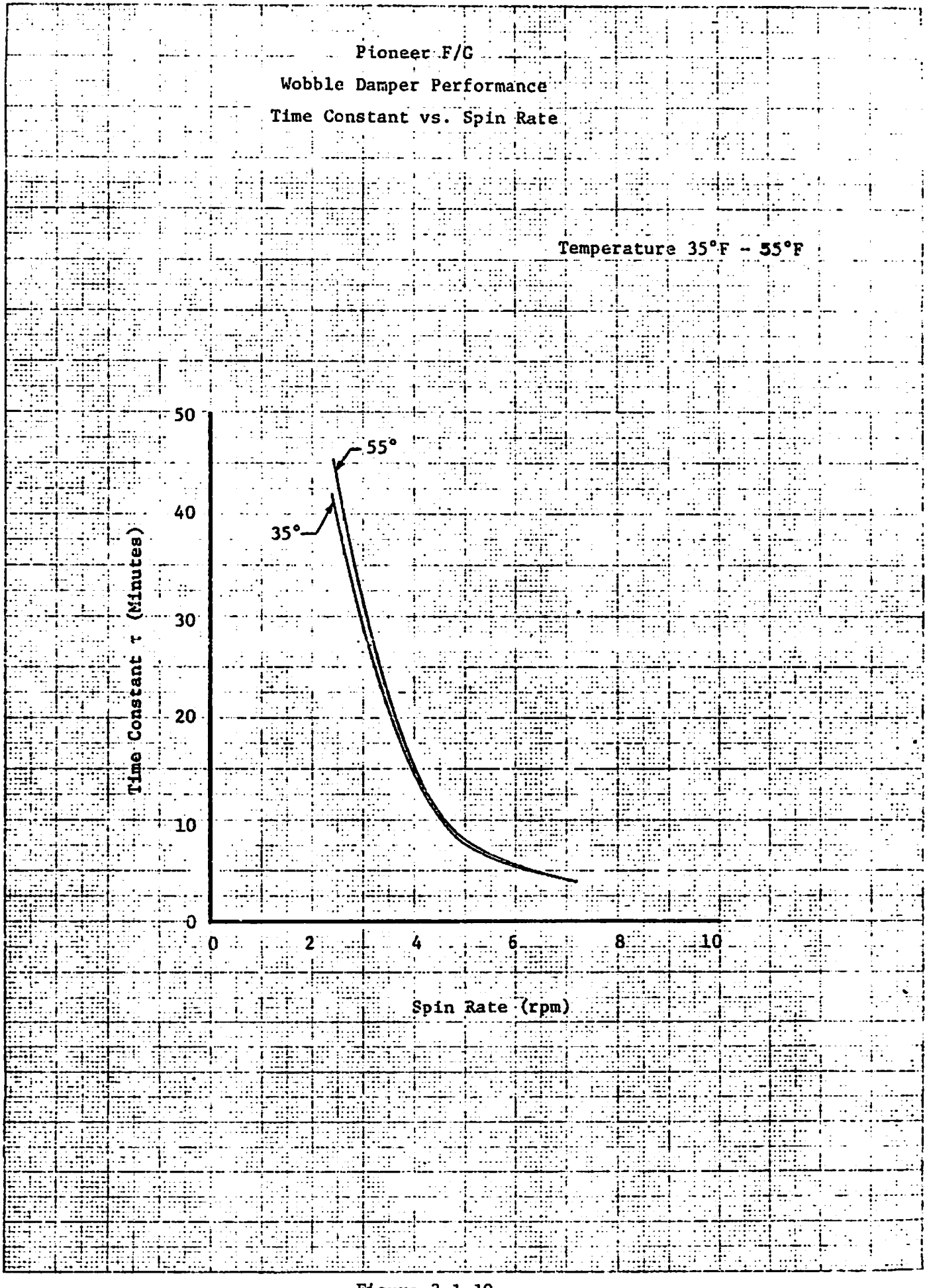

Pioneer F/G
Wobble Damper Performance
Time Constant vs. Spin Rate
Temperature 35°F - 55°F
55°
35°
Time Constant τ (Minutes)
50
40
30
20
10
0
0
2
4
6
8
10
Spin Rate (rpm)

Figure 3.1-10

The time t, therefore, required to reduce an initial wobble angle θ_0 to a specified value θ_1 can be calculated from

$$t = \tau \ln \frac{\theta_0}{\theta_1}$$

3.1.3 Mass Properties

The spacecraft mass properties are summarized in Table 3.1-1. The spacecraft is statically and dynamically balanced about the geometric centerline (Z axis) with appendages stowed. Upon deployment of the RTGs and magnetometer the spacecraft center of gravity shifts in the -X direction. However, because the RTGs and magnetometer are deployed in a plane containing the spacecraft center of gravity and parallel to the XY plane, the longitudinal principal inertia axis remains parallel to the spacecraft geometric centerline (Z axis). The principal axis by definition passes through the center of gravity; hence, the spacecraft in the deployed configuration spins about an axis that is parallel to the Z axis but is displaced a distance equal to the center of gravity shift. Table 3.1-1 shows this displacement to be 6.7 inches for the beginning point in the mission, i.e., full propellant load.

Depletion of the propellant onboard causes an additional center of gravity shift of 0.8 inch. Another effect of propellant expulsion is the introduction of dynamic unbalance about the longitudinal axis. The unbalance results from the propellant center of gravity being displaced in both the longitudinal (Z) and lateral (X) directions relative to the spacecraft center of gravity. The net effect of the dynamic unbalance introduced by propellant expulsion is a slight rotation of the spin axis with respect to its nominal orientation, which is parallel to the Z axis. The rotation occurs in the XZ plane. The magnitude of the rotation is given as a function of propellant expended in Figure 3.1-11.

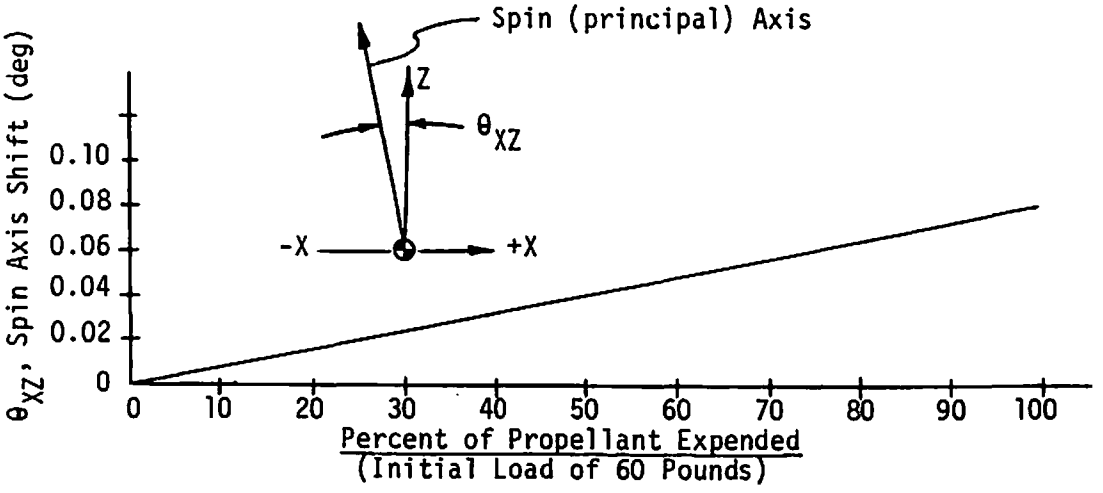


Figure 3.1-11. Spin Axis Shift vs Propellant Expended

Table 3.1-1. Spacecraft Mass Properties[1]

Condition	Weight (lb)	Center of Gravity (in.)[2]			Moment of Inertia (slug-ft^2)			$\frac{2 I_Z}{I_X + I_Y}$
		$\bar{X}$	$\bar{Y}$	$\bar{Z}$	I_X	I_Y	I_Z	
Stowed (launch)	553.0	0	0	19.2	68.5	61.6	99.0	1.52
Deployed--Full Propellant Load	553.0	-6.7	0	19.2	284.5	180.5	433.9	1.87
Deployed--Propellant Expended	494.3	-7.5	0	19.4	284.4	179.8	433.2	1.87

NOTES: [1] Final values to be supplied; values given are current estimates.

[2] Referenced datum: $\bar{Z}$ (longitudinal) measured from spacecraft separation plane.
$\bar{X}$ and $\bar{Y}$ (transverse) measured from spacecraft geometric centerline.

SECTION 3.1 NOTATION

C.G.	Center of Mass
I_x	Moment of Inertia about x principal axis
I_y	Moment of Inertia about y principal axis
I_z	Moment of Inertia about z principal axis
rpm	Revolutions per minute
RTG	Radioistope Thermoelectric Generator
t	Time
τ	Wobble damper time constant (nominal 8 minutes at 4.8 rpm)
θ	Wobble angle - angle between spin axis and momentum vector
θ_o	Initial Wobble angle
θ_{xz}	Shift of principal z axis from spacecraft z axis
$\bar{x}$	Location of Vehicle c.g. from spacecraft axes
$\bar{y}$	Location of Vehicle c.g. from spacecraft axes
$\bar{z}$	Location of Vehicle c.g. from separation plane

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3.2 THERMAL CONTROL SUBSYSTEM

3.2.1 Thermal Control Subsystem Description

Basically the Pioneer F/G thermal control concept consists of an insulated equipment compartment with passively controlled heat rejection via an aft mounted louver system.

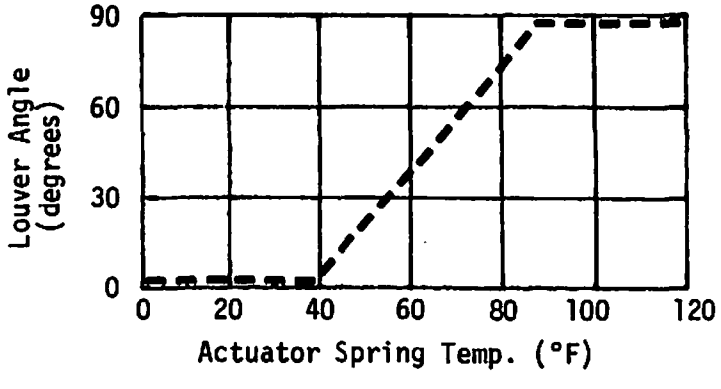
The side panel blankets adjacent to the stowed RTGs are 22 layers of 1/4-mil aluminized Kapton with 2-mil aluminized Kapton outer layers. The other side panel and forward blankets consist of similar blankets except made of Mylar. The aft surface blankets consist of 22 layers of 1/4-mil Kapton with 2-mil outer layers.

Functionally, the louver system acts to control the heat rejection of the radiating platform by varying the angular position of a set of highly reflective blades or shutters. A bimetallic spring, thermally coupled radiatively to the platform, provides the motive force for altering the angle of each blade. In a closed condition the heat rejection of the platform is minimized by virtue of the "blockage" of the blades while open louvers provide the platform with a nearly unobstructed view of space.

The Pioneer F/G louver system consists of 30 individually actuated blades or vanes, 24 of which are positioned radially around the interstage adapter and 6 of which are mounted on the aft side of the experiment platform. Each trapezoidal blade is supported on the larger end by a bimetallic spring/actuator housing assembly and on the smaller end by a lexan bearing assembly which, for the 2-bladed assemblies, are contained in the interstage ring. Each blade is constructed from 0.156 inch aluminum honeycomb core with 2-mil aluminum face sheets. The overall blade thickness is approximately 0.160 inch.

The bimetallic spring is thermally connected (radiatively) to the platform and varies the angular setting of the blades according to the platform temperature in its particular sector.

A louver angle-spring temperature plot is shown below:



Each actuator housing is individually insulated with an aluminized Mylar "hat." The platform surface underneath the louver blades is coated with high emittance second surface mirrors to obtain a very low α for the aft-sun condition and a high effective platform emittance in the open position. The louver blade facing the platform is bare aluminum (Infrared Absorptance = 0.04) to minimize radiative exchange between the platform and the blade.

A listing of the flight temperature measurements, and their location is presented in Figure 3.2-1.

Table 3.2-1 presents the spacecraft exterior surface's radiometric properties, i.e., solar absorptance (α) and I.R. emittance (ε). These properties are useful for surface temperature evaluation at various sun angles.

3.2.2 Functional Operation

3.2.2.1 On Stand. During on stand operation, with nominal air conditioning airflow, established temperatures internal to the compartment range from approximately 80°F in the experiment bay to 101°F for the PCU. The following table presents some of the major component temperatures expected during this condition.

PCU	101 °F	20°F/Hr
TRF	91 °F	20°F/Hr
Inverter	92°F	20°F/Hr
TWTA	99°F	20°F/Hr
Receivers	84°F	5°F/Hr
Exper. Platform	82°F	5°F/Hr

Also shown in the table is the approximate temperature rise of these components if air conditioning airflow is interrupted.

3.2.2.2 Steady State Component Temperatures. Major component temperatures at selected combinations A.U., sun angle and internal power dissipation are presented in Table 3.2-2. The temperatures presented are, in general, component average baseplate temperatures and in many cases, will not correspond to flight measurements made on components internal to the boxes, i.e., TWT collector, etc. These temperatures provide a measurement of nominal spacecraft state of health conditions.

Front sun conditions assume the sun on the front of the high gain dish (spacecraft shadowed) and are applicable for sun angles between 0° and 20°. For the side sun condition the sun is perpendicular to the spin axis and the temperatures are applicable over a range of approximately 90 $\pm$ 10°. In the Table,1 A.U. represents near earth and 5.5 A.U. at Jupiter.

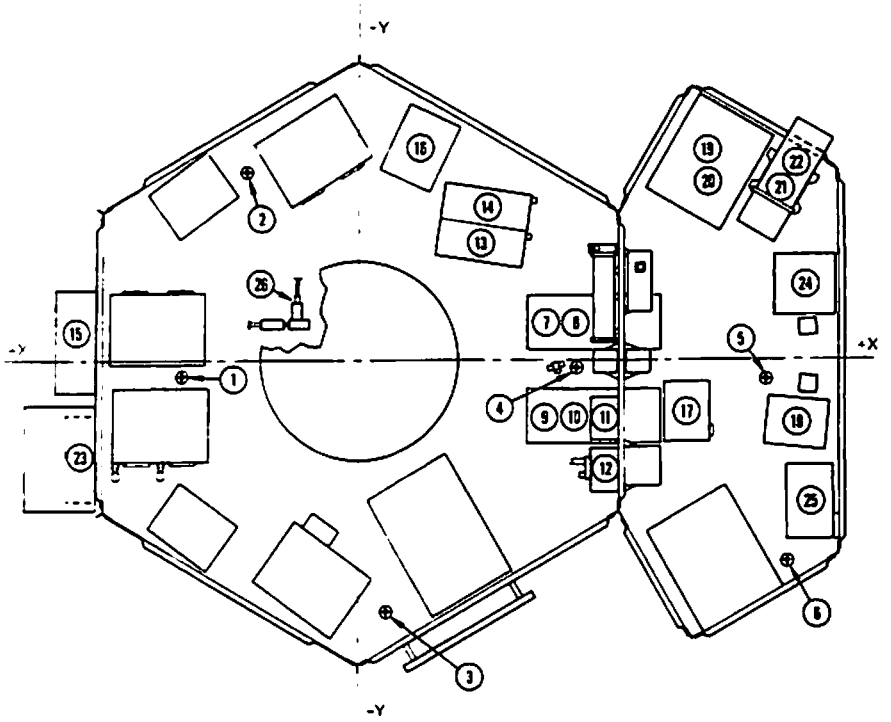


Figure 3.2-1

TEMPERATURE SENSOR LOCATIONS

IDENTIFICATION NO.	IDENTIFICATION YLM "Word"	DESCRIPTION	LOCATION
		Thermal	
1	C-301	Equipment Platform	On platform
2	C-302	Equipment Platform	On platform
3	C-304	Equipment Platform	On platform
4	C-318	Equipment Platform	On platform
5	C-319	Equipment Platform	On platform
6	C-320	Equipment Platform	On platform
		Communications	
7	C-205	TWTA-A	Collector, TWTA-A
8	C-207	Converter, TWTA-A	Power Supply, TWTA-A
9	C-228	TWTA-B	Collector, TWTA-B
10	C-221	Converter, TWTA-B	Power Supply, TWTA-B
11	C-206	Driver-A, Auxiliary Oscillator	TCXO Subassembly
12	C-223	Driver-B, Auxiliary Oscillator	TCXO Subassembly
13	C-222	Receiver-A	VCO Subassembly
14	C-227	Receiver-B	VCO Subassembly
		Power	
15	C-115	Battery	On housing near connector
		Attitude Control	
16	C-303	Stellar Reference Assembly(SRA)	On amplifier board
	C-317	Sun Sensor Assembly(SSA)	Between circuit boards in SSA
		Instruments	
17	E-~~102~~ 202	Magnetometer Electronics(JPL/Smith)	Internal to unit
18	E-109	Ultraviolet Photometer(USC/Judge)	Internal to unit
19	E-110	Charged Particle(U/Chicago/Simpson)	Internal to unit
20	E-213	Charged Particle(U/Chicago/Simpson)	Internal to unit
21	E-117	Infrared Radiometer, Low Range(CIT/Munch)	Internal to unit
22	E-201	Infrared Radiometer, High Range(CIT/Munch)	Internal to unit
23	E-118	Asteroid/Meteoroid Detector Electronics, Preamplifier(GE/Soberman)	Internal to unit
24	E-209	Trapped Radiation Detector(UCSD/Fillius)	Internal to unit
25	E-221	Geiger Tube Telescope(U/Iowa/Van Allen)	Internal to unit
	E-125	Cosmic Ray Electronics(GSFC/McDonald)	Internal to unit } Mounted on top of
	E-128	Cosmic Ray Detector(GSFC/McDonald)	Internal to unit } instruments compartment
	E-101	Arc Plasma Detector(Arc/Wolfe)	Internal to unit } (see Figure 3.1-2)
		Propulsion	
26	C-327	Propellant Supply	Propellant liquid at filter
	C-309	VPT Propellant Inlet Manifold	VPT Assembly No. 1, -Y axis
	C-326	VPT Propellant Inlet Manifold	VPT Assembly No. 2, +Y axis
	C-310	SCT Propellant Inlet Manifold	SCT Assembly, -Y axis
	C-311	VPT No. 1, TCA No. 2	Catalyst bed, +Y axis (Top)
	C-312	VPT No. 2, TCA No. 2	Catalyst bed, +Y axis (Bottom)
	C-328	VPT No. 3, TCA No. 1	Catalyst bed, -Y axis (Top)
	C-325	VPT No. 4, TCA No. 1	Catalyst bed, -Y axis (Bottom)
		RTG's	
	C-201	RTG No. 1 Fin Root	Cylindrical Case, Inboard RTG, -Y axis
	C-202	RTG No. 2 Fin Root	Cylindrical Case, Outboard RTG, -Y axis
	C-203	RTG No. 3 Fin Root	Cylindrical Case, Inboard RTG, +Y axis
	C-204	RTG No. 4 Fin Root	Cylindrical Case, Outboard RTG, +Y axis
	C-217	RTG No. 4 Hot Junction	Hot junction thermocouple, Outboard RTG, +Y axis
	C-218	RTG No. 3 Hot Junction	Hot junction thermocouple, Inboard RTG, +Y axis
	C-219	RTG No. 2 Hot Junction	Hot junction thermocouple, Outboard RTG, -Y axis
	C-220	RTG No. 1 Hot Junction	Hot junction thermocouple, Inboard RTG, -Y axis

Table 3.2-1.

SPACECRAFT EXTERIOR SURFACES RADIOMETRIC PROPERTIES

	GENERAL DESCRIPTION	SOLAR ABSORPTANCE (α)	EMITTANCE (ϵ)	RATIO ($\frac{\alpha}{\epsilon}$)
Antenna Subsystem				
High Gain Antenna, Front (+Z) Surface	White Paint	0.21	0.85	0.25
High Gain Antenna, Rear (-Z) Surface	Bare Aluminum	0.17	0.04	4.3
Mid Gain Antenna, Int.	Black Paint	0.95	0.86	1.1
Mid Gain Antenna, Ext.	White Paint	0.21	0.85	0.25
High Gain Feed Horn, Int.				
High Gain Feed Horn, Ext.	White Paint	0.21	0.85	0.25
Feed Movement Mechanism Insulation, Outer Surface	3-mil Alum. Mylar	0.18	0.75	0.24
Aft Omni Antenna, Ext. Surface	Bare Fiberglass	≈0.72	≈0.89	0.81
Attitude Control Subsystem				
SRA Flight Shade, Ext.	White Paint	0.21	0.85	0.25
SRA Flight Shade, Int.	Black Paint	0.95	0.86	1.1
SSA Insulation, Ext. Surface	2-mil Mylar	0.18	0.75	0.24
Electrical Power Subsystem				
Strip Heater Assy. (Shunt Plate)	White Paint	0.21	0.85	0.25
Battery Insulation Blanket, Ext. Layer	2-mil Aluminized Mylar	0.17	0.70	0.24
Battery Radiating Plate	Silver Backed Teflon	0.15 (Degraded)	0.66	0.23
Data Handling				
DSU Insulation Blanket, Ext. Layer	2-mil Mylar	0.17	0.70	0.24
DSU Exterior Radiating Plate	TBD	TBD	TBD	TBD
Propulsion				
Thruster Cluster Radiating Surface	Second Surface Mirrors	0.07	0.82	0.09
Thruster Cluster Insulation, Ext. Surface	3-mil Kapton	0.44	0.78	0.56
LHA Insulation, Ext. Surface	3-mil Kapton (Alum. Side Out)	0.12	0.04	3.0
Thermal Control Subproject				
Aft Insulation, Ext. Surface	2-mil Alum. Kapton	0.40	0.70	0.57
+Y -X & -Y -X Side Panel Insulation, Ext. Surface	2-mil Alum. Kapton	0.40	0.70	0.57
Remaining Side Panel & Top Insulation, Ext. Surface	2-mil Alum. Mylar	0.17	0.70	0.24
Strut Insulation, Ext. Surface	2-mil Alum. Mylar	0.17	0.70	0.24
Louver Blades	Bare Aluminum	0.17	0.04	4.3
Platform Surfaces below Louver Blades	Second Surface Mirrors	0.07	0.82	0.09
Miscellaneous				
RTG'S	White Paint	0.20	0.83	0.24
RTG Delta Frame & Bipods	Anodized Aluminum	≈0.2	≈0.7	0.29
RTG Guide Rods	Bare Aluminum	0.17	0.04	4.3
RTG Power Supply Cables, Outer Covering	Kapton	0.3	0.63	0.48
Kinnard Meteoroid Sensors, Side Facing Aft		0.36	0.09	4.0
Kinnard Meteoroid Sensors, Side Facing Antennas	Black Paint	0.98	0.90	1.1

Table 3.2-2.

PREDICTED TEMPERATURES OF MAJOR COMPONENTS (°F)
(AVERAGE BASEPLATE)

	ACCEPTANCE TEMPERATURES		FRONT SUN, NORMAL POWER		SIDE SUN, NORMAL POWER 1 AU	AFT SUN, 1 AU (AFTER 4 HRS) (4)	FRONT SUN, 5.5 AU		
	MAX.	MIN.	1 AU (EXP. ON)	5.5 AU (EXP. ON)			NORMAL POWER EXP. OFF	NORMAL POWER EXP. OFF TWTA'S OFF	REDUCED POWER (100 WATTS) EXP. ON
THERMAL									
Equipment Bay Platform, -X Axis	-	-	118	86	149	160	108	78	57
Equipment Bay Platform, -X, +Y Axis	-	-	73	59	108	121	62	48	51
Equipment Bay Platform, -Y Axis	-	-	62	50	85	99	48	27	44
Equipment Bay Platform, +X Axis	-	-	77	63	102	111	50	7	58
Experiment Bay Plat., +X Axis	-	-	61	50	96	96	24	-10	46
Experiment Bay Plat., +X, -Y Axis	-	-	54	46	81	84	23	-10	43
ATTITUDE CONTROL									
Stellar Reference Ass'y.	75(1)	0	49	31	87	69	4	-20	27
Sun Sensor Ass'y	164	5	160	5	140	—	5	5	5
CEA	105	20	76	60	102	111	58	38	50
POWER									
Battery	70(1)	30	77	53	104	110	60	39	37
PCU	160	20	127	91	156	167	116	85	59
Inverter	145	20	113	87	140	150	93	66	71
TRF	140	20	106	71	138	152	78	71	68
COMMUNICATIONS									
TWTA #1	125(2)	20	84	69	110	118	53	5	65
TWTA #2	125(2)	20	77	63	102	111	50	6	58
Drivers	100	20	72	56	100	106	42	14	50
Receivers	100	20	66	56	93	102	45	25	51
Conscan	105	20	76	60	102	111	58	37	50
PROPULSION									
Propellant Tank	140	40	77	59	103	104	55	33	50
DATA HANDLING									
DTU	95	20	72	58	97	108	55	30	51
DSU	105	20	72	55	96	106	50	28	48
DDU	105	20	76	60	102	111	58	37	50
ELECTRICAL DISTRIBUTION									
CDU	105	20	71	56	98	108	53	37	49
EXPERIMENTS									
Mag. Elect. (JPL/Smith)	149(3)	- 4	78	66	106	110	42	0	58
Meteoroid Elect. (LaRC, Kinnard)	122	-22	72	59	101	103	38	6	51
Char. Part. (U of Chig.,Simpson)	104	-20	46	38	83	75	27	-18	34
I.R. (CIT, Munch)	90	-22	60	48	92	95	39	21	42
Trapped Rad. (UCSD, Fillius)	104	-20	53	43	102	83	20	-16	39
UV Photo. (USC, Judge)	122	-40	63	53	101	103	23	-10	48
Geiger Tube (U of I, Van Allen)	100	-20	58	47	94	90	19	-12	44
Imaging Photo. (U of A, Gehrels)	120	-20	61	52	92	92	25	- 9	48
Cosmic Ray (GSFC, McDonald)	104	-20	58	49	95	88	9	-16	44
Plasma Anal. (ARC, Wolfe)	90	-20	70	60	102	102	24	- 4	53
Ast/Met. Elect. (Soberman)	122	-58	102	73	131	139	76	56	58

NOTES
1. Applicable for Front Sun Only
2. Platform Below TWTA'S
3. Limits from PC-220
4. Aft Sun Transient Starts from Side Sun

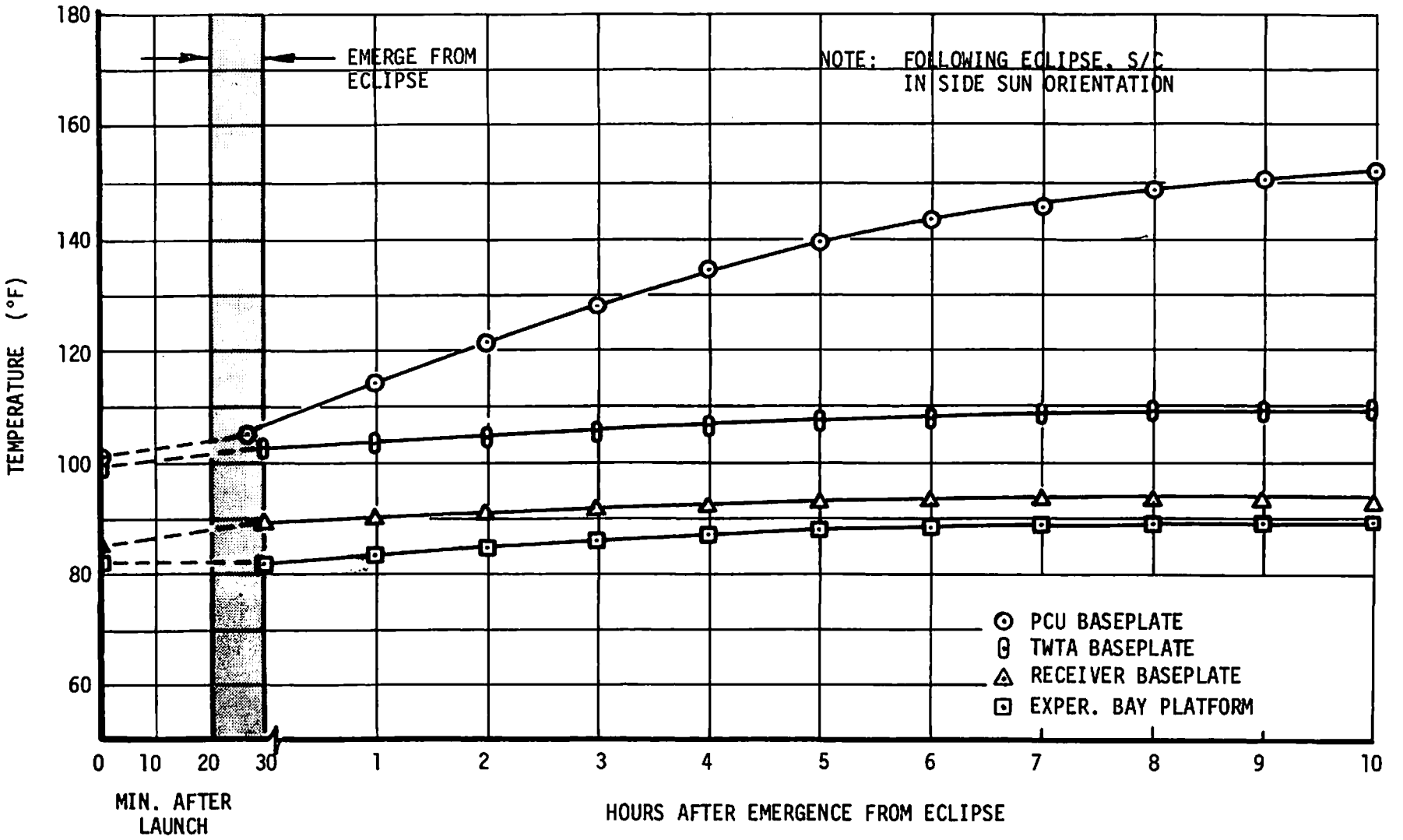
3.2.2.3 Transient Temperatures. Figure 3.2-2 presents the transient response of the PCU, TWTA, Receivers, and experiment platform from lift-off through the first 10 hours following emergence from the earth's shadow. It is assumed that the experiments are off at lift-off and are turned on simultaneous with spacecraft coming into the sunlight in a side sun attitude. The temperature changes in the first 30 minutes are relatively small due to long spacecraft thermal time constant and the lack of solar load. Eventually, in 15 to 20 hours, the spacecraft temperatures will assume the steady state side sun values shown in Table 3.2-2.

3.2.2.4 Electronic Equipment Operation. The only combinations of equipment power variations considered are normal power (4 RTGs), reduced power (3 degraded RTGs), experiments off and experiments and TWT off. The component temperature at these various power conditions are presented in Table 3.2-2.

Figure 3.2-2

COMPARTMENT EQUIPMENT TEMPERATURES FOLLOWING LIFT-OFF

(°F)



SECTION 3.2 NOTATION

α	Absorptance
A.U.	Astronomical Unit (Sun-Earth Distance)
CDU	Command Distribution Unit (Described in Section 3.9)
CEA	Control Electronics Assembly (Described in Section 3.3)
DDU	Digital Decoder Unit (Described in Section 3.9)
DSU	Data Storage Unit (Described in Section 3.5)
DTU	Digital Telemetry Unit (Described in Section 3.5)
ϵ	Emittance
EXP	Experiments
I.R.	Infra-red
PCU	Power Control Unit (Described in Section 3.7)
RTG	Radioisotope Thermoelectric Generator (Described in Section 3.8)
SCT	Spin Control Thruster
TRF	Central Transformer-Rectifier-Filter Assembly (CTRF)
TWTA	Travelling Wave Tube Amplifier (Described in Section 3.6)
VPT	Velocity Precession Thruster

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3.3 ATTITUDE CONTROL SUBSYSTEM (ACS)

3.3.1 ACS Description

3.3.1.1 Equipment Configuration, and Coordinate Definition. The ACS subsystem consists of a sun sensor assembly, a stellar reference assembly, a despin sensor assembly, and the control electronics assembly. The ACS units are identified and functionally described below.

- Control Electronics Assembly (CEA). The CEA processes sensor inputs, input commands and data and produces outputs to fire the appropriate velocity, precession and spin control thrusters. The CEA controls power to all units within the CEA and provides processed sensor pulses and telemetry outputs.

- Sun Sensor Assembly (SSA). The SSA provides sun pulses to establish a reference plane for open loop precession. It also serves as a redundant roll reference for indexing the experiments.

- Stellar Reference Assembly (SRA). The SRA provides star pulses which are used as the primary (and most accurate) roll reference for indexing the experiments. The SRA pulses may also be used as a closed loop precession reference or as an open loop precession reference if the precession is within 15° of the ecliptic plane.

- Despin Sensor Assembly (DSA). The DSA provides a discrete onboard signal when spin speed is reduced to the level required for RTG deployment. The DSA is used only in the despin operation. Redundant DSAs are used.

The locations of the ACS equipment on the spacecraft are shown in Figure 3.1-1. (For detailed information on equipment location, see Section 3.1.) The spacecraft coordinates are defined in this figure. The +z (z body) axis coincides with the spin axis with spin defined as positive along this axis as given by the right hand rule. The $+z_b$ axis also coincides with the axis of symmetry of the high gain antenna. The +x axis is located in the direction of the magnetometer boom, and the +y axis completes the orthogonal triad. The sun sensor is located near the edge of the high gain dish and has a field of view which lies in the +y, $\pm$z plane. The stellar reference assembly is located in the equipment compartment and has a field of view which lies in the plane defined by the z and S axes, where the S axis is 25° from the +y axis as shown. The despin sensors are located within the equipment compartment as shown, 37 inches from the c.g. of the spacecraft prior to deployment. The CEA is located within the equipment compartment as shown.

Although not part of the ACS subsystem, the thrusters perform an integral function with the ACS and are therefore also located in Figure 3.3-1. The Velocity-Precession Thrusters (VPT) all have their line of action parallel to the z axis. The thrusts of VPT 1 & 3 face in the +z direction while VPT 2 and 4 face in the -z direction. The line of

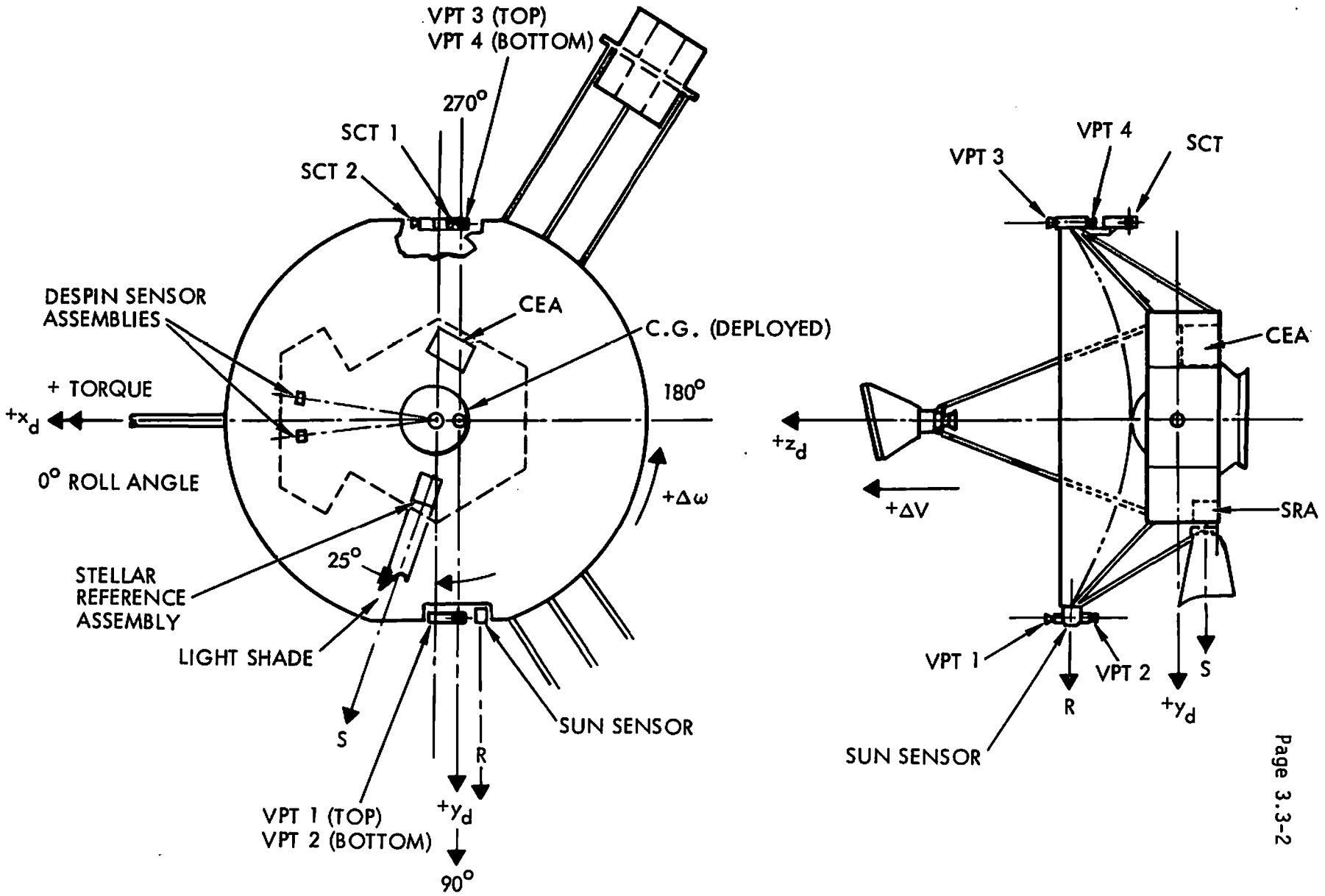


FIGURE 3.3-1 ACS COORDINATES, GEOMETRY AND THRUSTER IDENTIFICATION

action in each case lies in the z, y_d plane as nearly as possible where y_d is the y principal axis with the appendages deployed. The spin control thrusters (SCT) are located near thrusters VPT 3 and 4, near the -y axis. The line of action of the SCTs is parallel to the x axis.

3.3.1.2 ACS Functional Characteristics

3.3.1.2.1 Definition of Functions. The Attitude Control Subsystem (in conjunction with the Propulsion Subsystem) performs six basic functions.

- Despin the spacecraft from its high speed-stowed condition to an intermediate speed sufficient to permit RTG and magnetometer deployment.
- Control the spin speed after deployment in response to ground commands.
- Provide closed loop precession pointing of the spin axis toward earth, using pulses from the Conscan Signal Processor of the Communication Subsystem.
- Provide midcourse velocity (ΔV) control of the duration and direction of the midcourse corrections using either programmed or real time commands.
- Provide roll reference signals for referencing the scientific experiments and spacecraft subsystems.
- Provide telemetry of ACS functions indicative of unit status.

The processing of information is done within the CEA (from sensors, ground commands, and ground data) to perform the functions of the ACS subsystem. The CEA is a single assembly (partially redundant) and is comprised of four subassemblies which are:

- Program Storage and Execution Subassembly (PSE). The PSE contains the circuitry for storing and executing, upon command, the open loop precession and velocity programs. The PSE also provides the major telemetry interface between the ACS, and the Digital Telemetry Unit.
- Sensor and Power Control Subassembly (SPC). The SPC processes the SRA and SSA outputs, provides logic for selecting the desired reference sensor and switches power to the PSE and DSL subassemblies of the ACS. The SPC also provides timing control and clock references to the CEA.
- Two Duration and Steering Logic Subassemblies (DSL). Two identical and redundant DSLs (A and B) are employed in the CEA. Each DSL has complete capability, when appropriately commanded, of metering out the proper pulse length and steering the pulse to the appropriate valve drivers for executing any of the required maneuvers.

The detailed functions performed by the CEA in providing the desired operations are described below. There are 14 functions implemented in the CEA. These functions are in the subassemblies indicated:

1.	Power & Clock Switching for the CEA	Sensor and Power Control (SPC)
2.	Octant Generation and Timing Derivation	
3.	Star Gating	
4.	Star Delay	
5.	Sensor Selection	
6.	Thruster Pulse Duration Selection	Duration and Steering Logic (DSL)
7.	Thruster Selection	
8.	Real Time Operations	
9.	Conscan Operation	
10.	Despin Operation	
11.	Program Storage	Program Storage and Execution (PSE)
12.	Program Loading	
13.	Program Execution	
14.	Telemetry	

3.3.1.2.2 Modes of Operation. There are four modes of spacecraft control: precession, ΔV, spin, and despin. In each mode, there are a number of ways to provide the stimulus to initiate a particular operation.

- Precession. A precession can be accomplished four different ways: Fixed-angle, Conscan, Real-time, and Programmed.

  Fixed-Angle. A precession pulse is initiated at one of four fixed angles: 0°, 90°, 180°, or 270°, following each command from the ground.

  Conscan. The Conscan mode is a closed loop operation. Following a start closed loop ground command, a precession pair will fire each time a conscan pulse is received and as long as an enable signal is present. Pulses normally will occur each third revolution of the spacecraft.

  Real-Time. Real-time firing is initiated by ground command and fires a pre-selected valve pair for a pre-selected duration each time a command is received.

  Programmed Precession. A precession pulse is generated once per revolution at a programmed angle for a programmed period of time. A programmed delay period occurs before the precession or before a programmed ΔV.

- ΔV. There are three ways of achieving a ΔV maneuver:

  Programmed ΔV. Programmed ΔV is a continuous valve firing requiring signals ΔV ON No. 1 and ΔV ON No. 2 both to be true. Firing is terminated when one or the other or both

lines go false. The two ΔV ON signals come from the primary and redundant counters in the PSE.

Real-Time ΔV. Real-time firing is initiated by ground command and fires a pre-selected valve pair for a preselected duration each time a real-time command is received.

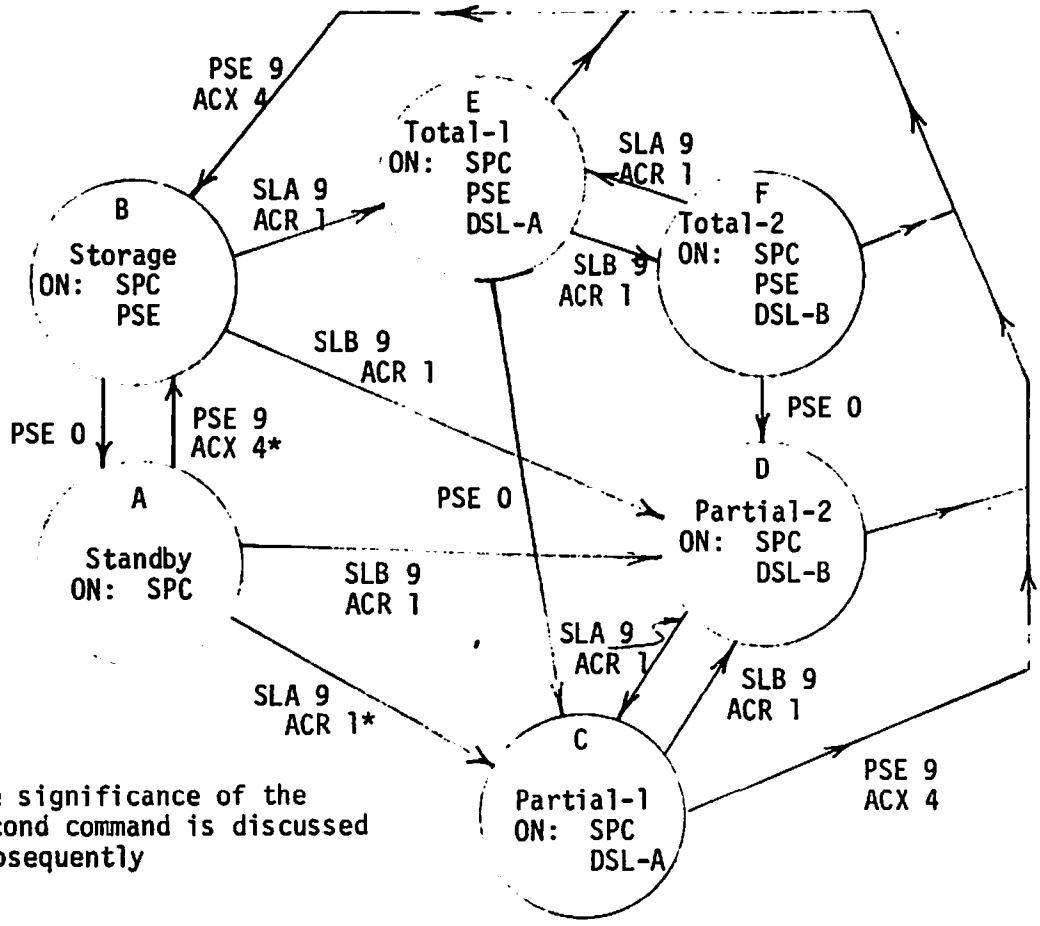
ΔV/SCT Maneuver. A ΔV/SCT maneuver is enabled by ground command VEL 4. Command VEL 4 enables the spin selection such that the first ΔV/SCT pulse (0°) will fire SCT No. 1 and the second pulse (180°) will fire SCT No. 2 and will continue to alternate as long as V/SCT stored program pulses are present. (see Section 4.5.2)

- Spin. Spin control command SPN 3 enables the selected pulsewidth to fire SCT 1 which produces an increase in the spin speed while SPN 4 decreases spin by firing SCT 2.

- Despin. Despin is initiated either by ground command (SPN 1) or by the sequencer in the DTU (see Section 3.5). The despin stop signal is produced by two redundant despin sensor assemblies (DSA No. 1 and DSA No. 2). This signal is on during the despin operation when the spin speed is reduced to about 21 rpm. The signal goes off when the spin speed drops below 21 rpm.

3.3.1.2.3 Power Modes. There are 6 power mode configurations for the CEA. Power is supplied to the SPC and the telemetry (located within the PSE) at all times and in all modes. The SPC and the CEA telemetry cannot be turned off in flight. The power modes are shown in Figure 3.3-2. The standby mode is the minimum power consumption mode normally used during cruise. The PSE and one of the DSLs can be switched either separately or together as indicated. If the PSE only is on, the mode is the storage mode. Programs to be executed may be stored, examined and corrected in this mode without the DSLs. If the stored program is to be executed, one of the DSLs must be turned on (mode total #1 or #2). One DSL can be turned on without the PSE for operation of thrusters in operating modes other than those requiring stored programs.

3.3.1.2.4 ACS Thermal Requirements. The normal operating temperatures and expected temperature ranges for the ACS equipment are given in Table 3.2-2 (Thermal Subsystem).



*The significance of the second command is discussed subsequently

Desired Mode	Subassemblies Powered	5V DC Max Power	28V DC Max Power
Standby (Mode A)	SPC PSE - TLM	1.2 watts	0.3 watts
Storage (Mode B)	SPC PSE	2.8	0.3
Partial-1 (Mode C)	SPC PSE - TLM DSL - A	1.9	0.42
Partial-2 (Mode D)	SPC PSE - TLM DSL - B	1.9	0.42
Total-1 (Mode E)	SPC PSE DSL - A	3.5	0.42
Total-2 (Mode F)	SPC PSE DSL - B	3.5	0.42

Figure 3.3-2. ACS POWER CONTROL

3.3.1.3 Program Storage and Execution Subassembly Description

3.3.1.3.1 General Description. The PSE provides circuitry for the storage and execution of spacecraft precession, velocity, and spin maneuvers. Program storage is accomplished with three sets of registers:

No. 1 Precession Angle Register (9 Bit) No. 1 Precession Redundant Magnitude Register (8 Bit) No. 1 Precession Primary Magnitude Register (11 Bit)	No. 1 Precession Registers (Total 28 Bits)
Delay Register (7 Bit) ΔV Redundant Magnitude Register (8 Bit) ΔV Primary Magnitude Register (13 Bit)	ΔV Registers (Total 28 Bits)
No. 2 Precession Angle Register (9 Bit) No. 2 Precession Redundant Magnitude Register (8 Bit) No. 2 Precession Primary Magnitude Register (11 Bit)	No. 2 Precession Registers Total 28 Bits)

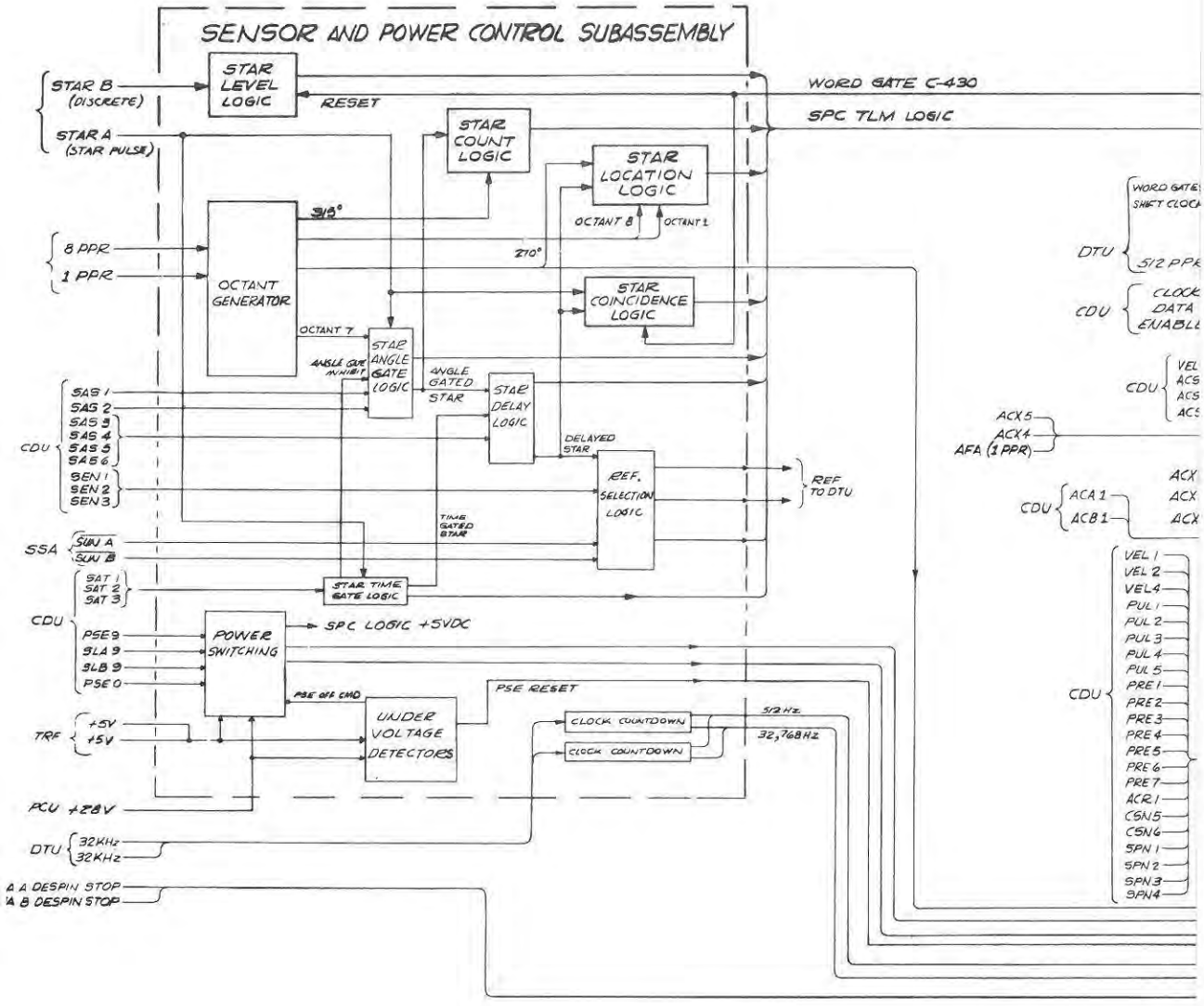
These registers are shown in the CEA block diagram, Figure 3.3-3, which consists of the block diagrams of the 3 subassemblies of the CEA, the PSE (center) the SPC (left hand side) and a typical DSL.

Upon sequence initiation, the contents of the above registers are transferred into their respective counters (angle counter, delay counter primary magnitude counter, and redundant magnitude counter) in accordance with the occurrence of the following PSE program sequence states.

Sequence State	Function	Command
S0	Reset	ACX4
S1	Delay	ACX1
S2	No. 1 Precession	
S3	Delay	
S4	Midcourse ΔV	
S5	Delay	
S6	No. 2 Precession	
S7	Program Complete	

All stored programs cycle through all eight sequence states unless interrupted by ground command. The data transfer in these states is depicted in Figure 3.3-4

Upon commanded initiation of the sequence (ACX1), the program enters state S1 from state zero. At this time, the delay word is transferred serially into the primary magnitude counter and also circulated within the delay storage register. The counter down-counts the delay word at a 1/64 Hz rate, until its magnitude is zero. The sequence then automatically advances to state S2. At this time, the No. 1 primary and redundant magnitude words are transferred into their respective counters. If the program is inhibited (command ACX2), two passes of serial TLM are obtained and data recirculation from the primary counter to the primary magnitude register is inhibited


SENSOR AND POWER CONTROL SUBASSEMBLY
STAR B (DISCRETE)
STAR A (STAR PULSE)
STAR LEVEL LOGIC
RESET
WORD GATE C-430
SPC TLM LOGIC
STAR COUNT LOGIC
STAR LOCATION LOGIC
OCTANT 8
OCTANT 1
315°
270°
8 PPR
1 PPR
OCTANT GENERATOR
STAR COINCIDENCE LOGIC
OCTANT 7
STAR ANGLE GATE LOGIC
ANGLE GATE INHIBIT
ANGLE GATED STAR
STAR DELAY LOGIC
DELAYED STAR
SAS 1
SAS 2
SAS 3
SAS 4
SAS 5
SAS 6
CDU
SEN 1
SEN 2
SEN 3
REF. SELECTION LOGIC
REF TO DTU
TIME GATED STAR
SSA
SUN A
SUN B
SAT 1
SAT 2
SAT 3
STAR TIME GATE LOGIC
POWER SWITCHING
SPC LOGIC +5VDC
PSE 3
SLA 3
SLB 3
PSE 0
PSE OFF CMD
PSE RESET
UNDER VOLTAGE DETECTORS
CLOCK COUNTDOWN
CLOCK COUNTDOWN
5/2 Hz
32,768 Hz
TRF
+5V
+5V
PCU +28V
DTU
32KHz
32KHz
A A DESPIN STOP
A B DESPIN STOP
DTU
5/2 PPR
CDU
CLOCK
DATA
ENABLE
CDU
VEL
ACS
ACS
ACS
ACX 5
ACX 4
AFA (1 PPR)
CDU
ACA 1
ACB 1
ACX
ACX
ACX
CDU
VEL 1
VEL 2
VEL 4
PUL 1
PUL 2
PUL 3
PUL 4
PUL 5
PRE 1
PRE 2
PRE 3
PRE 4
PRE 5
PRE 6
PRE 7
ACR 1
CSN 5
CSN 6
SPN 1
SPN 2
SPN 3
SPN 4

FIGUR

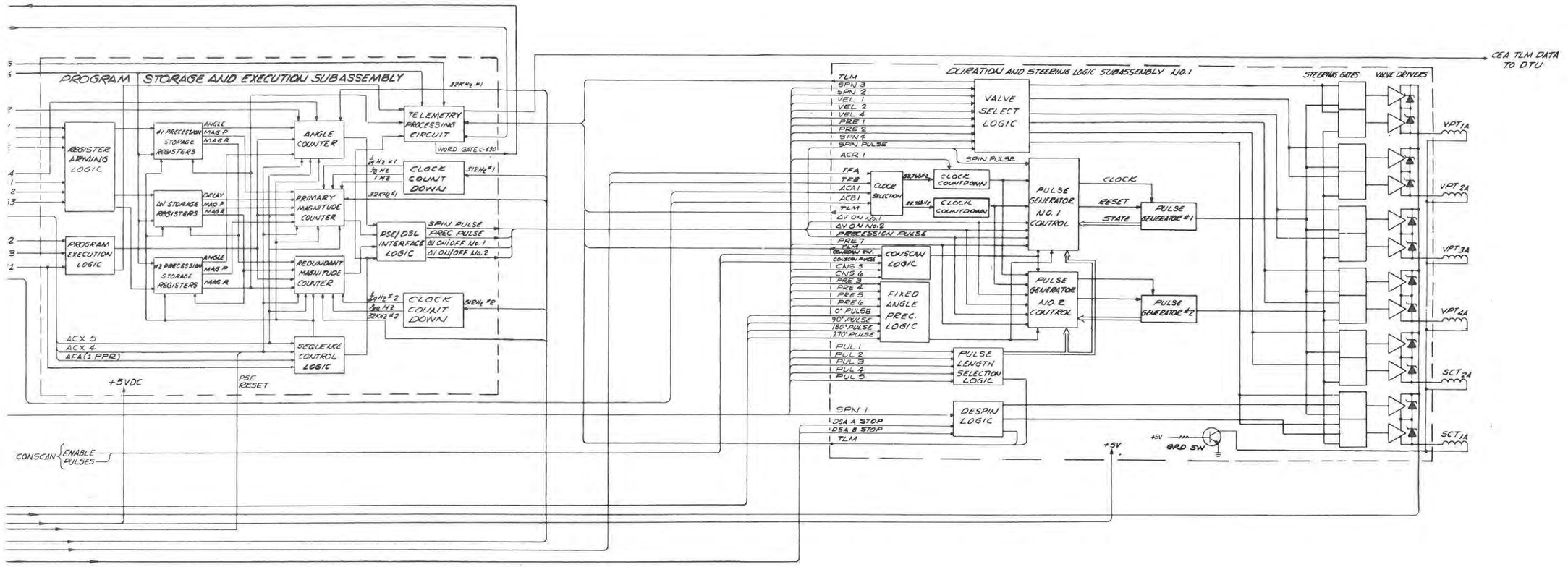


FIGURE 3.3-3 PIONEER F/G CEA BLOCK DIAGRAM

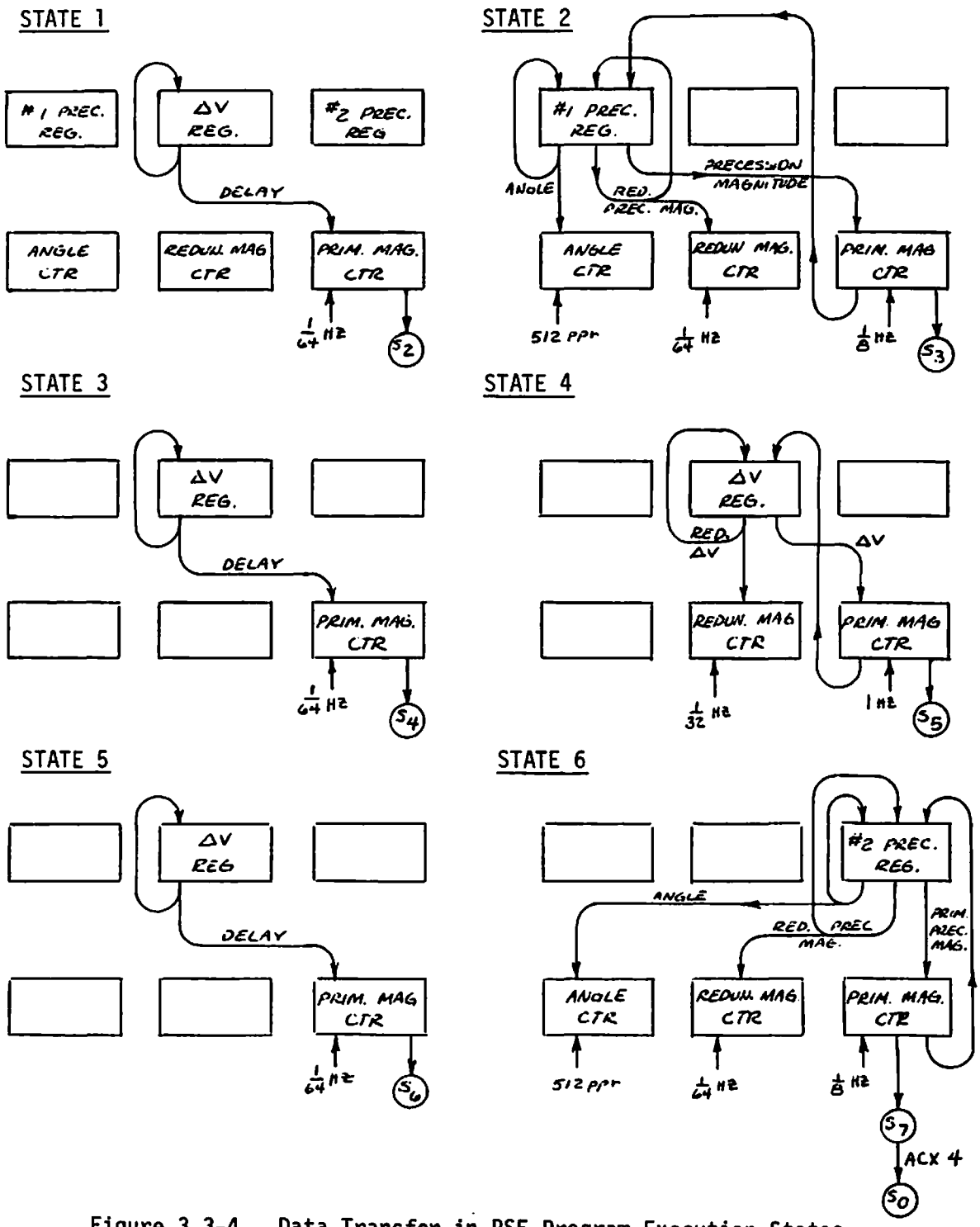


Figure 3.3-4. Data Transfer in PSE Program Execution States

except during the initial data transfer. During transfer, the redundant magnitude word is also circulated within the redundant magnitude storage register while the primary magnitude storage register receives data feedback from its counter. The counters then down-count the primary magnitude word at a 1/8 Hz rate and the redundant magnitude word at a 1/64 Hz rate. The first occurrence of the AFA (1 ppr) pulse following entry into sequence state S2, causes the No. 1 angle word to be transferred into the angle counter and also circulated within the No. 1 angle register. The counter then down-counts the angle word at a 512 ppr rate until its magnitude is zero. Upon completion of the angle word countdown, a pulse is issued from the PSE to the Duration and Steering Logic (DSL) for initiation of precession thruster firing. The angle transfer and countdown is performed on each occurrence of the 1 ppr pulse from the SPSG (each revolution of the spacecraft) for the duration of the sequence in state S2. Upon completion of the No. 1 primary precession magnitude word countdown, the sequence automatically advances to state S3. The delay word is again transferred to its counter and down-counted as described previously for sequence state S1. Completion of delay word countdown advances the sequence to state S4. Sequence entry into state S4 causes the primary and redundant ΔV words to be transferred into their respective counters as previously described for the No. 1 precession magnitude words. The counters then down-count the primary magnitude word at a 1 Hz rate and the redundant magnitude word at a 1/32 Hz rate. Sequence entry into state S4 causes ΔV thruster firing commands (redundant) to be sent from the PSE to the DSL. These commands remain active for the duration of the sequence in state S4. Upon completion of the ΔV primary magnitude word countdown, the sequence advances to state S5. The delay word is again transferred to its counter and down-counted as described for sequence state S1. Completion of the delay word countdown advances the sequence to state S6. The No. 2 precession primary magnitude, redundant magnitude, and angle words are transferred to their respective counters and down-counted as previously described for the No. 1 precession mode. Completion of the No. 2 precession primary magnitude word countdown advances the sequence to state S7 (TLM = 000), sequence complete. The sequence remains in this state until commanded to reset (command ACX 4) and returned to state S0. The sequence block diagram, Figure 3.3-4, shows the data transfer in each state.

Several commands are utilized in initiating and controlling the PSE program sequence. Commands ACS1, ACS2, ACS3 permit the entry of program data into the PSE storage registers. (Data will be in the next 4 words following the arm command). Command ACX1 initiates the execution of the program from state S0. Command ACX2 permits interruption (inhibiting) of the program sequence in any sequence state and command ACX3 restarts the sequence at the point where interrupted by ACX2. Command ACX5 causes the sequence to advance one state from that existing at the time of command receipt. Command VEL4 changes the No. 1 precession maneuver into the ΔV/SCT maneuver. Command ACX4 resets the sequence to state S0. Finally, if a master reset is received from the Sensor and Power Control Sub-assembly (SPC) indicating an undervoltage condition, the program will return to state 0. These commands have a number of operational restrictions listed below:

1) Never send sequence execute twice in row--(ACX 1). If this happens, program will end up in random state and will not step properly. ACX 1 should be preceded by ACX 4, sequence reset, except in ΔV/SCT mode.

2) Commands ACX 4 (sequence reset) and ACX 2 (sequence inhibit) will cause two passes of the serial telemetry to be read out. Commands which operate the PSE sequence or load data should not be sent during this interval. (See Section 3.3.1.3.2--Telemetry processing circuitry, Page3.3-16.

3) If the PSE is initially off, the PSE-on command (PSE 9) should be followed by sequence reset (ACX 4). If this is not done, a single thruster pulse might be fired when the DSL is turned on. The DSL will come on in a random state and should be cleared by clock reset, (ACR 1).

4) Sequence execute (ACX 1) must not be used as a sequence override once the program has been started or inhibited. The PSE will switch to random state and not step properly.

5) At the end of the program, the sequence state goes to state 7, an apparent but not a true state 0 in that the sequence state telemetry reads 000. To return to the zero state either sequence step (ACX 5) or sequence reset (ACX 4) may be used. In the ΔV/SCT mode sequence reset clears this mode while sequence step does not. To move out of state 0, sequence execute (ACX 1) must be used.

6) If the redundant magnitude registers run out before the primary registers, telemetry will read 000 for sequence status even though the state has not changed. TLM is restored at the end of the state in which this occurs.

7) Sequence reset (ACX 4) or sequence step (ACX 5) may be used at any time, although sequence step cannot be used to step from state 0 to state 1. If sequence step is used to step out of states 2, 4, or 6, the program remaining will be executed if the sequence is started a second time.

The PSE processes and sends to the DTU all digital telemetry from the Control Electronics Assembly (CEA). All CEA telemetry, except the PSE delay word, No. 1 and No. 2 angle words, No. 1 and No. 2 precession redundant magnitude words, and the ΔV redundant magnitude word, is available for readout to the DTU at any time. The above mentioned PSE words are available only when the PSE is commanded in the reset or inhibit states and storage register loading has been completed. Two complete passes of PSE serial telemetry readout are automatically obtained after commands either ACX 2 or ACX 4.

The PSE operates with switched and unswitched +5 vdc power from the SPC. The unswitched power is supplied to the telemetry processing portions of the PSE. The switched power supplies all other PSE circuits. Switched power is automatically removed (by the SPC) from the PSE upon detection of supply voltage less than +3.5v $\pm$ 0.5 vdc.

3.3.1.3.2 Block by Block Description. In the following paragraphs, each of the blocks shown in the PSE subassembly (Figure 3.3-3) are briefly described. The paragraphs include the register arming logic, the program storage registers, program counters, sequence control and execution logic, PSE/DSL interface logic, telemetry processing, and the clock countdown.

Register Arming Logic. The register arming logic provides circuitry for steering the input data and data clock to the proper storage register. Commands ACS 1, ACS 2, and ACS 3 permit the loading of the No. 1 precession registers, the ΔV registers, and the No. 2 precession registers, respectively. The receipt of each of these commands sets a corresponding latch (load enable) and clears (inhibits loading) the other two. Thus, the accidental loading of more than one register at a time is prevented. The register arming logic is functionally shown in Figure 3.3-5. The register arming logic also provides circuitry for inhibiting the entry of the first bit of each eight bit input data word. This feature is required as an operational safeguard. The register arming logic provides automatic disarming (inhibits loading) of the selected register after the receipt of four 8 bit data message words. The disarming also occurs at the receipt of command ACX 4 (sequence reset) from the CDU or a MR (master reset) signal from the undervoltage detector circuit in the SPC.

Program Storage Registers. The important parameters for the nine program storage registers described in Paragraph 3.3.1.3.1 are shown in Table 3.3-1 and Figure 3.3-5. There are three registers in a group with each group loaded via a single enable command. Table 3.3-1 below shows the maximum values that can be stored, the quantization, the number of bits and the enable commands.

Table 3.3-1. Program Storage Register Parameters

REGISTER GROUP	ENABLE	PROGRAM WORD	MAX. VALUE	QUANTIZATION	NO. BITS
#1 Precession Registers	ACS 1	Angle 1	360 deg	0.703125°	9
		Prec. Magnitude-1 Redun.	16,320 sec	64.0 sec.	8
		Prec. Magnitude-1 Prim.	16,376 sec	8.0 sec.	11
ΔV Registers	ACS 2	Delay	8,128 sec	64 sec.	7
		ΔV Magnitude-Redundant	8,160 sec	32.0 sec.	8
		ΔV Magnitude-Primary	8,191 sec	1.0 sec.	13
#2 Precession Registers	ACS 3	Angle 2	360 deg	0.703125°	9
		Prec. Magnitude-2 Redun.	16,320 sec	64.0 sec.	8
		Prec. Magnitude-2 Prim.	16,376 sec	8.0 sec.	11

WARNING: Order for sending data: prime magnitude, redundant magnitude, angle; least significant bit first

Figure 3.3-5 shows the stored program format, the loading sequence and the enable logic.

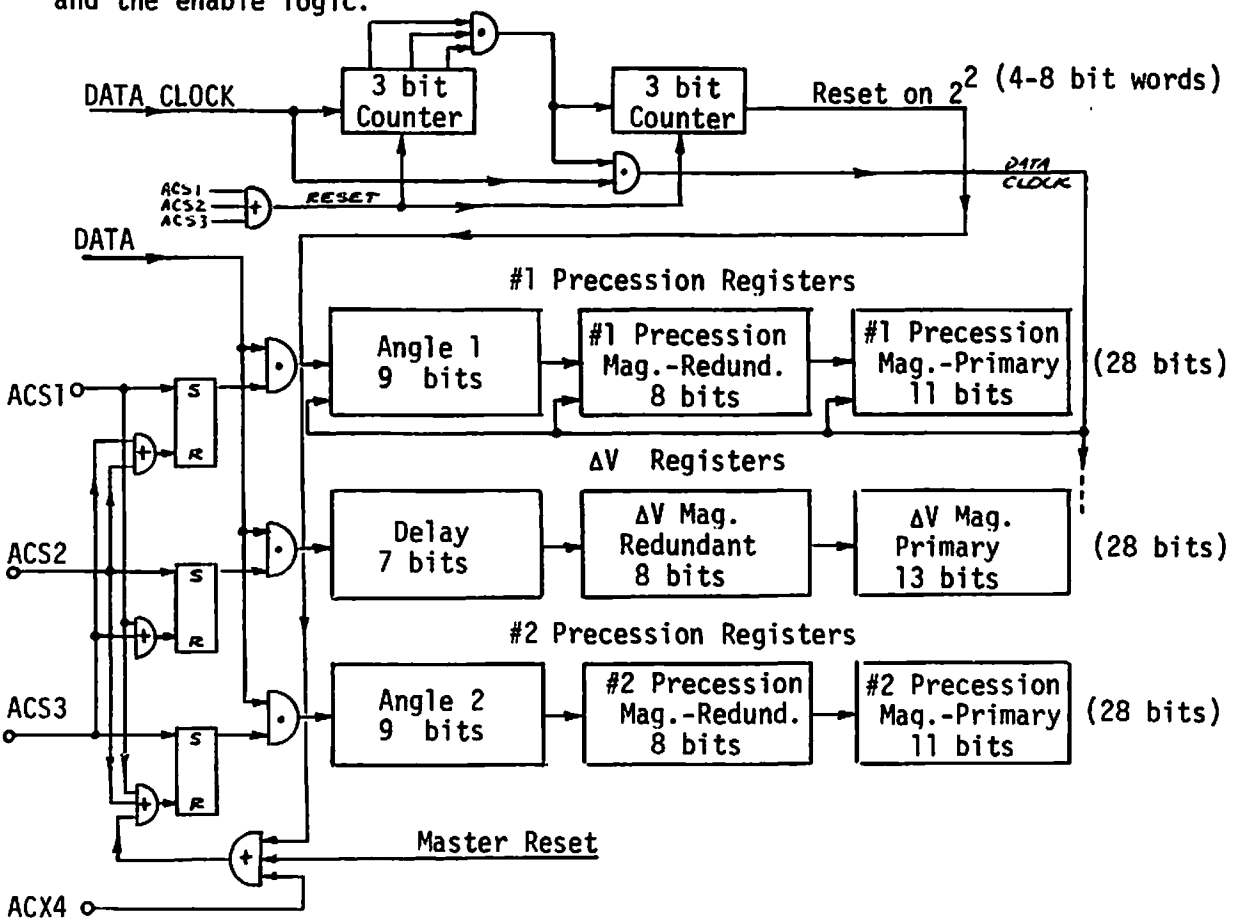


Figure 3.3-5. Program Storage Registers and Enable Logic

As shown in Figure 3.3-5 the input data is loaded serially from left to right. In the case of the precession registers the first 11 bits have as their destination the primary magnitude register, the next 8 bits the redundant magnitude register, the third group of 9 bits the angle register. Similarly, the ΔV register bits flow in sequence: the primary magnitude, redundant magnitude and delay register contents.

The above groups of 28 bits are composed of four words of 7 bits. The PSE receives four words of 8 bits each (total 32 bits), the first bit of which is stripped off.

Data Signal. The data signal produces coded binary inputs with the least significant bit occurring first. The data signal goes to high state for a "1" and to the low state for a "0." The pulse is shown in Figure 3.3-6. The first data bit of each serial data word input is always a "zero." The data is clocked into the registers at one bit per sec.

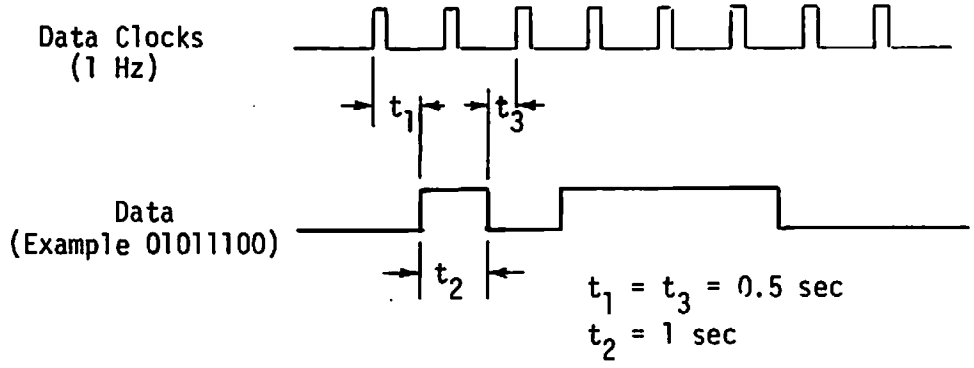


Figure 3.3-6. CDU Data Input Timing

Counters. The contents of the program register are transferred to three counters: The primary magnitude counter, the redundant magnitude counter and the angle counter. These counters obtain data from the program storage registers as described in Paragraph 3.3.1.3.1 and shown in Figure 3.3-4.

Following entry of the sequence into the No. 1 or No. 2 precession modes (States 2 and 6), the data from the angle register is clocked into a holding register when the 1 ppr (AFA) pulse occurs. One bit is subtracted from the value in the holding register via the subtract logic for each 512 ppr (AFC) pulse. This occurs by circulation of the data out of the holding register, through the subtract logic and back. When all zeros are detected in the data transfered back to the holding register, a pulse is issued to the DSL to fire the appropriate thruster. The angle counter is then dormant until the next AFA pulse. In the ΔV/SCT mode, the bit corresponding to value 256 is set to 1 after the pulse to the DSL is issued. This angle (180°) is then down-counted as before and a second pulse is issued. Angle countdown continues once per revolution of the spacecraft until the primary precession magnitude word countdown is complete at which time the sequence advances to the next state.

The primary magnitude counter processes the delay word in the states S1, S3, and S5, the precession primary magnitude word during S2 and S6, and the ΔV primary magnitude word during S4. The operation of the primary magnitude counter is essentially the same as the angle counter in that data is transferred to a holding register and recirculated through subtract

logic. For the primary counter, the data is circulated back into the storage register such that the stored value is decremented and indicated by telemetry. The redundant magnitude counter processes the redundant precession magnitude and ΔV words. It is essentially identical in operation to the primary and angle counters.

Program Execution and Sequence Control Logic. The program execution logic is controlled by a single (redundant) latch. This latch is set by the command to sequence execute (ACX 1) when in state 0 or by sequence override (ACX 3) when in states 1-6. The latch is reset by commands ACX 2 (sequence inhibit), ACX 4 (sequence reset) or by a master reset from the undervoltage detector. The program execute latch, when set, enables the clocks for the angle countdown, the delay countdown, and the primary and redundant magnitude countdown. It also inhibits the serial telemetry readout.

The sequence control logic steps the program through the states S0 through S6 following the sequence execute command. The basic element of the sequence control logic is a 6 flip-flop shift register. Each flip-flop is used to enable one of the states one to six. The sequence execute command causes the first flip-flop to go high, thus enabling state one. Either a sequence step command (ACX 5) or a completion signal from the primary magnitude counter (end of state) will clock the shift register such that the high state of one flip-flop goes low and the next goes high, thus enabling the next state.

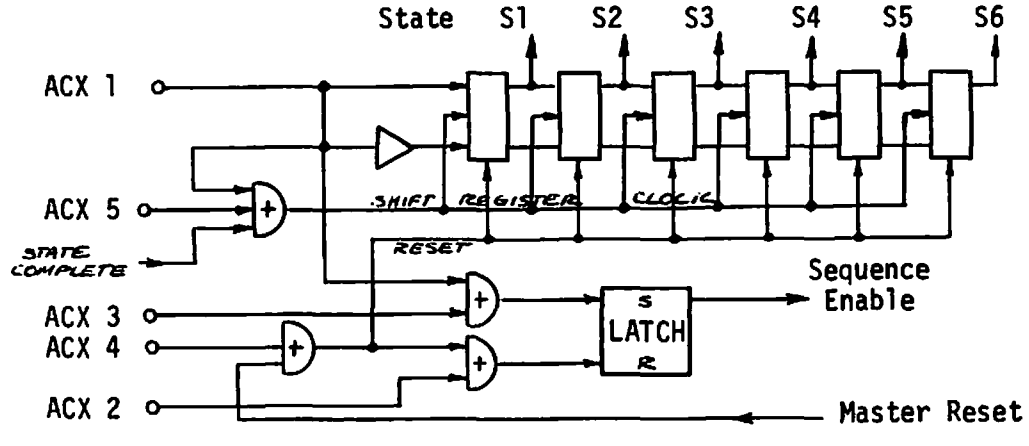


Figure 3.3-7. Program Execution and Sequence Control Logic

PSE/DSL Interface Logic. The PSE/DSL interface logic decodes the outputs from the counters and the sequence control logic to provide precession, spin and ΔV "commands" to the DSL. The precession and spin outputs are pulse signals which initiate the DSL "one-shots" (Section 3.3.1.4.2). These "one-shots" produce valve firings for a preselected time interval. The ΔV output to the DSL is a real time signal which causes the DSL to fire the ΔV thrusters for the duration of the command. All PSE outputs are redundant to the extent that no single output can permanently fail both DSLs if power is removed from the PSE before the second DSL is turned on. Separate lines are used for each DSL.

With the PSE not in the ΔV/SCT mode, a precession pulse is sent to the DSL each time an angle countdown completion occurs for the duration of both the first and second precession sequence states. Two (redundant) ΔV signals (duration determined by the programmed ΔV word magnitude), ΔV On #1 and ΔV On #2, are sent to the DSL during the midcourse ΔV sequence state. Both signals must be true to maintain ΔV.

When the PSE is in the ΔV/SCT mode, a spin pulse is sent to the DSL each time an angle transfer and countdown occurs (in state 2), with a second pulse sent 256 counts (180°) later. This continues until the primary (or redundant) counter has been decremented to zero, or the program is interrupted by command. The PSE outputs in all other states (1, 3, 4, 5, and 6) are unchanged from those that occur during a non-ΔV/SCT precession mode. Thus the angle and precession magnitudes stored in the #2 precession storage registers will be executed as an ordinary precession in the ΔV/SCT mode.

Telemetry Processing Circuitry. The PSE processes all CEA digital telemetry signals. Two types of telemetry interrogration occur within the PSE, "parallel" and "serial" described in the following paragraphs. For all the CEA telemetry, the DTU provides a continuous telemetry clock and twenty-three separate enables which are synchronized to the clock and of duration equal to six clock cycles. All telemetry data from the CEA to the DTU is transferred serially.

The "parallel" telemetry processing circuitry consists of a six bit shift register, strobe generation logic, gating for parallel loading of data into the register, and gating for enabling and steering of the telemetry clock. All CEA telemetry information (except the PSE angle, delay, and redundant magnitude words) is parallel loaded, six bits at a time, into the telemetry register and shifted serially to the DTU. This operation utilizes the first thirteen telemetry word enables from the DTU. The contents of the three primary magnitude registers of the PSE (which are loaded in parallel into the telemetry register) are available for telemetry at any time. However, if the telemetry interrogation occurs while these registers are transferring data to the counters, or in the act of being decremented, then ~~all bits of the telemetry will be clobbered~~ (shifted from the normal position by an indeterminate number).

This effect will be observed periodically, particularly at low telemetry bit rates.

For all "serial" telemetry, the angle, delay, and redundant magnitude information is shifted serially directly from the input storage registers to the DTU. This data is simultaneously circulated back into these storage registers to prevent its destruction by the telemetry interrogation. To accomplish this 49 bit data circulation, an additional five bit register is required to make the total shifts required for one complete circulation a multiple of six (9 TLM words). The angle, delay and redundant registers can not be interrogated by telemetry during operation of the program. This data is only read out on command. Upon receipt of commands ACX 2 or ACX 4, the program will be interrupted (inhibited or reset) and two cycles of telemetry from these registers will be read out. No sequence execute (ACX 1) sequence override (ACX 3) or register arming commands (ACS 1-3) should be sent while this telemetry is being read. These commands must be inhibited for a period of time determined by the TLM format and bit rate. Typical delay values are given below for the C-4 format. It is assumed that the commands are not timed with the telemetry interrogation such that the delay must last for 3 TLM passes.

Bit Rate (bits/sec)	Required Delay (sec)
2048	0.28125
1024	0.5625
512	1.125
256	2.25
128	4.5
64	9.0
32	18.0
16	36.0

Clock Countdown. The clock countdown circuits consist of two binary counters. The SPC provides the PSE with two 32,768 Hz clocks (A and B) and two 512 Hz clocks (No. 1 and No. 2 divided by the SPC from the 32,768 Hz clock B and A, respectively). The PSE divides the 512 Hz No. 1 to obtain 1 Hz, 1/8 Hz, and 1/64 Hz clocks for controlling the countdown rate of the primary ΔV magnitude word, the precession magnitude words, and the delay word, respectively. Also the 32,768 Hz clock B is used as the shift clock for the angle and primary magnitude counters. The 512 Hz No. 2 is divided to obtain 1/32 Hz and 1/64 Hz clocks for controlling the countdown rate of the redundant ΔV and precession magnitude words, respectively. The 32,768 Hz clock A is used as the shift clock for the redundant magnitude counter. The PSE can operate without clock A, but not without clock B. If clock A fails, only the redundant portion of the PSE is lost.

Loading of Data into the PSE. Although data is normally loaded with the PSE in state 0, it can be performed in any state if the sequence is inhibited. The program can be continued using its new data. As an example of the data input, assume that the program described in Section 4.4 has generated the following data to be entered into the PSE.

	Nominal Value	Quantized Value	Number of Bits
1st Precession Angle (α_R) $\alpha 1$	140.52°	140.625°	200
1st Precession Magnitude (ψ)	61.6°	220 steps	344 (2752 sec)
V Magnitude	37 m/sec	1027.8 sec	1028
2nd Precession Angle (α_R)*	320.52°	320.625°	456
2nd Precession Magnitude (ψ)	61.6°	220 steps	344

*Same thruster pair should be used for the return precession (see Section 4.1).

The quantized value of angle is selected from the great circle program as the optimum value of α_R and is assumed fully corrected for pulse centroid, etc. The number of bits is determined from the angle register quantization of 0.703125°/bit (i.e., 360°/512 bits). The precession magnitude is also fully corrected for impulse calibration, spin rate, quantization, etc. and is the optimum value calculated for minimum error in final orientation. For this example, it is assumed that the step size is 0.28°, and the spin rate is 4.80 rpm. Total time for the maneuver, rather than the number of steps, is entered in the magnitude register to minimize errors incurred via spin rate change during the maneuver. The nominal time to be entered is 2750 sec. Since the quantization is 8 sec, the nearest bit number is 344 (2752 sec). There is an uncertainty in execution of one bit in this register since spin speed changes may occur during execution.

For the ΔV, a quantization of 0.036 m/sec per second is assumed. For the return precession, the same thruster pair is used as for the 1st precession to eliminate errors due to differences in thrusters. This preferred thruster pair should also be the same pair used for the calibration (earth acquisition) maneuver(s).

The loading of the data is performed as follows. The ACS 1 command arms the 1st precession angle and magnitude registers. The next four words after this command contain the data to be entered. The data bit stream goes in least significant bit first:

	MSB XXXXXXXOX ANGLE	XXXXXOXXX REDUNDANT MAGNITUDE	XXXXOXXXXXXO LSB MAGNITUDE
Max no. of bits	9 bits	8 bits	11 bits
Quantization	512 p/rev	1/64 Hz	1/8 Hz
Maximum value	360°		16,376 sec
Number to be entered**	n=200(140.625°)	*	m=344(2752 sec)
Example	0110010$\bar{0}$00	00101$\bar{0}$100	00101011000$\bar{0}$
	90° 45° 5.625°		2048 sec 512 sec

NOTES: Bits indicated by $\overline{0}$ are entered as zero in the bit stream, but will be stripped from the data before loading into the registers.

*For the redundant magnitude register data, use the primary register number rounded to the next highest number, and add one bit. This extra bit is required since the registers are counted using different, asynchronous clocks.

**A description of the great circle program used to obtain n and m is given in Section 4.4.1.

The ΔV is 1028 seconds and the delay is 1024 sec as described before.

	MSB XXXXXXX$\overline{0}$ DELAY	XXXXXXX$\overline{0}$X REDUNDANT MAGNITUDE	XXXXXX$\overline{0}$XXXXXX$\overline{0}$ LSB MAGNITUDE
Max no. of bits	7 bits	8 bits	13 bits
Count rate	1/64 Hz	1/32 Hz	1 Hz
Maximum value	8128 sec		8192 sec
Number to be entered	n = 16	*	m = 1028
Example	0010000$\overline{0}$	0010001$\overline{0}$0	0010000000100$\overline{0}$
	1024 sec	1024 sec	64 sec 1024 sec 4 sec

Again, the redundant magnitude is calculated by rounding the primary magnitude to the next highest number and adding one bit. Note that the primary register can never be filled to its maximum capacity and be executed accurately since the uncertainty in the redundant register execution of one bit (32 sec) will dominate. This is also true for the precession registers. This effect occurs because the primary and redundant counters are counted down in parallel. The output of both counters is required to keep the sequence operating to prevent gas expulsion in the event of a failure. If the redundant counter is counted to zero before the primary counter, the sequence status telemetry indication will change to 000 (from any state), and the thruster firing will be terminated. The primary counter will continue to downcount, and upon completion of its count, the sequence status telemetry will be restored, and the sequence will be advanced to the next state.

Important restrictions on data loading and execution are given below.

1) The angle registers cannot be loaded with a value (number of counts) greater than 511. If spin speed changes occur during a precession maneuver, the SPSG may put out either 511 or 513 pulses per revolution instead of the normal 512. The last pulse should therefore be considered unusable.

2) The first pulse executed by a program will probably not be of the correct length if the pulse generator (one shot) in the DSL has not fired a pulse. This occurs because the flip-flops in the one shot are not reset until after a pulse is issued to the thrusters. When the DSL comes on, these flip flops may come on in random states such that the first pulse may be shorter than that selected.

3.3.1.3.3 Clock Failure Modes.

Clock Failed	Function Lost	Corrective Action
TFA		
32,768 Hz #1	SPC-Star logic	None
512 Hz #1	SPC-Fixed angle precession	Use real time pulses
	DSL	Switch DSL to TFB
TFB		
32,768 Hz #2	PSE	Switch to fixed angle precession
512 Hz #2	DSL	Switch DSL to TFA
AFA 1 ppr AFC 512 ppr	PSE SPC DSL	Switch SPSG in DTU
CDU		
Data clock	PSE-Data loading	Switch to redundant CDU

3.3.1.4 Duration and Steering Logic Subassembly Description.

3.3.1.4.1 General. The Control Electronics Assembly (CEA) contains two identical Duration and Steering Logic (DSL) subassemblies which provide redundant valve driving capability. Power to the DSLs is supplied by the Sensor and Power Control Subassembly (SPC) and is an "exclusive or," in that power can only be supplied to one DSL at a time. Each DSL drives one of the redundant valve coils such that either DSL, when powered, can drive the valves. The DSL accepts commands and signals from the CDU (Command Distribution Unit), PSE (Program Storage and Execution Unit), the CSP (Conscan Signal Processor), the SPC and the DSA (Despin Sensor Assembly) and provides response to these input by controlling the duration of firing, selection of valve pairs to be fired and finally firing the desired valves.

3.3.1.4.2 Detailed Unit Functional Description. The DSL generates the pulse widths for controlling the duration of the various valve firings, provides valve selection, provides logic to despin the spacecraft continuously until command to stop, counts down the 32,768 Hz input clock frequency to develop four other clock frequencies, and provides logic such that a precession may be made at preselected angles.

To fire a given thruster or thruster pair, 3 signals must be present in the DSL: A valve select enable, a maneuver enable signal, and a pulse duration or thruster on signal. These signals are provided by command to the valve selection logic, by command to the maneuver enable logic, by command to the pulse selection logic, and by signal from the PSE. The despin thruster can be fired independently via the despin logic.

Figure 3.3-3 shows a block diagram of the CEA. The right hand side of this figure shows the DSL in detail. Each of the blocks in this diagram will be explained in detail in the following paragraphs.

Valve Select Logic. The valve select logic consists of command storage latches and decoding gates. The three storage latches are logically interrogated such that one precession pair, one ΔV pair and one spin thruster are always selected. The maneuver enable logic provides the signal which determines precession, ΔV, or spin maneuvers. The valve selection is "anded" with a maneuver enable flip-flop output which is generated by an execute command from either the ground or the PSE. The type of enable is determined by the mode of operation, and since three of the six available valve pairs are continuously selected, a mode will be selected by the type of execute command received.

Clock Selection Logic. The DSL accepts two independent asynchronous 32,768 Hz clocks, TFA and TFB. The clock selection network is used to prevent a failure in one clock from failing both redundant DSLs. By adding two clock select commands to control the clock select logic, a failure in only one clock will not fail either of the two DSLs. By command, clocks No. 1 or No. 2, or clocks No. 1 and No. 2 can be selected to clock the pulse duration logic. Command ACR 1 will result in the normal mode with both

references TFA and TFB used as clock frequencies #1 and #2 respectively. Both clocks can be driven by the same time frequency reference by first sending clock reset (ACR 1) followed by the clock select command (ACA 1 or ACB 1). No automatic cross reset occurs with these commands.

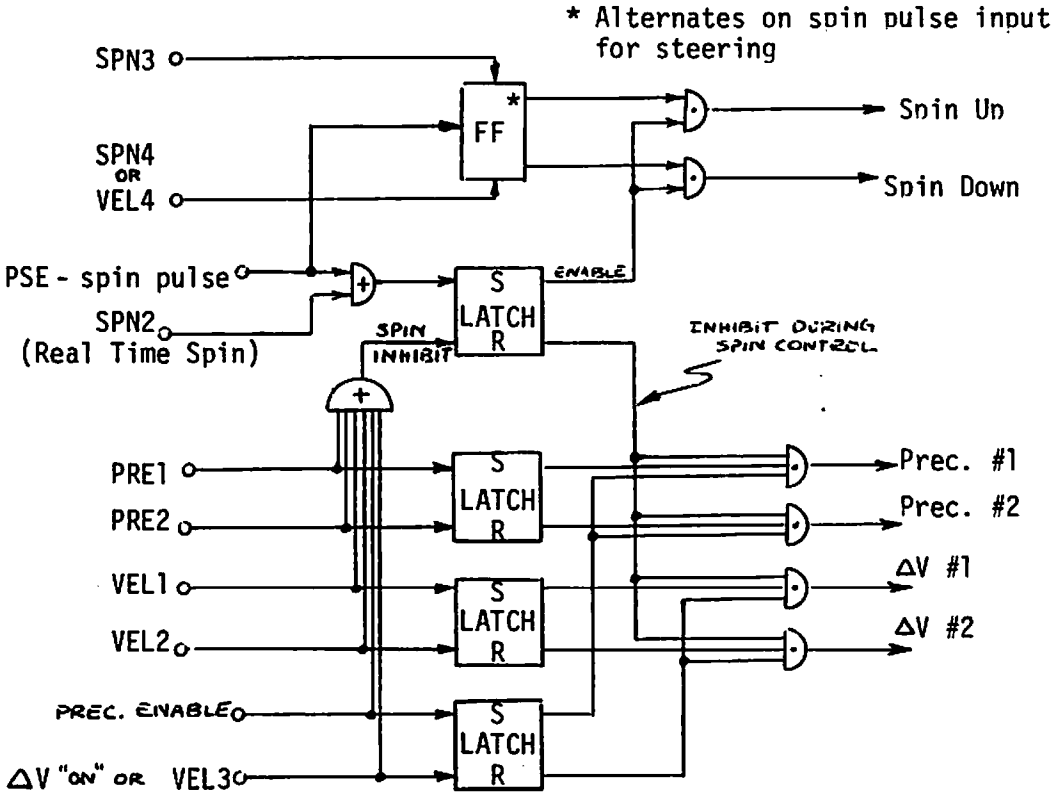


Figure 3.3-8. Valve Select Logic

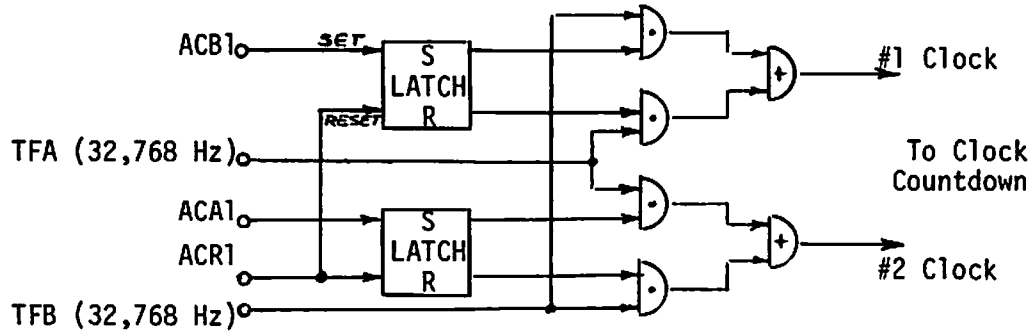
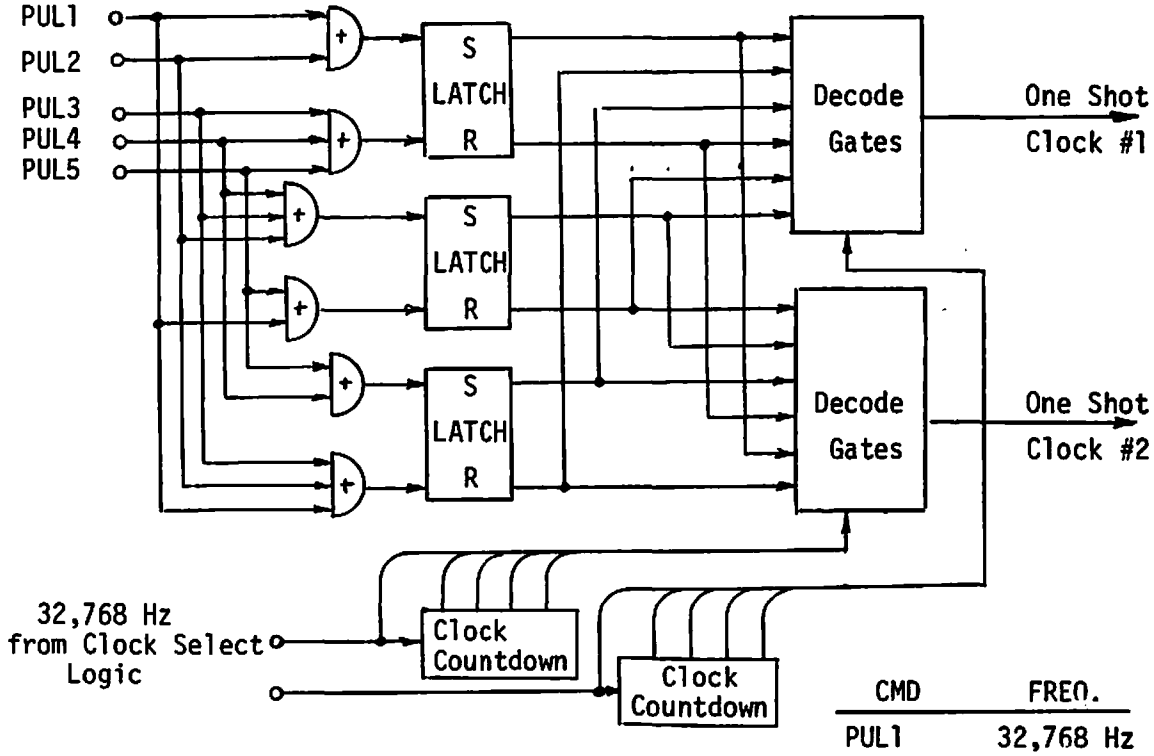


Figure 3.3-9. Clock Selection Logic

Table 3.3-2. Command Switching Logic

Symbol	Command	Result
ACR 1	Clock Reset	TFA feeds clock countdown #1 TFB feeds clock countdown #2
ACA 1	Select Clock A	TFA feeds both countdown chains
ACB 1	Select Clock B	TFB feeds both countdown chains if ACB 1 follows ACR 1

Pulse Length Selection Logic. Ground commands are accepted by the pulse selection logic and determine the state of the three R-S latch circuit switch outputs. These outputs are decoded and "anded" with the different frequencies from clock countdown to supply the various frequencies to the pulse generators for generating the proper firing durations. When DSL power is turned on, the latches in this circuit come on in random states three of which are "forbidden" combinations. It is good practice to select a pulsewidth and a thruster pair after the DSL "on" command.



CMD	FREQ.
PUL1	32,768 Hz
PUL2	8,192
PUL3	2,048
PUL4	1,024
PUL5	512

Figure 3.3-10. Pulse Length Selection

Pulse Generators. To meet the failure mode requirements, it is necessary to have two identical but independent pulse duration generators #1 and #2. (Each pulse generator is also called a "one shot.") Both pulse generators must operate to fire the thrusters so that a single failure will not cause gas expulsion. In normal operation, the pulse generators are driven by clock frequencies which are independent of one another in the clock reset mode. There are five different pulse durations generated by changing the clock frequencies to the generators as described in the previous paragraph on the pulse length selection logic.

Since the one shot start and the one shot clock frequency pulses are asynchronous, there is an uncertainty in the pulse length of one pulse period at whatever frequency is input. (see Figure 3.3-10.) This error is uniformly distributed and should average 1/2 the period for a large number of pulses. Errors in pulse execution are discussed in Section 3.3.1.4.3 and are summarized in Table 3.3-3.

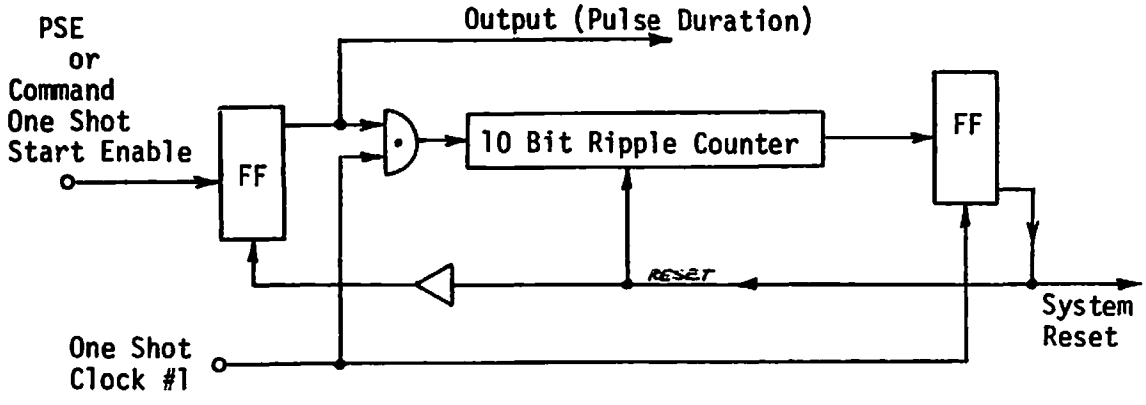


Figure 3.3-11. Typical Pulse Generator (One Shot #1)

The logic diagram in Figure 3.3-11 shows a typical pulse generator. The one shot start enable sets the flip-flop and the 10 bit counter starts counting the clock pulses from the pulse length selection logic. After count 1024, the second flip-flop is set which resets the enable flip-flop and the counter. The output which is taken from the enable flip-flop lasts 1024 clock pulses. The second flip-flop is reset by the next clock pulse.

Fixed Angle Precession Logic. The fixed angle logic accepts ground commands to select a desired roll angle at which a precession pulse shall be fired.

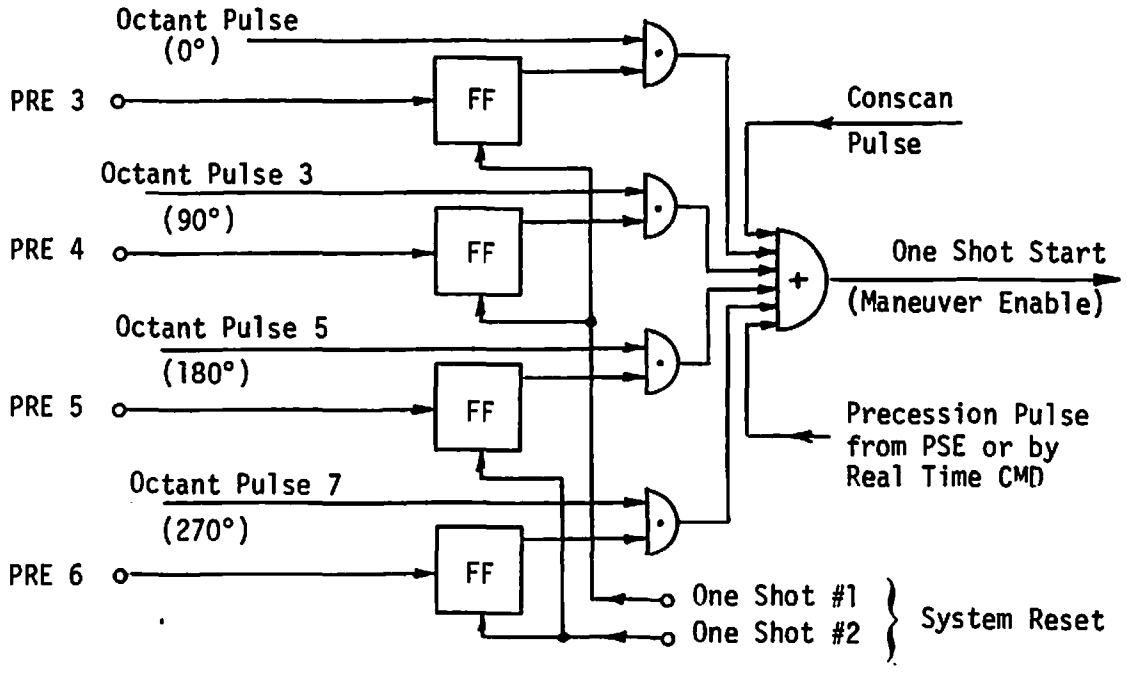


Figure 3.3-12. Fixed Angle Precession Logic

Four flip-flops receive the fixed angle precession commands and gate their respective octant pulses to the one shot. Two system reset lines from the pulse generators are used to delete the commands.

Conscan Logic. The conscan logic consists of a control flip-flop whose output when set is "anded" with the conscan threshold enable and the CSP execute pulse train to fire the selected precession valve pair. The control flip-flop is set to enable conscan pulses by command CNS 5 and reset, disabling pulses by CNS 6. If the conscan threshold indicates (even momentarily) that the threshold has been reached, this drop of the threshold signal will clock (reset) the control flip flop and disable conscan. If conscan is to be continued, it must be re-enabled by command CNS 5.

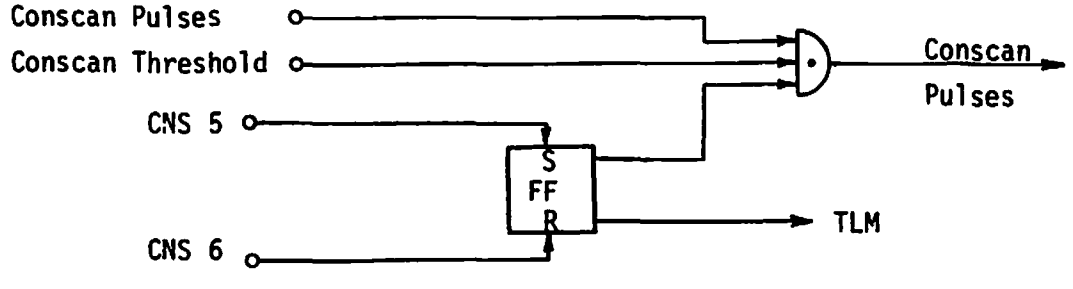


Figure 3.3-13. Conscan Logic

Valve Drivers. Each valve driver consists of two drivers connected in series where one driver switches the voltage to the second and the inputs to both are controlled by the valve select and duration logic. Both drivers must operate to drive one coil of the thruster valve and fire the thruster.

A safety circuit checks the presence of the -5V line before firing the thruster. The thruster valves have two coils each, one driven by each DSL. Only one coil need be energized to fire the thrusters.

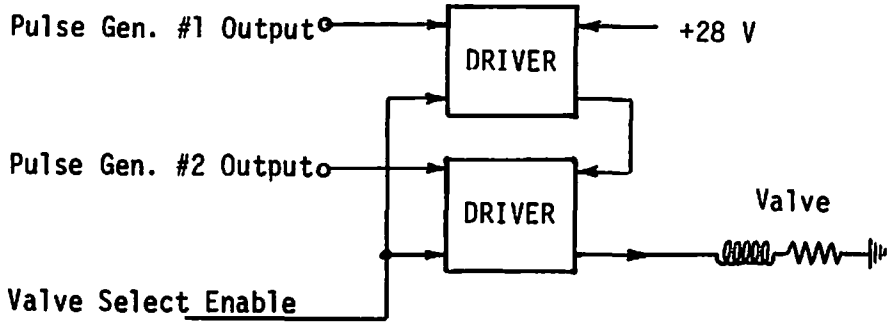


Figure 3.3-14. Valve Drivers

Despin Logic. The two redundant DSAs provide a signal to terminate despin when the spin speed drops below 22 ± 1.5 rpm, the speed desired for deployment of the RTGs. Each DSA is a switch which closes at spin rates above 22 rpm and opens for spin rates less than 22 rpm. A block diagram of the despin logic is given below.

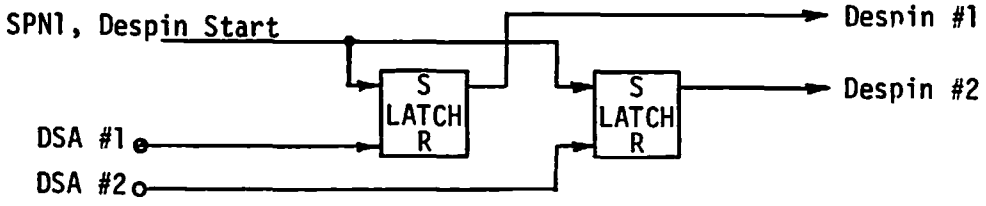


Figure 3.3-15

The signal to start despin is obtained either from the sequencer or from ground command, (SPN 1). This signal sets two latches. The output of the latches, Despin #1 and Despin #2 both must be true to fire the despin thruster. These signals go to the appropriate gates in the valve select logic to energize SCT 2.

To terminate despin, if either DSA resets one of the latches, the thruster will be shut off. Once despin is terminated by the DSAs, steady despin cannot be commanded again. Further spin rate changes are effected in pulses on command as described in Section 3.3.2.2

3.3.1.4.3 Pulse Centroid Location Errors. The quadrature pulses within the CEA are nominally initiated at roll angles 0°, 90°, 180°, and 270°. The actual precession directions differ from these angles because of the finite pulse widths, and time delays in pulse execution. The sources of time delay are both random and systematic. Some of these are centroid shift from command, centroid uncertainty, SSA errors (bias and random) SPSG and CEA errors, and thruster misalignment error.

Centroid Shift. The difference between the command signal centroid and the thruster pulse centroid is known to be about 29.5 msec for the 125 msec pulse length. This centroid shift varies for the selected pulsewidth and from one thruster to another. A known shift of about 4 msec occurs with changes in tank pressure. The uncertainty in the centroid is about ± 3 msec, 3σ

SSA Error. An electronic time delay exists in the SSA but only amounts to an error of about 0.006° at 4.8 rpm. Numerous random errors also exist; electronic triggering, alignment, thermal and mechanical stability, differential responsivity and spacial uniformity. These errors vary with sun aspect angle $\emptyset$, the angle between the spin axis and the sun and yield RSS values of about 1° at $\emptyset = 1°$ and 0.373° at $\emptyset = 10°$, at nominal spin.

SPSG. The spin period sector generator contributes timing errors on the order of 150 μsec.

CEA. The sun pulse delay in the CEA is less than one μsec. Errors in the one shot metering of the pulse are at most 0.002 sec for the 2 sec pulsewidth. The CEA error decreases linearly to 31 μsec for the 31.25 msec pulsewidth.

Thruster Alignment. Angular shift in the torque vector due to misalignment is approximately $\pm$ 0.192°.

The various errors described above can be combined to yield estimates of the pulse centroid location (precession direction) from the initiation of the command. The centroid is obtained by adding the thrust-to-command shift to the command centroid (1/2 pulsewidth). The CEA one shot error is converted from a uniform distribution to gaussian (multiplied by $3/\sqrt{12}$) and added to the centroid errors. The random errors are RSS'd separately to yield the shifts and uncertainty given.

Table 3.3-3. Nominal Pulse Execution Errors

Pulsewidth (sec)	Pulse Centroid Shift from Reference at 4.8 rpm (delay)		PK-PK Uncertainty (deg)
	Degrees	Seconds	
0.03125	1.3°	0.045	0.087
0.125	2.56°	0.092	0.087
0.5	8.06°	0.280	0.089
1.0	15.26°	0.530	0.094
2.0	29.68°	1.031	0.114

For this table, a constant centroid shift of the gas pulse from the electrical pulse centroid of 29.5 msec was assumed. Additional uncertainty of thruster alignment ($\pm$ 0.192°) and sun sensor pulse uncertainty (0.373° at ϕ = 10°) must be included in the final centroid uncertainty. The uncertainty in the pulse centroid location is a uniformly distributed error and will average 1/2 the given value for long pulse trains.

3.3.1.5 Sensor and Power Control Subassembly (SPC) Description. The left hand side of Figure 3.3-3 shows the SPC block diagram. The SPC processes the SRA outputs to acquire and lock onto Canopus, provides logic for roll reference selection, supplies phase coherent 32,768 Hz and 512 Hz clock outputs, generates 30 μs reference pulses at 0°, 90°, 180°, and 270°, and warns the PSE of an under voltage condition on the 5 volt line. In the following paragraphs, each block of the SPC will be described in detail.

3.3.1.5.1 Clock Countdown Chains and Clock Inhibit. Two redundant countdown chains are provided for generation of the reference frequencies. They are driven by independent asynchronous time frequencies A and B (TFA and TFB). The redundancy applies for the 32,768 Hz and 512 Hz signals with the 16,384 Hz, 256 Hz and 4 Hz being available from the first countdown chain only. The DSLs use the 32,768 Hz clock only, while the PSE uses 32,768 Hz and 512 Hz clocks. The 512 Hz lines are commanded on by ACS1 or ACX3 and off by ACX2 or ACX4. A command to start or override the stored program, ACX1 or ACX3, stored in a latch circuit, will turn the 512 Hz reference on. Similarly, a command ACX2, ACX4, or a master reset will inhibit the clock inputs to the PSE.

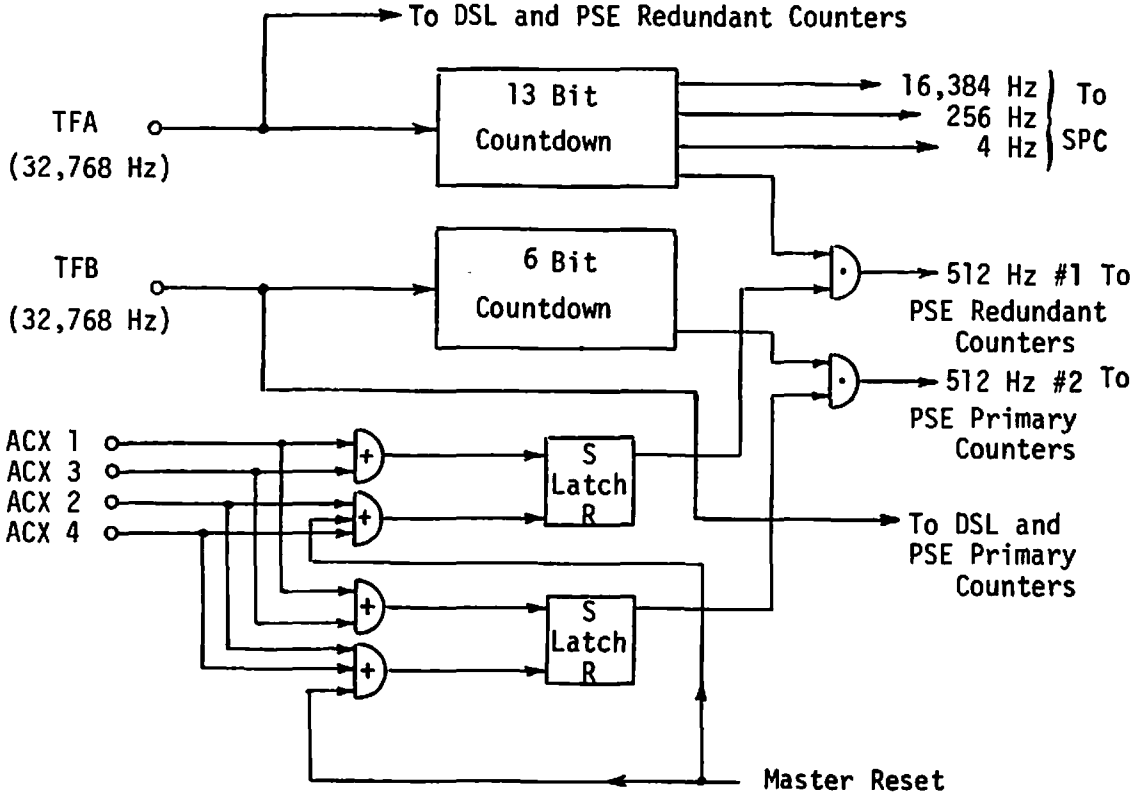


Figure 3.3-16. Clock Countdown and Clock Inhibit

3.3.1.5.2 Undervoltage Detectors and Power Switching. The power switching is done by the correct actuation of the relays K1, K2, and K3. Power switching to DSLs is done by ground command as follows:

PSE 9 switches the power to both DSLs off and switches PSE power on.
SLA 9 switches DSL No. A on and DSL No. B off; does not affect PSE
SLA 9 switches DSL No. B on and DSL No. A off; does not affect PSE
PSE 0 switches the PSE off; does not affect the power to the DSLs.

The switching block diagram (Figure 3.3-2) shows various power switching states and the associated commands.

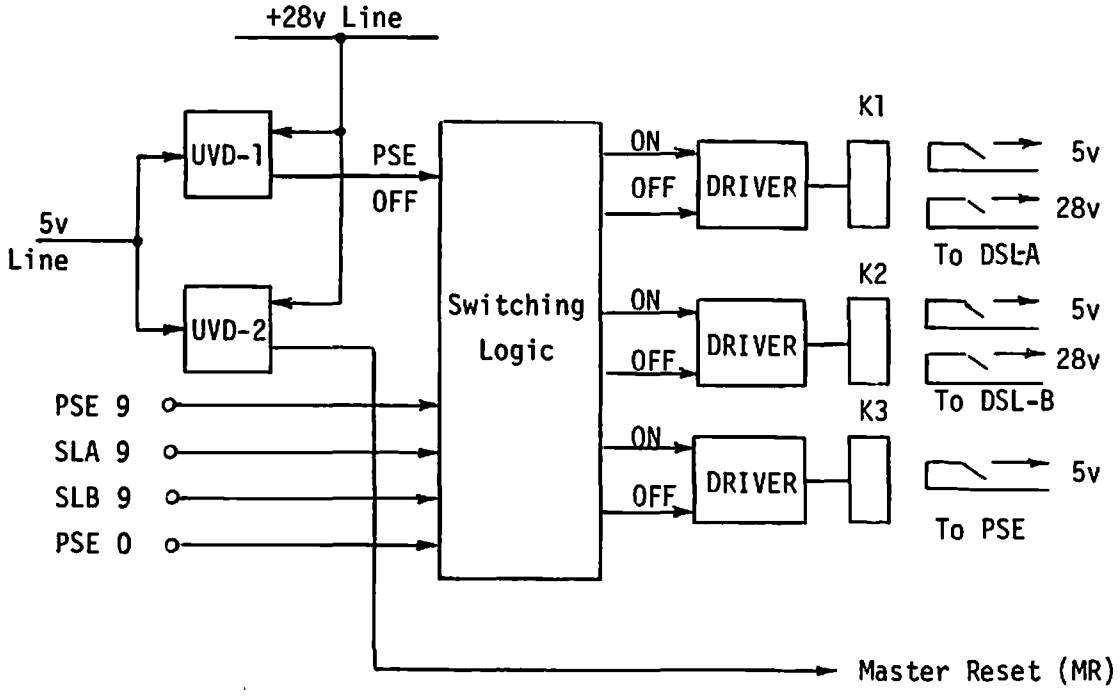


Figure 3.3-17. Undervoltage Detectors and Power Switching

The +5v line is monitored by two undervoltage detectors UVD-1 and UVD-2. When the line voltage drops below its limit of 3.5v $\pm$ 0.5v, UVD-1 turns the PSE off while UVD-2 produces the master reset pulse (MR) which resets the PSE and stops the 512 Hz clocks. The two undervoltage detectors are employed for redundancy and are identical except for the difference in their functions.

The PSE commands (PSE9, SLA9, SLB9, and PSE0) are fed into the switching logic. The logic actuates the relays K1, K2, and K3 through three driver stages, which in turn switches on the power to the DSLs and the PSE.

3.3.1.5.3 Octant Generator.

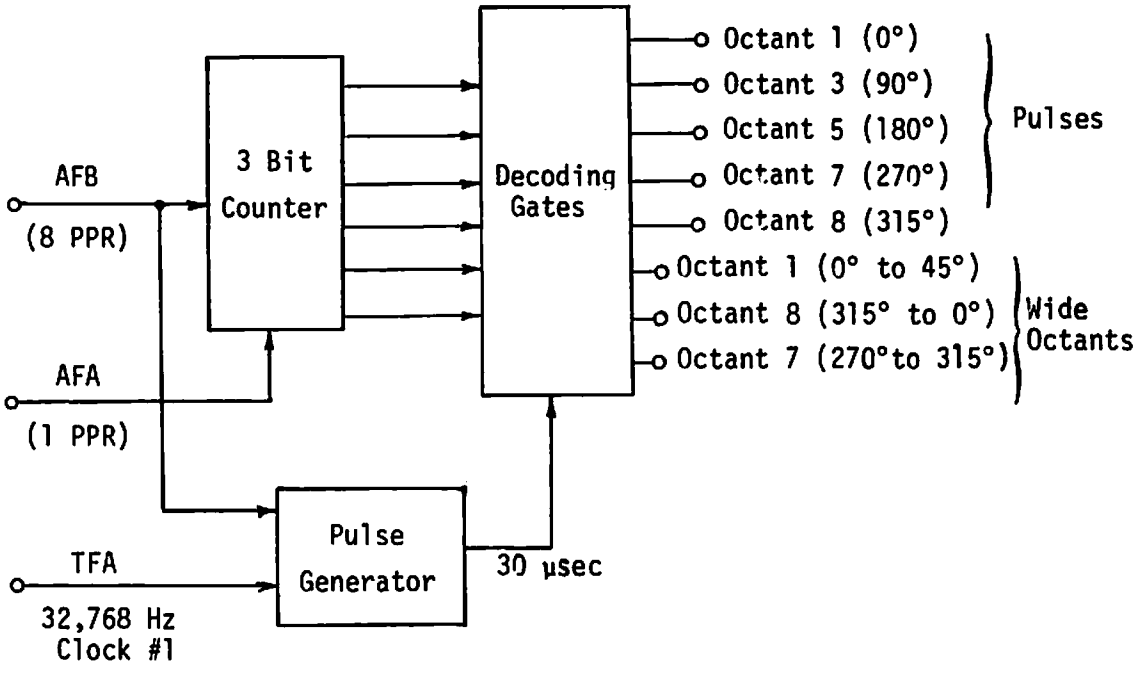


Figure 3.3-18. Octant Generator

TFA and AFB are square pulse trains and AFA is a 60 µsec pulse. Angle frequencies A and B (AFA and AFB) are synchronized to the spacecraft spin rate, and come from the spin period sector generator in the DTU. A three bit counter which is started at every revolution by AFA, counts the 8 pulses of AFB. A narrow pulse of about 30 µs is generated at the beginning of each pulse of AFB by the pulse generator. The counter and the pulse generator feed decoding gates which produce the following outputs:

Narrow Pulses (30μs Wide)	Wide Octants (45° Wide)
Octant 1 pulse (0°)	Octant 1 (0° to 45°)
Octant 3 pulse (90°)	Octant 8 (315° to 360°)
Octant 5 pulse (180°)	Octant 7 (270° to 315°) (Gate)
Octant 7 pulse (270°)	
Octant 8 pulse (315°)	

These outputs are used as follows: The narrow pulses of Octants 1, 3, 5, and 7 are sent to the DSLs for use with the quadrature precession capability; the Octant 8 pulse resets the star counter. Wide octants 1 and 8 are used for star location logic and Octant 7 is used to enable the star angle gate.

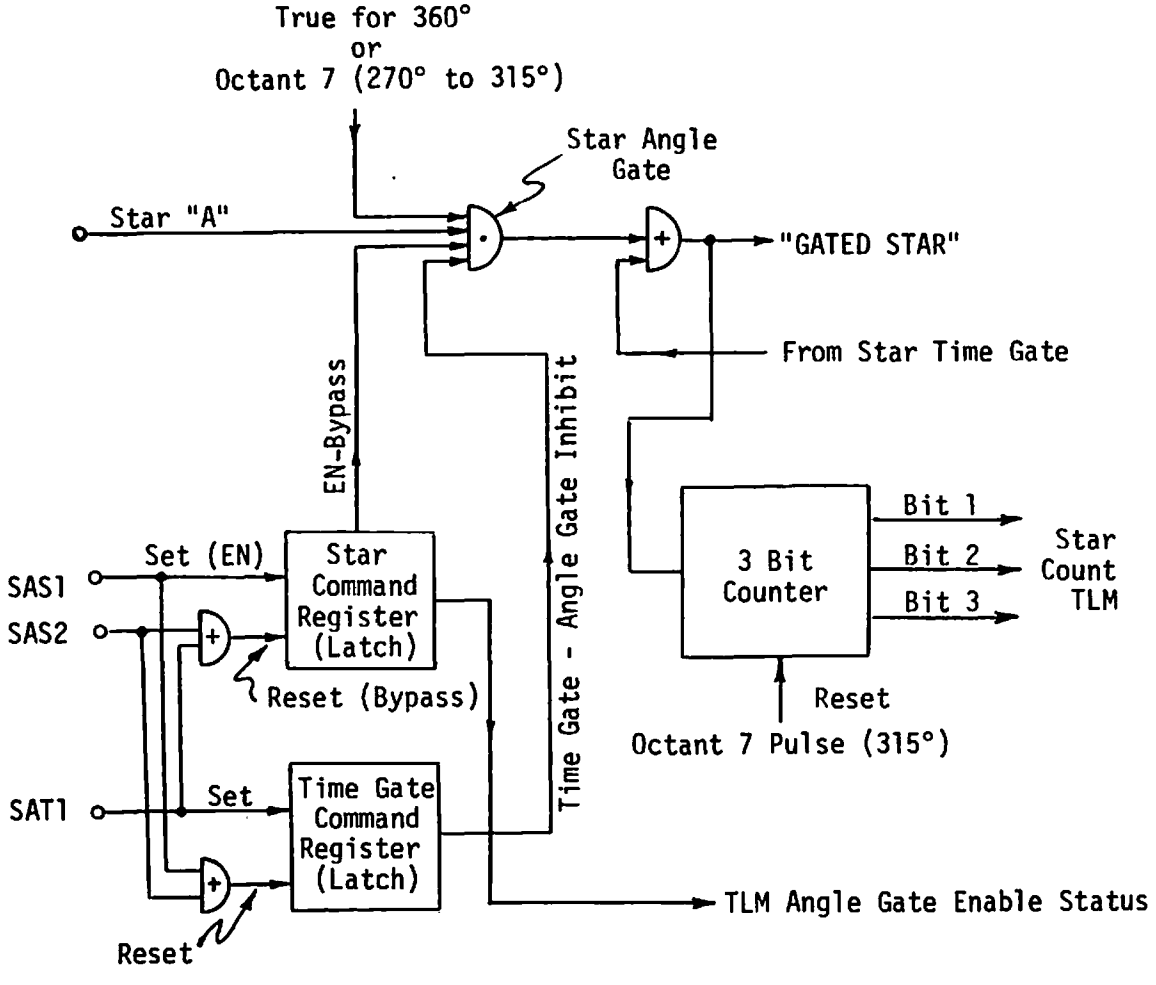


Figure 3.3-19. Star Angle and Star Count

3.3.1.5.4 Star Angle Gate. The commands SAS1 and SAS2 are stored in a command register latch. SAS1 enables the angle gate allowing only star pulses within the 270° -315° sector 7 to pass. SAS2 disables the angle gate permitting the entire 360° star scan output to pass. Both of these

commands disable the time gate. Command SAT1 inhibits the angle gate such that no pulses will be passed through. The time gate is enabled by this command and all stars after the first are blanked for the time entered in the time gate register (Section (3.3.1.5.5). When SAT1 is commanded, the time gate sends an inhibit function to the angle gate and an enable function to the time gate. The "gated star" is output to the star count and to the star delay. The star angle gate enables a region of 45° approximately centered at a roll angle corresponding to Canopus (See Section 4.3).

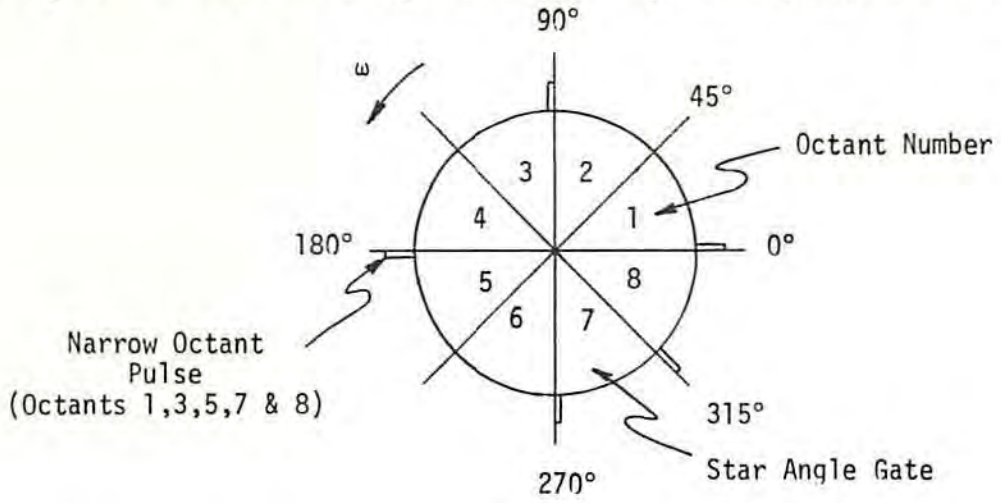


Figure 3.3-20. Octant Locations

3.3.1.5.5 Star Time Gating and Star Advance. The value to be entered in the star time gate is controlled by commands SAT2 and SAT3 which respectively increase or decrease the value in the time gate register by the amount of the least significant bit. This register stores the value of the desired delay until changed by these commands. The start-stop logic controls the operation of the time gate. A gated star arrives and enables a pulse generator in this logic which in turn produces a 30 μ sec transfer strobe signal using the 32 KHz and 16 KHz input reference frequencies. The strobe transfers the complement of the data stored in the time gate register into the 6 bit ripple counter. The most significant bit is always preset to zero in the ripple counter with the result that the time gate cannot be set to a value less than 8 seconds. To set the time gate at 8 seconds, all zeros should be loaded into the time gate register. Only the 5 least significant bits are controllable by ground command and are read on telemetry. After the strobe transfer of data, the 4 Hz clock is enabled by the start-stop logic and the ripple counter begins to count upwards. The decoding gate observes the ripple counter and when the counter reaches 111111, the decoding gate sends a reset pulse to the start-stop logic. The decoding gate output resets a flip-flop in the start stop logic which re-enables the time gate for the next cycle. The decoding gate input is "anded" with the 4Hz ripple counter input clock such that it remains all ones for a full 1/8 second. The flip-flop is held in the reset condition for this time such that the time gate cannot be restarted, although stars can pass through the time gate during this time.

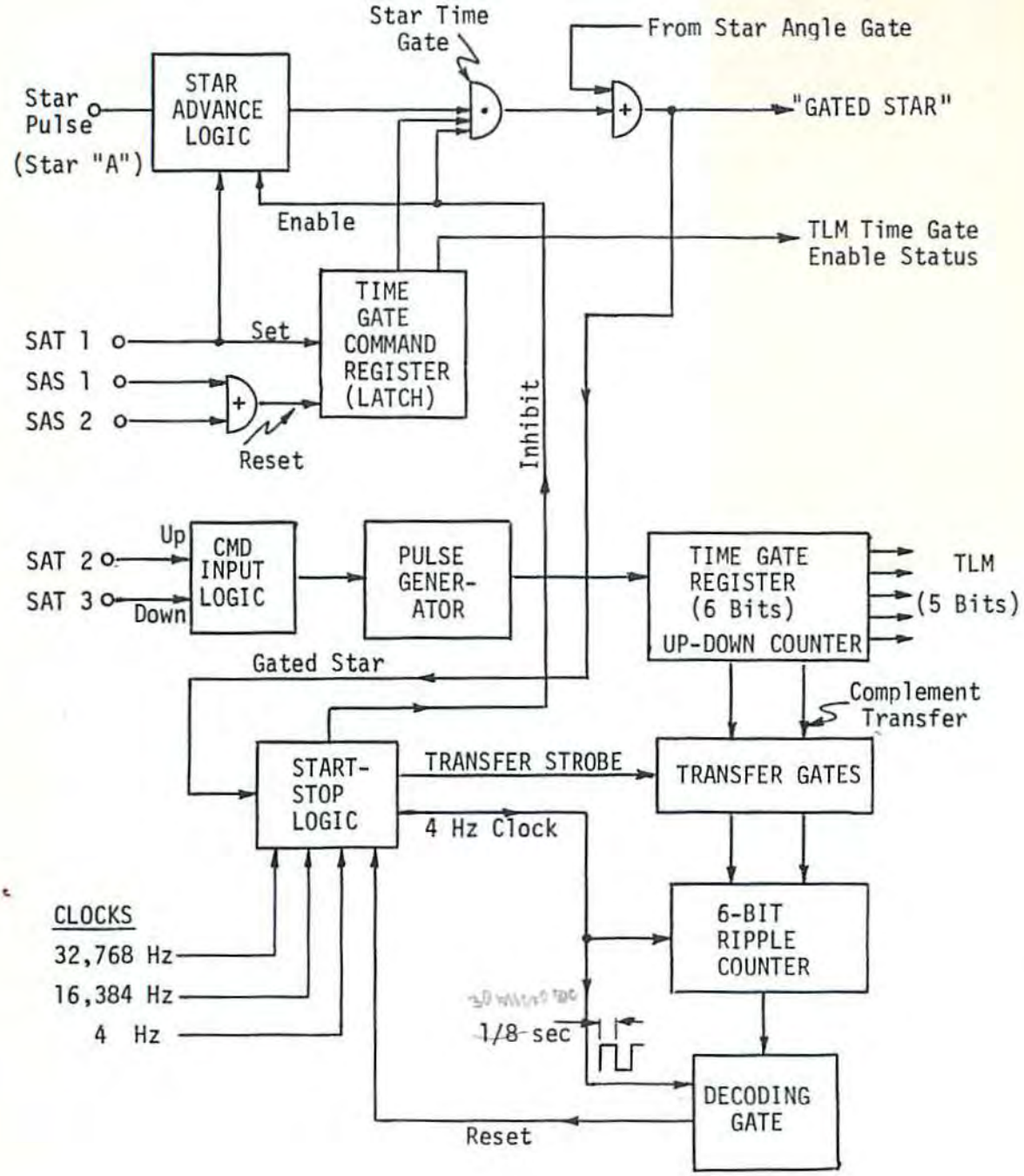


Figure 3.3-21. Star Time Gate and Star Advance

The gated star is asynchronous with the 4 Hz time reference into the time gate. Since the first 4 Hz pulse can arrive either instantly or 1/4 sec after the gated star, there is a 1/4 sec uncertainty in the value set into the time gate. The value set into the time gate must be at least 1/8 second less than one revolution period in order to lock onto a star. The minimum opening of the time gate is 1/8 sec (hold time in reset) while the maximum opening is the hold time plus the uncertainty, 3/8 sec.

The start-stop logic sends an inhibit signal to the star time gate which prevents star pulses from being passed. This inhibit starts upon arrival of the gated star and ends with the"one"state of the decoding gate. The star time gate is also controlled by the time gate command register latch which is also shown in the previous block diagram for the angle gate logic. The commands SAT1, SAS1, and SAS2 are used with the command register latches such that either the time or the angle gate operates but not both.

The star time gate enable command (SAT1) will cause the time gate to lock onto the second star that occurs after the time of the command. The command acts both as an enable and an advance. All SAT1 commands after the first are treated only as star advance commands. The star advance command sets a flip-flop in the star advance logic which blocks the first star pulse after the time gate inhibit is over. The second star pulse is allowed to pass and results in the advance of the time gate to the next star. This second star becomes the gated star which starts the time gate again. Star advance occurs only once per command.

Example:	MSB LSB
Data in Time Gate Register (TLM)--	X01101
Data Transferred to Ripple Counter--	010010
Count in Ripple Counter at Time Gate End--	111111

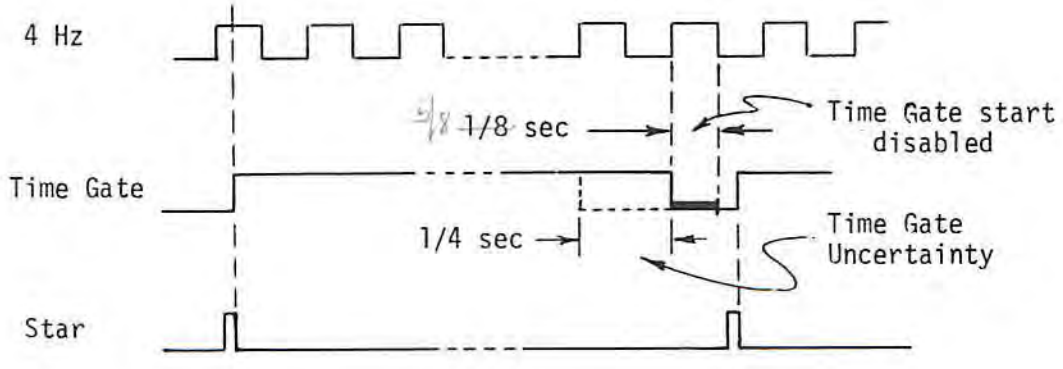


Figure 3.3-22. Time Gate Timing

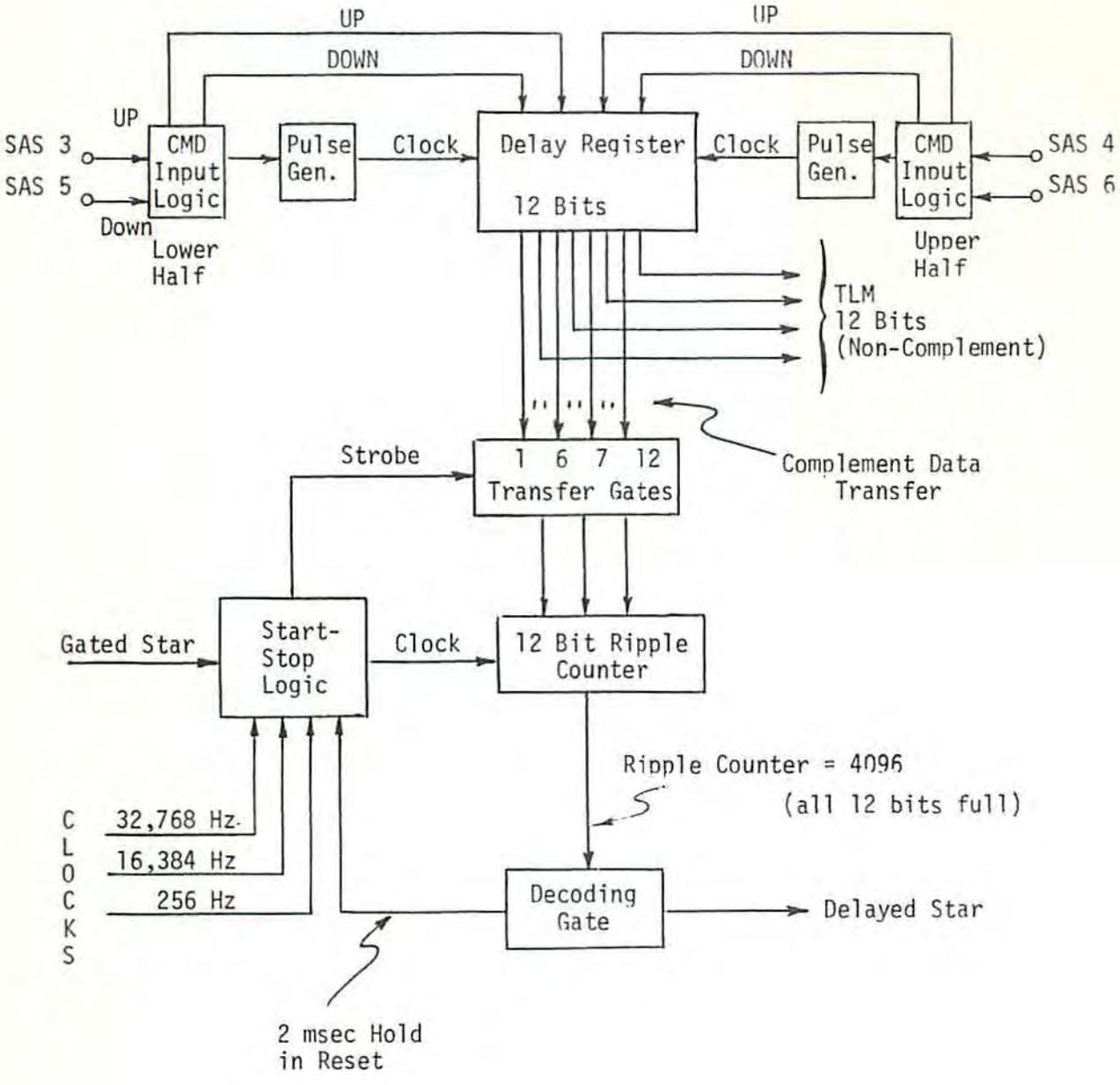


Figure 3.3-23. Star Delay Logic

3.3.1.5.6 Star Delay. The star delay circuit is similar in operation to that of the time gate. The delay is stored in a register which maintains its value until changed by ground commands. This register can be changed up or down in increments of 0.25 sec (SAS4 or SAS6) or in increments of 1/256 sec (SAS3 or SAS5). All 12 bits of the delay register are telemetered. The gated star pulse into the start-stop logic initiates action which is identical to that of the star time gate circuit. The gated star pulse operates a pulse generator which produces a strobe. This strobe causes the complement of the data in the delay register to be transferred to the 12 bit ripple counter. The ripple counter then begins to count upward, receiving a 256 Hz clock count input from the start-stop logic. When all ones (count = 4095) are detected in the ripple counter by the decoding gate, a delayed star pulse is issued and the start-stop logic is reset in preparation for the next gated star. The star delay must not be set greater than the period between the same gated star (one spacecraft revolution period) since the delay logic is disabled until the delayed star pulse is issued. Any star which occurs during the time that a star is being delayed will be blocked out by the delay circuit.

3.3.1.5.7 Reference Selection. Commands SEN1, SEN2 and SEN3 are given to select "Sun A", "SunB", and "Star" signals for the reference outputs to the DTU. The decoding gates for these commands are fully redundant as shown in Figure 3.3-24.

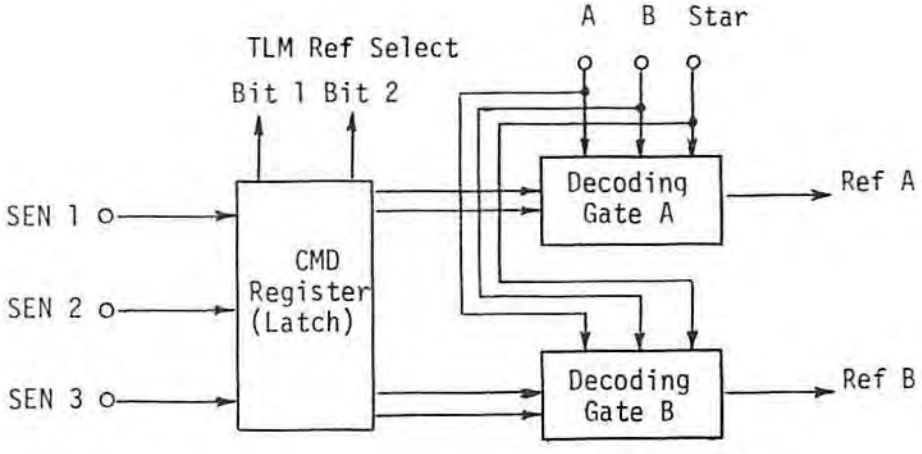


Figure 3.3-24. Sensor Select Logic

The commands are stored in a command register with 2 bits coded as below:

COMMAND	REGISTER BITS	
SEN 1 (Sun Ref. A)	1	1
SEN 2 (Sun Ref. B)	X	0
SEN 3 (Star Ref.)	0	1

The output of the command register drives two redundant decoding gates which produce the reference "A" and "B" from the sun and star signals.

3.3.1.5.8 Star Location. The occurrence of a star in Octant 1 or Octant 8 is registered in two flip-flops and constitues the star location telemetry output. The register is reset by the 270° pulse from the octant generator (see Figure 3.3-18).

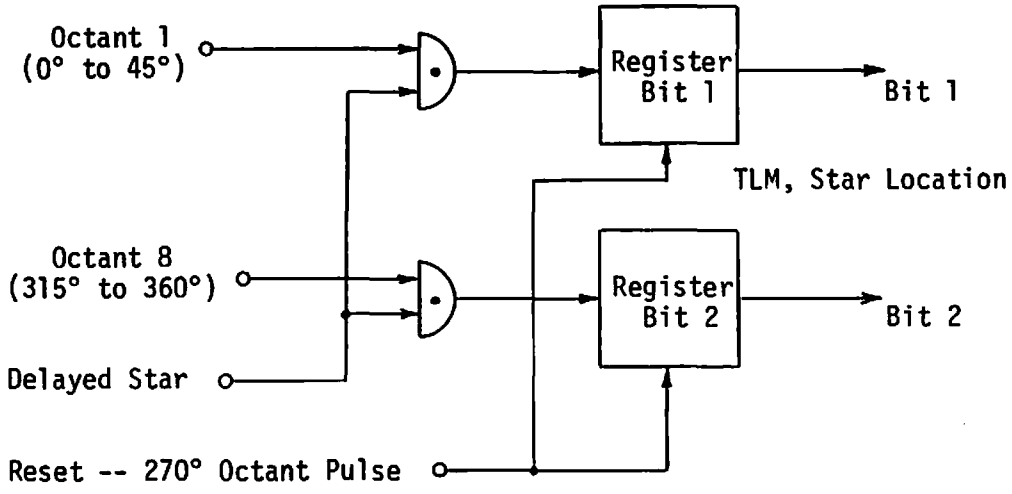


Figure 3.3-25. Star Location Logic

3.3.1.5.9 Star Level. This is a one bit register which signals the presence of "Star B" (discrete) signal. Its output feeds the star level TLM line. The register is reset after the telemetry has been read. The presence of this signal indicates that the "star" is greater than 180% of Canopus.

3.3.1.5.10 Star Coincidence.

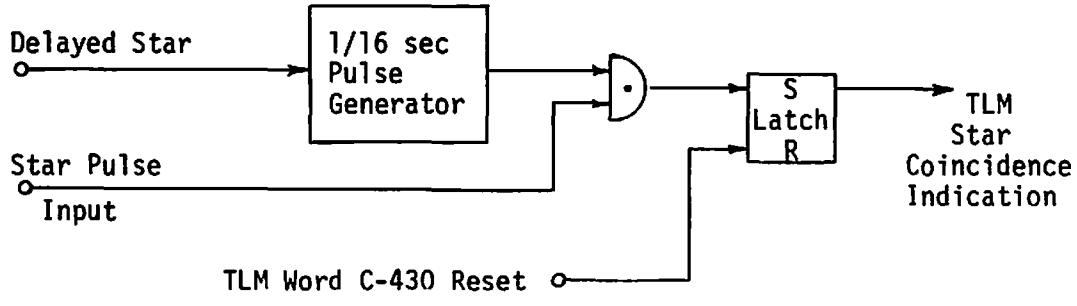
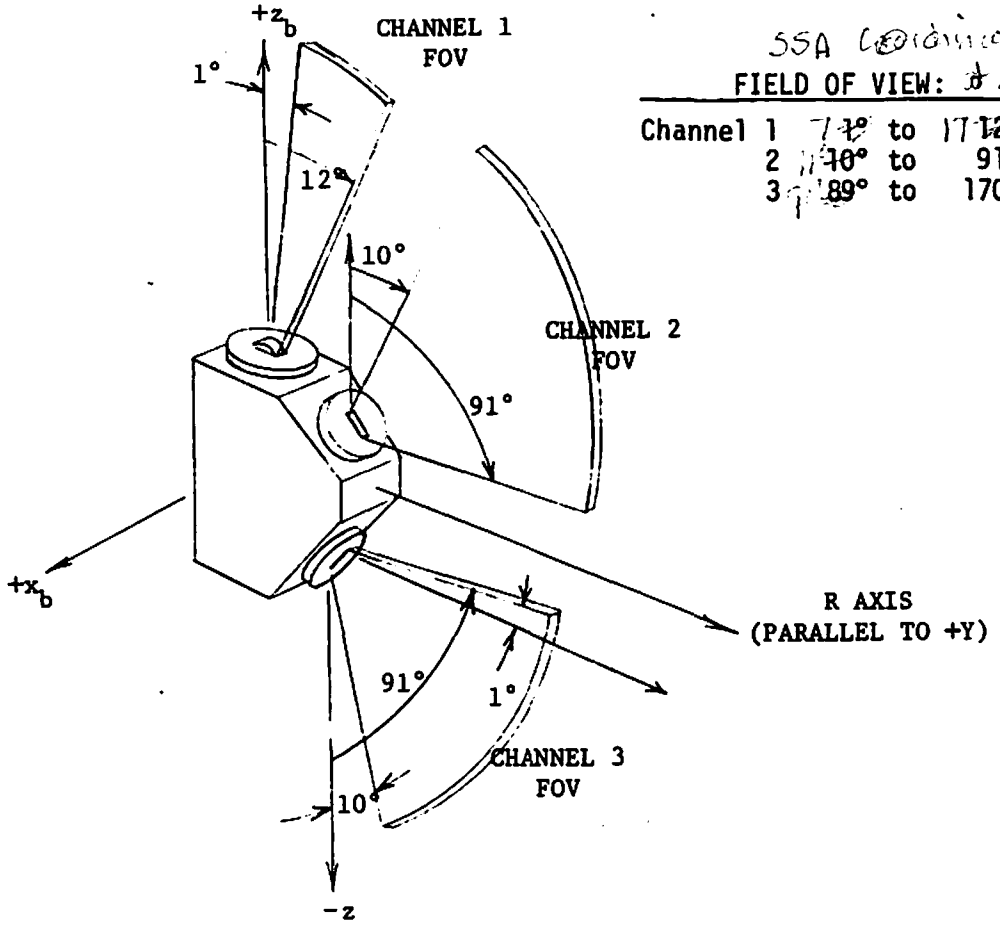


Figure 3.3-26. Star Coincidence Logic

The delayed star produces a gate pulse 1/16 sec wide. This gate has a timing uncertainty of one clock period, 1/256 sec or 3.90 msec. Any pulse arriving during this gate will latch an RS latch circuit which is reset after the telemetry is read.

3.3.1.6 Sun Sensor Assembly Description. The Sun Sensor Assembly (SSA) is a single assembly mounted at the edge of the high gain antenna dish along the +y axis as shown in Figure 3.3-1, Section 3.3.1.1. It operates from 0.9 to 6.0 AU from the sun over a spin speed range of from 2 to 85 rpm, and gives one pulse per revolution. The sun sensor detects the sun as it passes through the z, +y plane and issues a pulse. The sun sensor consists of 3 channels with overlapping fields of view as indicated below.



SSA coordinates S/C coordinates
FIELD OF VIEW: ~~*~~ Jupiter

Channel 1	~~7°~~	to	~~17°~~ 12°
2	~~81°~~ 10°	to	91°
3	~~89°~~	to	170°

Figure 3.3-27. Sun Sensor Field of View

The location of the sun sensor defines the R axis (Reference axis) as shown in Figure 3.3-27. This axis is parallel to the +y axis.

The sun sensor is internally redundant in the identical channels 2 and 3. These redundant parts are referred to as Sun A, Sun B, and can be selected by command. Selection of Sun Sensor A gives output from Channels 2A and 3A. Selection of Sun B gives output of Channels 1, 2B, and 3B as shown in Figure 3.3-28.

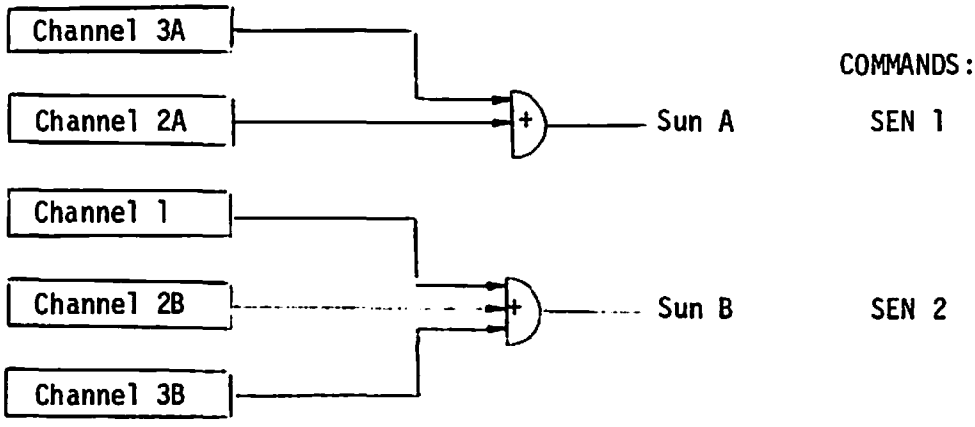


Figure 3.3-28. Sun A, Sun B Block Diagram

Each channel of the sun sensor is designed using two detectors (for a total of 10) behind a slit as shown.

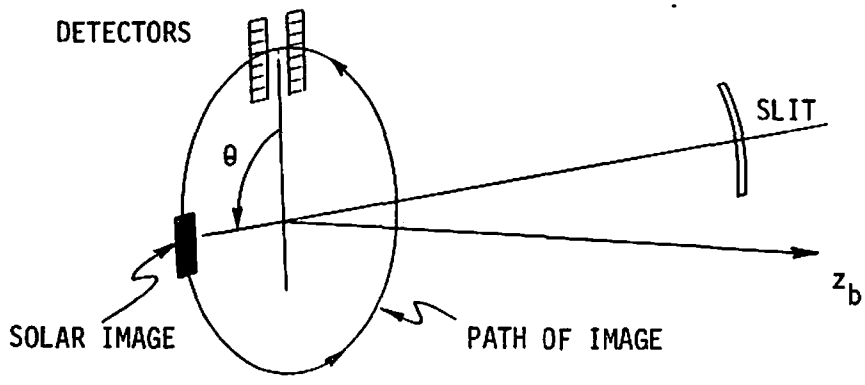


Figure 3.3-29. Idealized Representation of Optical Principle

The sun pulse is derived using the differential photocurrent of the two detectors and is produced when the solar image formed by the optical slit (and a lens for Channel 1) passes over the detectors such that the differential exceeds a positive threshold. The individual signal output of each detector is shown below in Figure 3.3-30. The differential output

is also displayed. The width of the sun sensor field of view is nominally that determined by accuracy, however, the detectors may sense the sun within as much as 30° from the R axis in channel 1 near Jupiter (6° for channels 2 and 3 near Jupiter). This corresponds to the angular width of the detector responses shown below. It is possible for the sun sensor to continue producing pulses in a channel even if the first detector (#1, negative detector) fails. However, degraded performance will occur in the form of increased angular error ($\dot{\theta}_e$). This angular error can be measured on orbit by using the star location logic (Section 3.3.1.5.8).

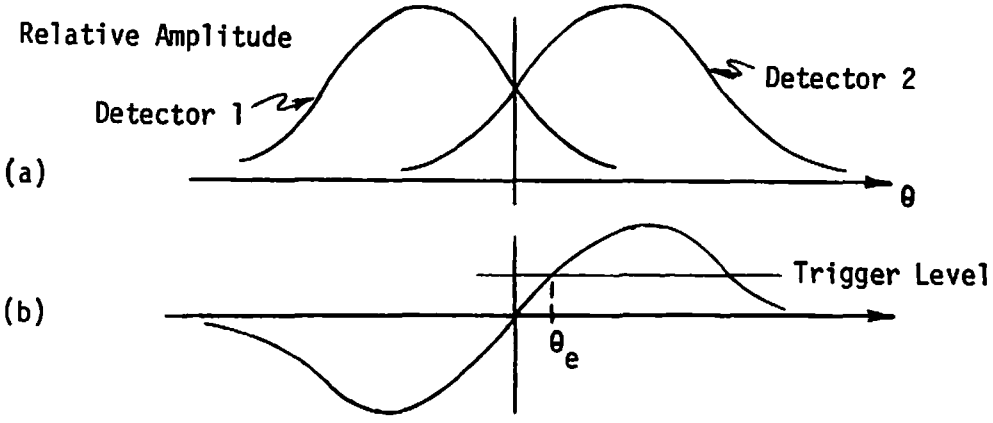


Figure 3.3-30. (a) Individual Detector Outputs
(b) Differential Outputs

The sun sensor consumes about 0.5 watts of power. Albedo levels from the earth and Jupiter result in about 0.02 μamp max. The trigger level is approximately 0.2 μamps so that only the sun should produce pulses. Nominal accuracy of the sun sensor is as shown in Figure 3.3-31.

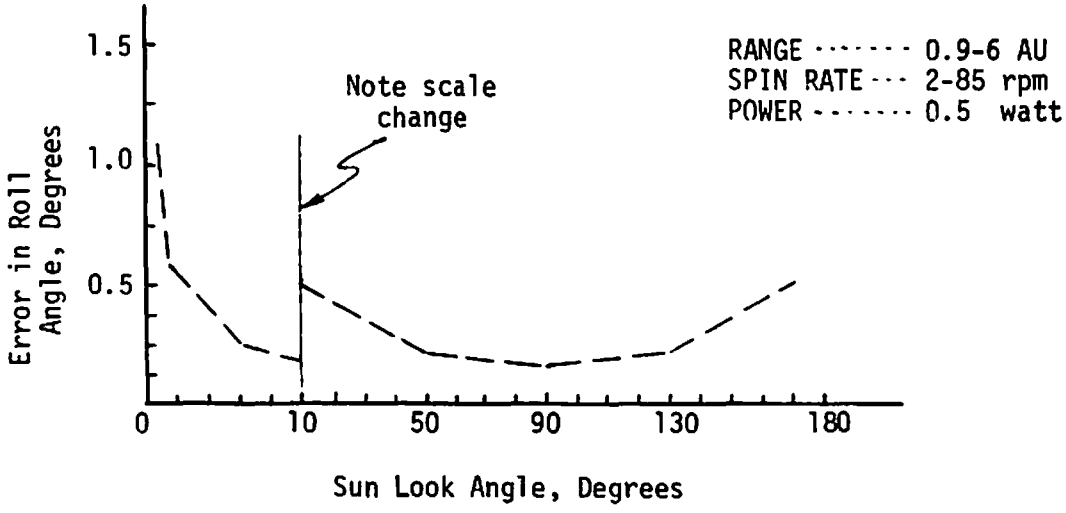


Figure 3.3-31. Roll Angle Error

3.3.1.7 Stellar Reference Assembly Description. The Stellar Reference Assembly (SRA) is a single, non-redundant unit mounted (except for its light shade) within the spacecraft equipment compartment. It is a star "pipper," not a tracker. A single pulse is issued when a source enters the field of view. The primary star selected for reference purposes is Canopus, but sources such as other stars, planets, asteroids and the sun also may be detected. The field of view of the SRA is ±19° from the S axis in the S-Z plane (see Figure 3.3-32) and has a width of about 0.5° determined by optical resolution and obscuration. The SRA operates over a spin rate range of 2 to 5.8 rpm and has two outputs, a signal and a discrete as follows:

Signal Output: Energy ≥ 50% of equivalent Canopus
Discrete Output: Energy ≥ 180% of equivalent Canopus
Accuracy: ±0.16° (signal) ±0.75° (discrete)
Power: 0.5 watt

A complete SRA accuracy summary is presented in Table 3.3-4.

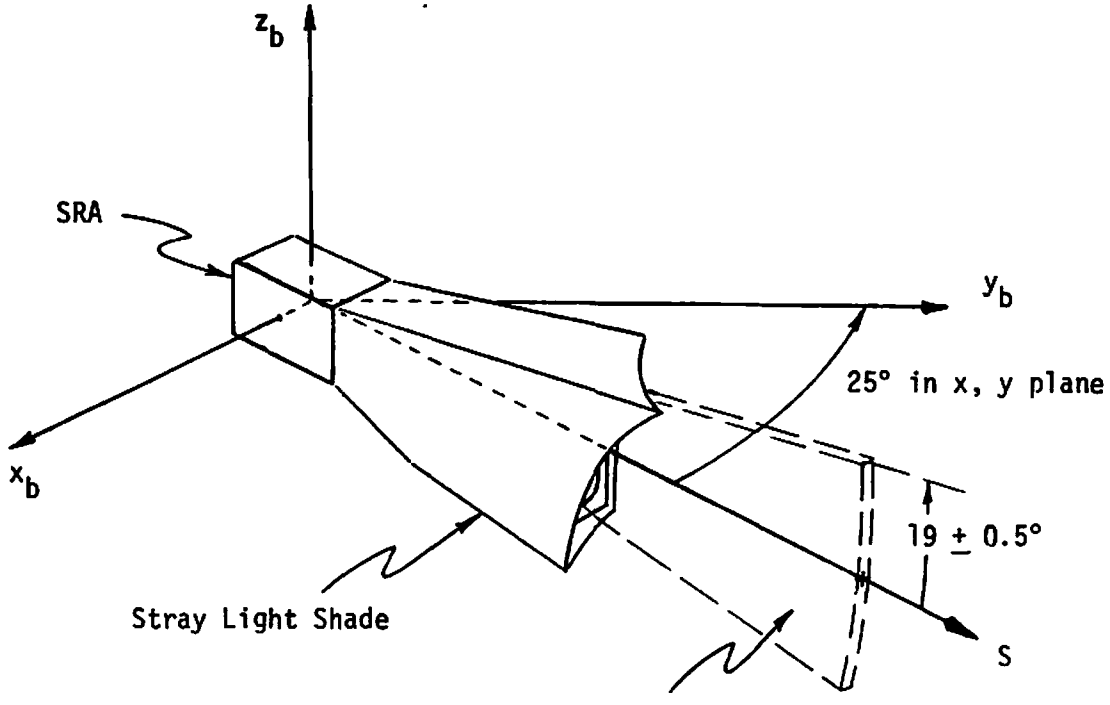


Figure 3.3-32. SRA and Light Shade

Table 3.3-4. SRA Accuracy Summary

Error Source	Error Magnitude	
	Bias (deg)	Jitter (deg 3σ)
Nominal over 2 to 5.8 RPM spin rate range	±0.069	±0.03 Max
Field of View non-uniformity	±0.073	±0.01 (Est)
Vibration	±0.04 (Approx)	
Thermal/Vacuum		
1) 95°F	-0.081	+0.043
2) 0°F	-0.039	+0.023
Electronics	±0.03 (Approx)	±0.01 Max, SNR≥30
Physical Alignment	±0.10	
RSS Nominal	0.157	±0.06*

* Delay adds 0 to +0.115° @ 4.8 rpm.

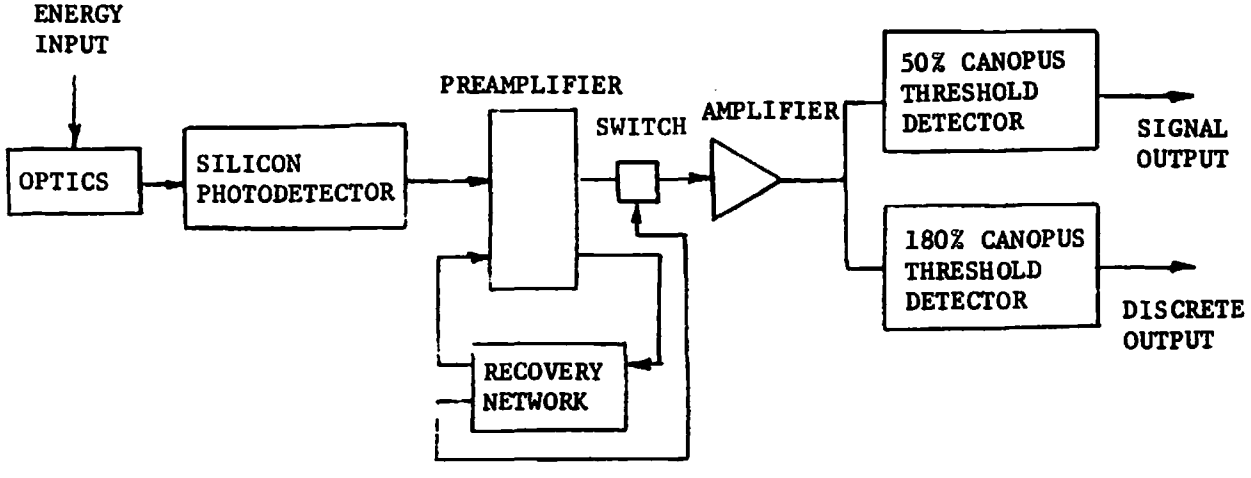


Figure 3.3-33. Stellar Reference Assembly (SRA) Block Diagram

A block diagram of the stellar reference assembly is displayed in Figure 3.3-33. The recovery network senses high level light inputs to the detector (such as Jupiter during flyby) and essentially returns the SRA electronics to normal operation. With this circuit in operation, the SRA will normally not produce pulses in response to Jupiter. If pulses do occur, they will probably coincide with Jupiter's trailing edge. The time required for the circuit to return to an acceptable operating level after saturation is 0.3 second worse case.

An indication of the effect of temperature on SRA errors is displayed in Figure 3.3-34. The curve indicates that in order to keep the SRA errors negligible, the operation temperature should not be greater than 25°C. The SRA temperature is given by word C-303 of the analog telemetry.

The light shade depicted in Figure 3.3-32 is necessary to prevent sunlight reflected from the spacecraft structure, appendages or the planet Jupiter. The coordinate system of Figure 3.3-35 is a planar representation (stereographic projection) of a polar spherical coordinate system attached to the rotating spacecraft. The light shade and SRA field of view are as indicated in the figure. The SRA field of view is a narrow (0.5°) slit, 40° long. The stray light shade prevents all light sources outside of the region (defined by the boundary lines shown and containing the SRA FOV) from affecting the SRA star detection. Specifically, the shade prevents direct light from Jupiter from saturating the SRA (operating the recovery network) at those times when Canopus is to be sensed. The trajectory representing Jupiter's location is treated as though the spacecraft were not spinning, with the SRA pointed at Canopus as shown. The orientation of each of the spacecraft axes are as indicated for this configuration. The Jupiter trajectory is a nominal encounter trajectory defined by

Launch date	March 4, 1972
Encounter date	December 1, 1973
$R_{periapsis}/R_{Jupiter}$	3.0
θ	14°

Additional explanation of Figure 3.3-35 and its relationship to other encounter considerations is provided in Section 4.7 (Specific Preparation and Encounter Operations).

3.3.1.8 Despin Sensor Assembly Description. The Despin Sensor Assembly is a single assembly mounted within the S/C equipment compartment as shown in Figure 3.1-1. The DSA is redundant in that two are used per spacecraft. The DSA senses centrifugal acceleration of 0.5 g's ±5%. It provides a bilevel (switch) output. The switch is opened prior to despin and closes when the radial acceleration sensed is reduced to 0.5 g's. The DSA is a spring-mass system with a natural frequency of 12 Hz and a damping ratio of 3. The DSA will nominally terminate despin at a spin rate of 22.1 rpm. However, this may vary as much as ±1 rpm for large amplitude wobble. Upon deployment of the RTGs and Magnetometer, the spin rate should be 5.0 rpm. This should normally be corrected to 4.8 for cruise.

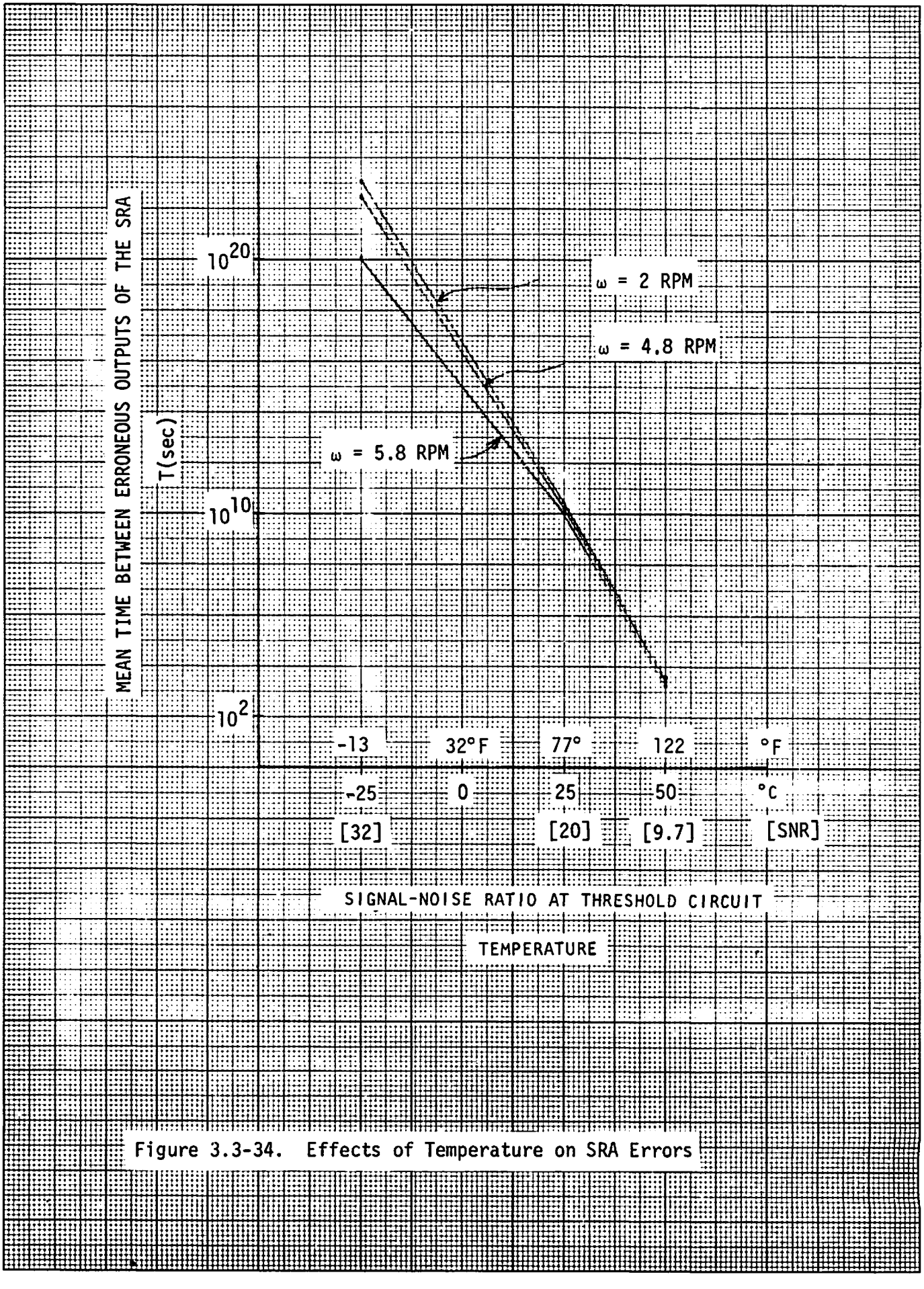


Figure 3.3-34. Effects of Temperature on SRA Errors

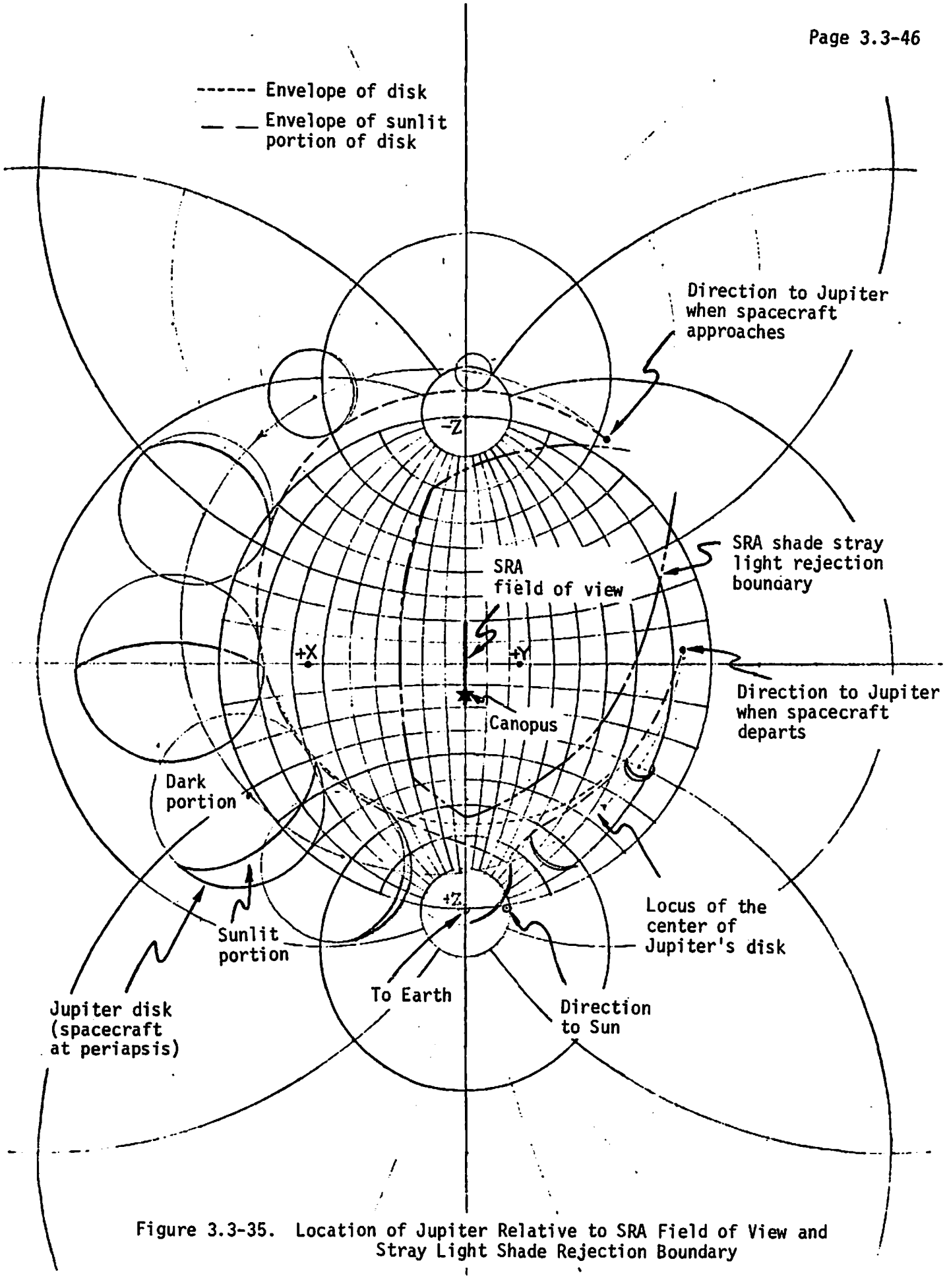


Figure 3.3-35. Location of Jupiter Relative to SRA Field of View and Stray Light Shade Rejection Boundary

3.3.2 Operation by Sequence

The major propulsive maneuvers to be performed by ACS consist of despin, open loop precession maneuvers (including ΔV, or trajectory correction), spin control, conscan, and ΔV/SCT. The general characteristics and important results of open loop maneuvers and ΔV/SCT are discussed in the various paragraphs of Section 4 since these functions are dependent on a number of other subsystems and are therefore discussed at the system level. Since despin and spin control are performed within ACS and require only propulsion subsystem operation, these are discussed in this section.

For operation of the spacecraft, the CEA can be operated in a number of modes associated with the primary functions of precession, ΔV, and spin control. These modes are:

	ACS Units Required
Precession	
Programmed	SPC, PSE, DSL
Conscan	DSL, SPC clocks
Fixed Angle	SPC, DSL
Real Time	DSL, SPC clocks
ΔV	
Programmed	SPC, PSE, DSL
Real Time	DSL, SPC clocks
ΔV/SCT	SPC, PSE, DSL
Spin Control	
Real Time	DSL, SPC clocks
Despin	DSL

No special set of commands are required to place the CEA in these modes other than to power the required units. Operation of the CEA in any of these modes is achieved via individual commands discussed in the following paragraphs.

3.3.2.1 Despin. Prior to injection and separation from the TE-364 third stage of the launch vehicle, Pioneer F/G is spin stabilized at about 60 rpm. After separation, the spin will have an upper bound at about 83 rpm with 0.9° wobble, a nominal spin of 60 rpm and 1.3° wobble, and a lower bound spin of 56 rpm and 1.8° wobble. The despin operation must reduce this spin rate to about 22 rpm. Deployment of the appendages will then result in the spin rate being reduced to about 5 rpm. For normal system operation, the despin is initiated by the sequencer (CDU, Section 3.9). At a predetermined time the sequencer will send a signal to the despin logic within the DSL. The DSL must have been turned on previously, command SLA9 or SLB9 and reset, command ACR1. The despin logic (discussed in Section 3.3.1.4) then turns on spin control thruster #2 which continues to fire until either of the redundant despin sensor assembly switches closes. This occurs at a 0.5g acceleration level, 37 inches from the c.g., the location of the DSAs. In the event that the sequencer should fail to initiate despin, despin can be initiated by ground command SPN1, (despin start).

Failure of the despin function caused by the failure of the despin logic is overcome by switching to the other DSL and commanding despin start (commands SLA9 or SLB9 followed by ACR1 and SPN1).

The despin logic within each DSL is wired directly to the despin thruster, SCT #2. No command is necessary. However, during despin the spin up thrusters (or any other) can be fired by command.

During the despin maneuver, wobble is expected to grow to as much as 5.6°. Damping of the wobble will not be effective until deployment of the magnetometer (see Section 4.1, Sequencer Events).

3.3.2.2 Spin Control. Spin control requires the use of one DSL which must be powered (see Section 3.3.1.2.3). It can be performed in a real time mode, however the command timing can be controlled from either the ground or by use of the stored command capability (Section 3.9). A typical command sequence starting from the ACS standby mode is as follows:

SLA 9	Turns on DSL-1
ACR 1	Clock Reset always follows SLA 9 and SLB 9 unless clock failure has occurred.
PUL 2	Selects 0.125 sec pulsewidth
SPN 3	Latches SCT #1 for increasing spin rate
SPN 2	Executes real time spin pulse

The following table indicates the nominal spin rate change for the selected pulse width.

Command	Pulse Width	Spin Change rpm	Precession (deg)	ΔV Max
PUL 1	31.25 ms	0.0032	0.0099	0.00048 m/sec
PUL 2	125.0 ms	0.0128	0.0396	0.0022
PUL 3	0.5 sec	0.0514	0.158	0.008
PUL 4	1.0 sec	0.103	0.317	0.0176
PUL 5	2.0 sec	0.205	0.634	0.0352

The precession resulting from the firing of the spin control thruster is caused by the thruster location 1.09 ft above the vehicle c.g. The magnitude of the precession is dependent on spin speed with values given for a nominal spin of 4.8 rpm. Normally, this precession is undesirable, but if the pulse is properly timed to occur at some known roll angle, the precession can be used to improve pointing accuracy. The telemetry conscan A sin θ pointing information can be used to determine the desired precession angle. The precession also can be cancelled if desired by performing spin control with two pulses from the same thruster fired 180° apart. This can be done by timing two SPN2 commands to occur an integral number of full revolutions plus one-half period apart. For example, at 4.8 rpm spin rate, two SPN2 commands, preceded by SPN3 and PUL3 should be sent 37.1 sec apart with spin rate change taken into consideration.

3.3.2.3 Conscan. The conscan function is described within the Communications Subsystem in Section 3.6; however, the conscan must be enabled within the CEA. The required commands are as follows:

- First, one DSL must be powered (see ACS power modes, Section 3.3.1.2.3) using commands SLA9 or SLB9 followed by ACR1. This transfers the CEA to either Partial 1 or Partial 2 power mode from the standby mode, the assumed state.

- The desired thruster pair must be selected. This depends on the phase within the Conscan Signal Processor (0° or 180°) for the pulse firing. If 0° phase reference is selected in conscan, the -X axis torque must be used in ACS, command PRE2.

- The proper pulsewidth must be selected. Normally, the 0.125 sec pulsewidth will be used with the medium gain antenna (PUL2) and the 0.03125 sec pulsewidth (PUL1) will be used with the high gain antenna.

- The spin period sector generator within the DTU must also be set up. The most desirable reference is Canopus, (SEN3) particularly for conscan operation near the sun line; however, the sun can be used (SEN2) if conscan does not cause loss of the sun within its deadzone (see Section 3.3.1.2.7). Note that SEN1 commands the sun sensor output (Sun A) which does not contain channel one. Since sun A has a 10° deadzone rather than the 1° deadzone of sun B, it cannot be used for conscan late in the mission. With Canopus as the reference, it is essential to lock onto the star. For this procedure see Section 4.3. The mode of the SPSG can be any of the three.

The conscan function cannot begin until enabled via the closed loop start command, CNS5. Should it become necessary, the conscan function can be stopped via CNS6, closed loop stop. Conscan should stop automatically by the conscan enable signal which comes from the conscan signal processor. This enable tests if the conscan has reached the deadzone threshold. Once this signal indicates pointing (even momentarily) within the deadzone, conscan will be terminated automatically and must be re-enabled via CNS5.

An enable signal must exceed threshold in order for this command to set the enable latch.

SUMMARY OF CONSCAN FUNCTION COMMAND REQUIREMENTS IN ACS

Start from ACS Standby Mode

Commands	
SLA 9 or SLB 9	DSL 1 or DSL 2 on
ACR 1	Clock reset
PRE 1	For 180° conscan reference
PRE 2	For 0° conscan reference
PUL 1	Use with high gain antenna
PUL 2	Use with medium gain antenna
SEN 2	Sun reference
SEN 3	For star reference. To be followed by commands required for Canopus lock (Sec. 4.3)
RIP 3	Average SPSG mode
RIP 5	ACS mode
	Verify threshold exceeded
CNS 5	Closed loop start

3.3.3 Attitude Control Subsystem Interface Signals

In the following tables and figures, the input and output interface signals from ACS are summarized. They are presented in five subsections covering the CEA reference signals, commands, thruster nominal performance, storage register command--telemetry correlation, and the annotated telemetry list.

3.3.3.1 CEA Reference Signals

INPUT SIGNAL LINE	INPUT SOURCE	USE
SUN A SUN B	Sun Sensor Assembly	SPC - sensor select then to DTU-SPSG
Star A (star pulses) Star B (discrete)	Stellar Reference Assembly	SPC - star delay star time gate star level logic star coincidence gate star count
DSA A DSA B	Despin Sensor Assemblies	DSL - despin logic DSL A & B
Conscan A (pulses) Conscan B (threshold)	Conscan Signal Precessor	Conscan Logic
Commands Data Input - CDU	Command Distribution Unit	PSE SPC (see command list) DSL PSE register arming logic
Time frequency A (32, 768 Hz) Time frequency B (32, 768 Hz) (AFA = 1 ppr) Angle frequency A (1 ppr) Angle frequency B (8 ppr) (AFB = 8 ppr) Angle frequency C 512 ppr (AFC = 512 ppr) Word gates (23) Telemetry Clock Data Clock	Digital Telemetry Unit	SPC - clock countdown then PSE seq. control logic DSL - clock selection logic Sequence control logic PSE - TLM control and output
+5 Vdc Line +28 Vdc Line	Power Subsystem	5 Volts to PSE SPC DSL 28 volts to DSL valve drivers.

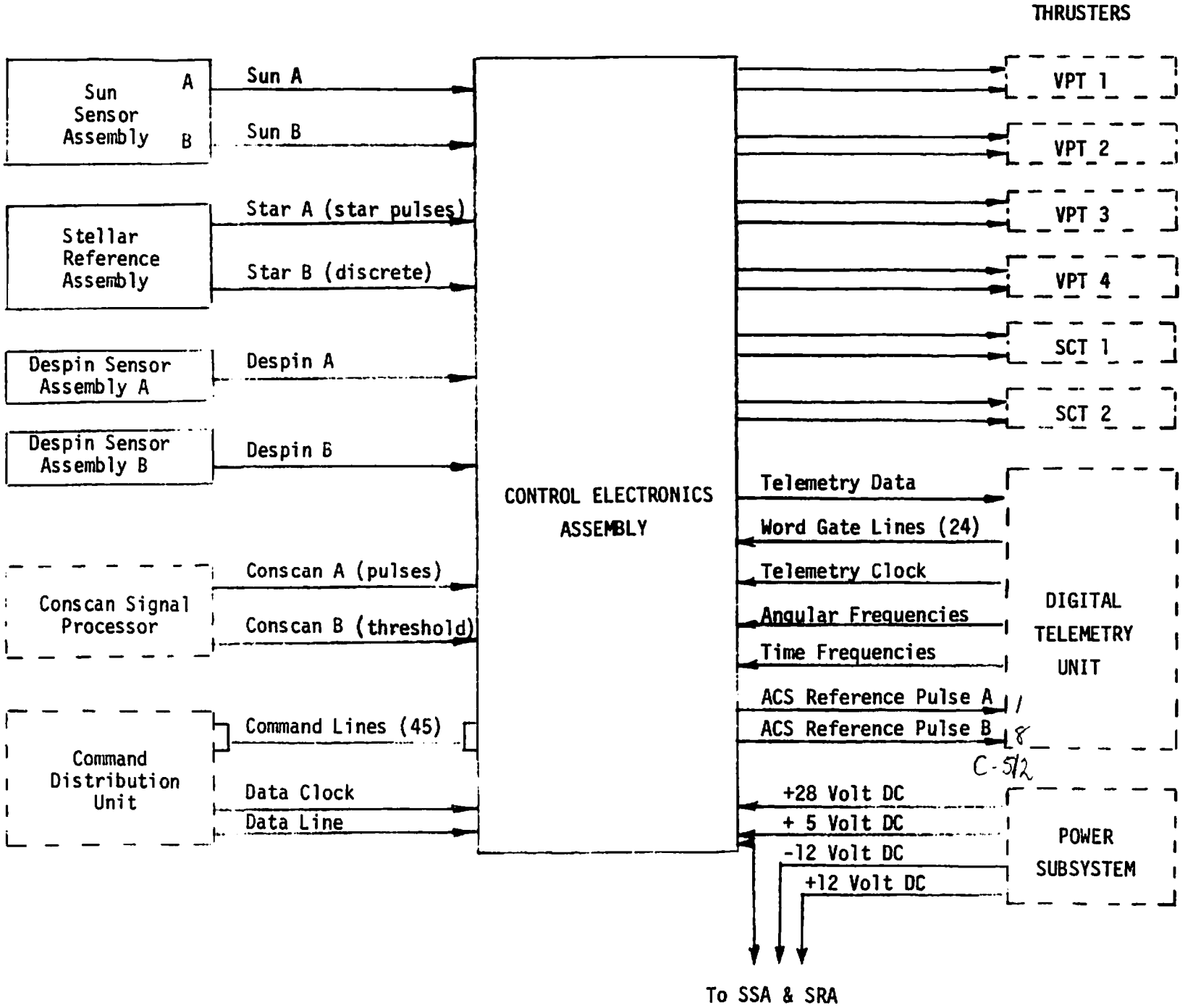


Figure 3.3-36. CEA Reference and Interface Signals

3.3.3.2 CEA Command List

Symbol	Octal Code	Command	Function of Command	Quantization	Comment
PSE 9	166	ACS Power Standby	Supplies power to PSE subassembly and removes power from both DSL's		
SLA 9*	036	DSL-A ON	Supplies power to DSL-A subassembly and removes power from DSL-B.		
SLB 9*	047	DSL-B ON	Supplies power to DSL-B subassembly and removes power from DSL-A.		
PSE 0	025	PSE-OFF	Removes power from PSE subassembly		
ACS 1	233	Arm Register 1	Enables entry of data into first precession registers for 4 data words (32 bits).		
ACS 2	244	Arm Register 2	Enables data entry into ΔV registers for 4 data words (32 bits)		
ACS 3	255	Arm Register 3	Enables data entry into second precession registers for 4 data words (32 bits).		
ACX 1	222	Sequence Execute	Starts program sequence.		Caution: See Section 3.3.1.3.1 for restrictions on ACX 1-5 commands
ACX 2**	345	Sequence Inhibit	Inhibits continuation of program sequence; provides two readouts of serial telemetry.		
ACX 3	010	Sequence Override	Overrides ACX 2; restarts sequence at point inhibited.		
ACX 4	101	Sequence Reset	Returns sequence to initial state; provides two readouts of serial telemetry.		

* Should be followed by ACR 1 - clock reset
** Should be followed by either sequence override ACX 3 (to continue) or sequence reset ACX 4 (to start over)

Symbol	Octal Code	Command	Function of Command	Quantization	Comment
ACX 5	042	Sequence Step	Advances stored program sequence counter to next state.		
PUL 1*	022	Pulse length 1	Selects 31.2 msec pulse length	0.08°	PUL 1-5 are latched. Full tank assumed for quantization.
PUL 2*	213	Pulse length 2	Selects 125 msec pulse length	0.32°	
PUL 3*	143	Pulse length 3	Selects 0.5 sec pulse length	1.28°	
PUL 4*	301	Pulse length 4	Selects 1.0 sec pulse length	2.56°	
PUL 5*	055	Pulse length 5	Selects 2.0 sec pulse length	5.12°	
PRE 1	266	Precession Pair 1	Selects "positive" (+X torque direction) precession thruster pair (VPT 2, 3).	+	Latched
PRE 2	322	Precession Pair 2	Selects "negative" (-X torque) prevession thruster pair (VPT 1, 4).	-	Latched
PRE 3	230	0° precession pulse	Initiates single precession step at α= 0		One pulse per command on PRE 3-7
PRE 4	241	90° precession pulse	Initiates single precession step at α= 90°		
PRE 5	252	180° precession pulse	Initiates single precession step at α= 180°		
PRE 6	263	270° precession pulse	Initiates single precession step at α= 270°		
PRE 7	274	Real time precession pulse	Produces single real time precession step.		

* See tables Section 3.3.3.1

Symbol	Octal Code	Command	Function of Command	Quantization	Comment
VEL 1	277	ΔV pair 1	Selects aft thruster pair for "positive" ΔV. (VPT 2, 4) (Acceleration in +Z direction).	+	Latched
VEL 2	333	ΔV pair 2	Selects fore thruster pair for "negative" ΔV. (VPT 1, 3) (Acceleration in -Z direction).	-	Latched
VEL 3	205	Real time ΔV pulse	Produces single ΔV increment		
VEL 4	216	ΔV SCT Mode Enable	Enables ΔV step perpendicular to spin axis at S/C orientation angle.		ΔV/SCT requires program.
SPN 1	330	Despin Start	Initiates despin operation (Selects SCT No. 2)		
SPN 2	227	Real time spin pulse	Produces single spin pulse		
SPN 3	224	Spin direction up	Selects SCT 1 for increasing spin rate	+	Latched
SPN 4	312	Spin direction down	Selects SCT 2 for decreasing spin rate	-	Latched
SEN 1	303	Sensor Select, Sun A	Selects A output of sun sensor for reference		Latched - no output in channel 1
SEN 2	314	Sensor Select, Sun B	Selects B output of sun sensor for reference		Latched
SEN 3	325	Sensor Select, Star	Selects star sensor output for reference		Latched

Symbol	Octal Code	Command	Function of Command	Quantization	Comment
SAS 1	256	Star Angle Gate Enable	Enables star pulses for selected angle and disables time gate		Latched
SAS 2	334	Star Angle Gate Disable	Permits entire 360° star scan output and disables time gate		Latched
SAS 3	245	Star Delay Up Lower Half	Adds one increment to lower half of star pulse	1/256 sec	3.906 msec = 1/256
SAS 4	373	Star Delay Up Upper Half	Adds one increment to upper star time delay register	0.25 sec	
SAS 5	323	Star delay down lower half	Subtracts one increment from lower half of star pulse time delay register	1/256 sec	3.906 msec = 1/256
SAS 6	337	Star delay down upper half	Subtracts one increment from upper star time delay register	.25 sec	
SAT 1	340	Star Time Gate Enable	Blanks out star pulses during time gate and inhibits angle gating		Advances star Latch (section 3.3.1.5).
SAT 2	351	Star time gate up	Adds one increment to the star time gate register	0.25 sec	
SAT 3	362	Star time gate down	Subtracts one increment from the star gate register	0.25 sec	
ACA 1	336	Time frequency Select 1	Selects first DTU 32,768 Hz clock output		See section 3.3.1.4.2
ACB 1	347	Time frequency Select 2	Selects second DTU 32,768 Hz clock output		
ACR 1	206	Clock reset	Resets selection of ACA 1 or ACB 1 commands		See section 3.3.1.4.2
CNS 5	352	Closed loop start	Enables conscan pulse to enter DSL		Latch
CNS 6	363	Closed loop stop	Disables conscan pulse inputs		

3.3.3.3 Thruster Command Signals Thruster Nominal Performance Summary

Pulse Length (sec)	0.0312	0.125	0.500	1.00	2.00
Selection Command	PUL 1	PUL 2	PUL 3	PUL 4	PUL 5
Pulse length uncertainty (sec) (random, uniform variation)	.000031	.00012	.0005	.001	.002
Precession Magnitude (Full Tank)					
PRE 1, +X Torque Selection PRE 2, -X Torque Selection PRE 7, Execute, Real Time	0.08°	0.32°	1.28°	2.56°	5.12°
Fixed Angle Precession Direction					
PRE 3, Execute, 0°	1.3°	2.56°	8.06°	15.26°	29.68°
PRE 4, 90°	91.3°				
PRE 5, 180°	181.3°				
PRE 6, 270°	271.3°				
ΔV Magnitude					
VEL 1, Selects +Z acceleration VEL 2, Selects -Z acceleration VEL 3, Execute, Real Time VEL 4, Selects V/SCT Mode: Program required in PSE	0.0011 m/sec	0.0045 m/sec	0.018 m/sec	0.036 m/sec	0.072 m/sec
Spin					
SPN 1, Starts Despin, ω > 22 rpm SPN 2, Execute, Real Time SPN 3, Selects $+\Delta\omega$ SPN 4, Selects $-\Delta\omega$	0.0032 rpm	0.0128 rpm	0.0514 rpm	0.103 rpm	0.205 rpm

3.3.3.4 Precession--ΔV Register Command and Telemetry Correlation

Precession 1 and 2	Magnitude (1/8 Hz)	Redundant Magnitude (1/64 Hz)	Angle (512 ppr)
WEIGHT	X 1 2 4 8 16 32 64 X 128 256 512 1024	1 2 4 X 8 16 32 64 128	1 X 2 4 8 16 32 64 128 256
COMMAND	1 2	3	4
Bit	1 2 3 4 5 6 7 8 1 2 3 4 5	6 7 8 1 2 3 4 5 6	7 8 1 2 3 4 5 6 7 8
TELEMETRY			
(Prec. 1) Word	C_{411}	C_{428}	C_{429}
Bit	1 2 3 4 5 6 C_{412} 1 2 3 4 5	2 3 4 5 6 C_{429} 1 2 3	4 5 6 C_{430} 1 2 3 4 5 6
TELEMETRY			
(Prec. 2) Word	C_{415}	C_{422}	C_{424}
Bit	1 2 3 4 5 6 C_{416} 1 2 3 4 5	6 1 2 3 4 5 6 C_{424} 1	2 3 4 5 6 1 2 3 4

ΔV	Magnitude (1 Hz)	Redundant Magnitude (1/32 Hz)	Delay (1/64 Hz)
WEIGHT	X 1 2 4 8 16 32 64 X 128 256 512 1024 2048 4096	1 X 2 4 8 16 32 64 128	X 1 2 4 8 16 32 64
COMMAND	1 2	3	4
Bit	1 2 3 4 5 6 7 8 1 2 3 4 5 6 7	8 1 2 3 4 5 6 7 8	1 2 3 4 5 6 7 8
TELEMETRY			
Word	C_{412} C_{413}	C_{425} C_{426}	C_{427}
Bit	6 1 2 3 4 5 6 C_{414} 1 2 3 4 5 6	5 6 1 2 3 4 5 6	1 2 3 4 5 6 C_{428} 1

X = bit stripped from input data - input as zero

3.3.3.5 ACS Telemetry

TELEMETRY SIGNAL	DESCRIPTION	WORD	BIT NO.	STATE	COMMAND	COMMENT
Precession Pair	Pair 1 (2 & 3) Pair 2 (1 & 4)	C-403	1	0 1	PRE 1 PRE 2	Precession in +X Direction -X
Pulse Length	0.03125 0.125 0.500 1.000 2.000	C-403	2 3 4	001 011 111 110 100	PUL 1 PUL 2 PUL 3 PUL 4 PUL 5	
ΔV Pair	Pair 1 (2 & 4) Pair 2 (1 & 3)	C-403	5	0 1	VEL 1 VEL 2	ΔV in +Z direction -Z
Spin Control Direction	Spin Up Spin Down	C-403	6	0 1	SPN 3 SPN 4	
Despin Status	Despin Start Despin Complete	C-410	1	1 0	SPN 1	Sets Latch in DSL Spin < 22 rpm
Conscan Enable	Conscan Start Stop	C-410	2	1 0	CNS 5 CNS 6	
Clock Select	DTU Clk A Select Reset DTU Clk B Select Reset	C-410	3 4 	0 1 0 1	ACA 1 ACR 1 ACB 1 ACR 1	
Angle Gate Enable Status	Disabled Enabled	C-410	5	0 1	SAS 2 SAS 1	Angle gate 360° wide Angle gate 45° wide
Star Level	Discrete	C-410	6	0 1	-	Reset at end of TLM word C-430 Star > 180% Canopus

TELEMETRY SIGNAL	DESCRIPTION	WORD	BIT NO.	STATE	COMMAND	COMMENT
Star Delay Max Delay = 15.9961 sec	Lower half LSB Increment Up Down Upper Half LSB Increment Up Down	C-419 C-420	1-6 1-6	1 = Bit Present 1 = bit Present	 SAS 3 SAS 5 SAS 4 SAS 6	12 bit up-down binary counter. Bit 1 is LSB. Quantization = $\frac{1}{256}$ sec. Uses 6 bits of word C-419 Second (Upper) Half of 12 Bit Binary Counter. Bit 1 of Word C-420 is LSB with Quantization 1/4 sec. Uses 6 bits of Word C-420
Star Count	Zero Stars 1 Star 2 3 4 5 6 7	C-421	1 2 3	000 100 010 110 001 101 011 111	-	Binary Counter Reset on 315° Pulse
Power Status	DSL-A On Off DSL-B On Off PSE On Off	C-421	4 5 6 	1 0 1 0 1 0	SLA 9 PSE 9 SLB 9 PSE 9 PSE 9 PSE 0	
Star Location	Octant 1 Star No Star Octant 8 Star No Star	C-432	1 2 	 1 0 1 0	-	Reset on 270° Pulse Star in Octant 1 Star in Octant 8 One's in Both Bits 1 & 2 is extremely unlikely.

TELEMETRY SIGNAL	DESCRIPTION	WORD	BIT NO.	STATE	COMMAND	COMMENT
Inhibit State	Sequence Start Inhibit Continue Reset	C-432	3	 1 0 1 0	 ACX 1 ACX 2 ACX 3 ACX 4	Use ACX 1 only to start Return to Sequence State 0
ARM Status	Register 1 2 3	C-432	4 5 6	100 010 001	ACS 1 ACS 2 ACS 3	
Sequence Status	State: 0 Delay 1 1 Prec. 2 Delay 3 ΔV Delay 5 2 Prec. 6 7	C-431	1 2 3	000 100 010 110 001 101 011 000	ACX 4 ACX 1 Change State via ACX 5	Note Warnings on Use of Commands in PSE Description, Ref: Section 3.3.1.3 All one's in this TLM indicates improper program operation from ground. Program complete - not a true zero state.
Time Gate Enable Status	Enable Disable	C-431	4	0 1	SAT 1 SAS 2 SAS 1	Inhibits Angle Gate Opens Angle Gate to 360° Opens Angle Gate to 45°
Reference Select	Sun A Sun B STAR	C-431	5 6	1 1 1 0 0 1	SEN 1 SEN 2 SEN 3	
#1 Precession Magnitude Register	See Table 3. Section 3.3.3.2	C-411 C-412	1-6 1-5	1 = Bit Present	ACS 1 to arm	Bit 1 of Word C-411 is LSB. Register is Binary Counter, with LSB Quantization = 8 sec. 11 Bit Register

TELEMETRY SIGNAL	DESCRIPTION	WORD	BIT NO.	STATE	COMMAND	COMMENT
ΔV Magnitude Register	See Table 3.	C-412 C-413 C-414	6 1-6 1-6	1 = Bit Present	ACS 2 to Arm	Word C-412 Bit 6 is LSB of Binary Counter with Quantization = 1 sec.
#2 Precession Magnitude Register	See Table 3.	C-415 C-416	1-6 1-5	1 = Bit Present	ACS 3 to Arm	Word C-415 Bit 1 is LSB of Binary Counter with Quantization = 8 sec. 11 Bit Register
Star Coincidence	Coincidence No Coincidence	C-416	6	1 0	-	Delayed Star and Incoming Star Pulse from SRA Occur within 1/16 sec. Reset by word C-430 trailing edge.
Star Time Gate	No. of Bits in Time Gate: Example: Normal Gate Up Gate Down	C-404	12345	 11010* 00110 01010		Binary Counter: Bit 1 is LSB Bit 5 is next MSB, Bit 6 is MSB and is always 1 (not on TLM) Quantization = 1/4 sec *Example = 11 x .25 = 2.75 sec + 8 sec = 10.75 sec blanking period.
ΔV/SCT Status	ΔV/SCT Mode Enable Disable	C-404	6	 1 0	 VEL 4 PSE 0 or ACX 4	
Spacing Bits		C-422	1-5			All Zeroes
#2 Redundant Precession Mag Register	See Table 3.	C-422 C-423 C-424	6 1-6 1	1 - Bit Present	ACS 1	Word C-422 Bit 6 is LSB of Binary Counter. Bit 1 of Word C-424 is MSB. Quantization = 64 sec. 8 Bit Register

TELEMETRY SIGNAL	DESCRIPTION	WORD	BIT NO.	STATE	COMMAND	COMMENT
#2 Precession Angle Register	See Table 3.	C-424 C-425	2-6 1-4	1 = Bit Present	ACS 3	Word C-424 Bit 2 is LSB of Binary Counter. Bit 4 of Word C-425 is MSB. Quantization = .703125°. 9 Bit Register.
ΔV Redundant Mag. Register	See Table 3	C-425 C-426	5, 6 1-6	1 = Bit Present	ACS 2	Word C-425 Bit 5 is LSB of Binary Counter. Bit 6 of word C-426 is MSB. Quantization = 32 sec. 8 Bit Register
Delay Register	See Table 3	C-427 C-428	1-6	1 = Bit Present	ACS 2	Word C-427 Bit 1 is LSB of Binary Counter. Bit 1 of Word C-428 is MSB. Quantization = 64 sec. 7 Bit Register
#1 Redundant Precession Mag. Register	See Table 3.	C-428 C-429	2-6 1-3	1 = Bit Present	ACS 1	Word C-428 Bit 2 is LSB of Binary Counter. Bit 3 of Word C-429 is MSB. Quantization = 64 sec. 8 Bit Register
#1 Precession Angle	See Table 3.	C-429 C-430	4-6 1-6	1 = Bit Present	ACS 1	Word C-429 Bit 4 is LSB of Binary Counter. Bit 6 of Word C-430 is MSB. Quantization = .703125°. 9 Bit Register
SSA Temperature	Analog	C-317	-			Range = TBD
SRA Temperature	Analog	C-303				Range = TBD

SECTION 3.3 NOTATION

ACS	Attitude Control Subsystem
AFA	Angle Frequency A = 1 ppr
AFB	Angle Frequency B = 8 ppr
AFC	Angle Frequency C = 512 ppr
α_R	Desired angle α to be entered in program angle register
AU	Astronomical Unit (93×10^6 miles)
CDU	Command Distribution Unit (Described in Section 3.9)
CEA	Control Electronics Assembly
CMD	Command
CSP	Conscan Signal Processor (Described in Section 3.6)
Deg	Degrees
ΔV/SCT	Mode for executing a ΔV normal to the spin axis by firing the spin control thrusters (SCT's) alternately
DSA	Despin Sensor Assembly
DSL	Duration & Steering Logic, part of CEA
DTU	Digital Telemetry Unit (Described in Section 3.5)
FOV	Field of View
Hz	Hertz, cycles per second
KHz	KiloHertz, cycles per second
LSB	Least Significant Bit
m	millisecond
MR	Master Reset - occurs if voltage on 5 volt line drops below 3.5v.
MSB	Most Significant Bit
m/sec	meters per second
msec	millisecond
μsec	Microsecond
n	number of counts in angle counter
PK-PK	Peak-to-Peak
ppr	Pulses per revolution of the spacecraft
PSE	Program Storage & Execution, part of CEA
ψ	total angular length of precession measured along precession path

SECTION 3.3 NOTATION (CONT'D)

R	R axis; parallel to +y axis, defines the sun sensor pointing direction
rpm	revolutions per minute
R-S	Reset - Set latch circuit
RSS	Square root of sum of squares
RTG	Radioisotope Thermoelectric Generator (Described in Section 3.8)
S	S axis; lies in x, y plane, -25° from +y axis. Defines SRA pointing direction.
SCT	Spin Control Thruster - line of action parallel to z axis.
SNR	Signal to Noise Ratio
SO-S7	State Number - Sequence of 8 states for stored program execution, page 3.3-7.
SPC	Sensor & Power Control, part of CEA
SPSG	Spin Period Sector Generator (Described in Section 3.5)
SRA	Stellar Reference Assembly
SSA	Sun Sensor Assembly
TFA	Time Frequency A (32,768 Hz Clock A)
TFB	Time Frequency B (32,768 Hz Clock B)
TLM	Telemetry
UVD	Undervoltage Detector
vdc	Volts, Direct Current
VPT	Velocity Precession Thruster - line of action parallel to z axis.

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3.4 PROPULSION SUBSYSTEM

3.4.1 General Description

3.4.1.1 Equipment, Configuration

3.4.1.1.1 Subsystem Installation. The propulsion subsystem is a monopropellant system using a hydrazine fuel reacting spontaneously with a catalyst to produce hot gas which is expended through six individually controlled thruster nozzles to affect spacecraft maneuvers. The subsystem consists of the following major assemblies which are installed on the spacecraft as shown in Figure 1.

- Propellant Supply Assembly (PSA)
- Two Line Heater Assemblies (LHAs)
- Three Thruster Cluster Assemblies (TCAs)

The PSA is aligned with the spacecraft Z axis and is entirely contained within the equipment compartment. It provides the storage tank for the hydrazine fuel and gaseous nitrogen pressurant used for expulsion of the fuel in a blowdown mode during thruster firings. The two LHAs, which are routed external to the equipment compartment to the +Y and -Y axes TCA mounting platforms, supply fuel from the PSA to the TCAs.

One TCA, located on the -Y platform, contains a thruster pair with nozzles aligned to the +X and -X axes. Fired singly, these thrusters provide initial despin after spacecraft separation from the third stage. Thereafter, they are used for periodic spacecraft spin rate adjustments about the Z axis.

The two remaining TCAs, mounted on the +Y and -Y platforms respectively, each contain thruster pairs with nozzles aligned to the +Z and -Z axes. Fired in opposition, these thrusters provide a moment about the X axis with resultant precession of the spacecraft about the Y axis. Spacecraft velocity gains are accomplished by firing thruster pairs in either the +Z or -Z direction after precession to the desired maneuver attitude.

3.4.1.1.2 Subsystem Assemblies

3.4.1.1.2.1 Propellant Supply Assembly. The PSA illustrated in Figure 2 consists of the following components.

- Propellant and Pressurant Tank
- Pressure Transducer
- Filter
- Temperature Transducer

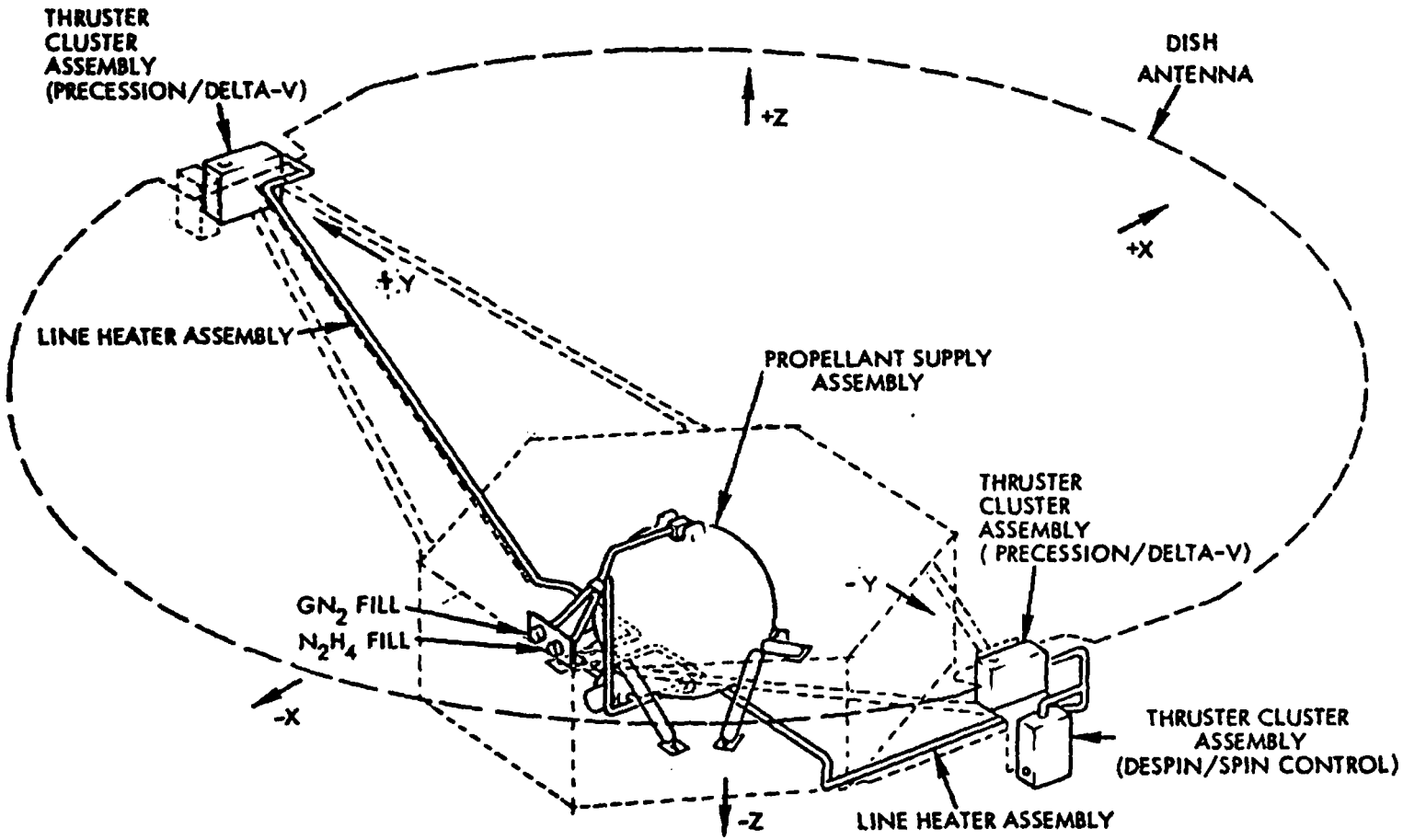


FIGURE 3.4-1 PIONEER F/G PROPULSION SUBSYSTEM

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Figure 3.4-2
Pioneer F/G Propellant Supply Assembly

- Fill and Drain Valves
- Interconnecting Lines

The propellant and pressurant tank is a 2300 cubic inch volume titanium sphere containing an elastomeric diaphragm which is attached to the inner wall of the tank near the girth weld which joins the two tank hemispheres during fabrication (see Figure 3). The tank stores hydrazine fuel on one side of the diaphragm, and contains the gaseous nitrogen used to expel the fuel on the other side.

The tank is loaded with a maximum of 60 lb of fuel through the hydrazine fill port. At least 99 percent of the fuel loaded may be expelled during the mission. Depending on the final definition of spacecraft mission duty cycle requirements, the tank may be off loaded providing less than the maximum fuel capacity.

The gaseous nitrogen pressurant is loaded through the GN_2 fill port. The loaded pressure will depend on the amount of fuel loaded, but will generally range from 460-550 psia within the 40-110°F allowable temperature range with the 60 lb maximum capacity. The pressure and temperature transducers located in the tank propellant outlet line provide a means for telemetry monitoring of subsystem blowdown status and estimating propellant utilization during the mission. A propellant filter in the line is designed to filter out all solid particles larger than 15 microns to preclude TCA contamination and possible subsequent failure.

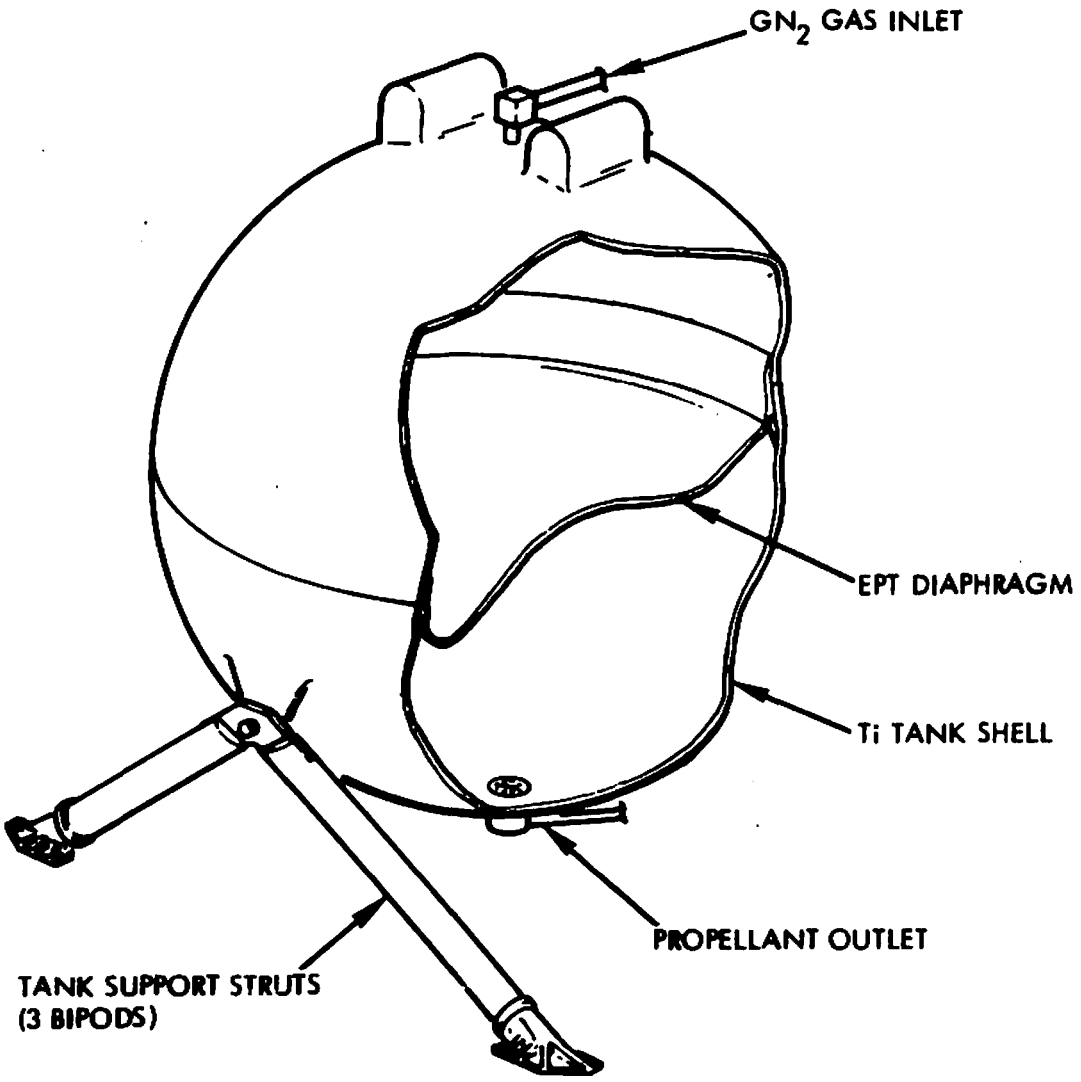
3.4.1.1.2.2 Thruster Cluster Assembly. All three TCAs used for spacecraft spin/despin, precession and ΔV maneuvers are identical in configuration. The typical TCA shown in Figures 4 and 5 consists of the following components.

- Two thrusters with 90 degree nozzles aligned in opposition to each other.
- Two propellant valves.
- Three one-watt radioisotope heater units.
- Two pressure switches (one for each thruster).
- Three temperature transducers.
- One baseplate.
- Upper and lower heat shields.

Fuel enters the TCA through a single inlet line to a manifold block which divides the flow to the individual thrusters. Fuel inlet temperature is monitored via telemetry by a thermistor surface mounted on the manifold. On-off control of flow to each thruster is provided by the solenoid operated propellant valve which contains redundant coils and seats.

PROPELLANT TANK ASSEMBLY

FIGURE 3.4-3



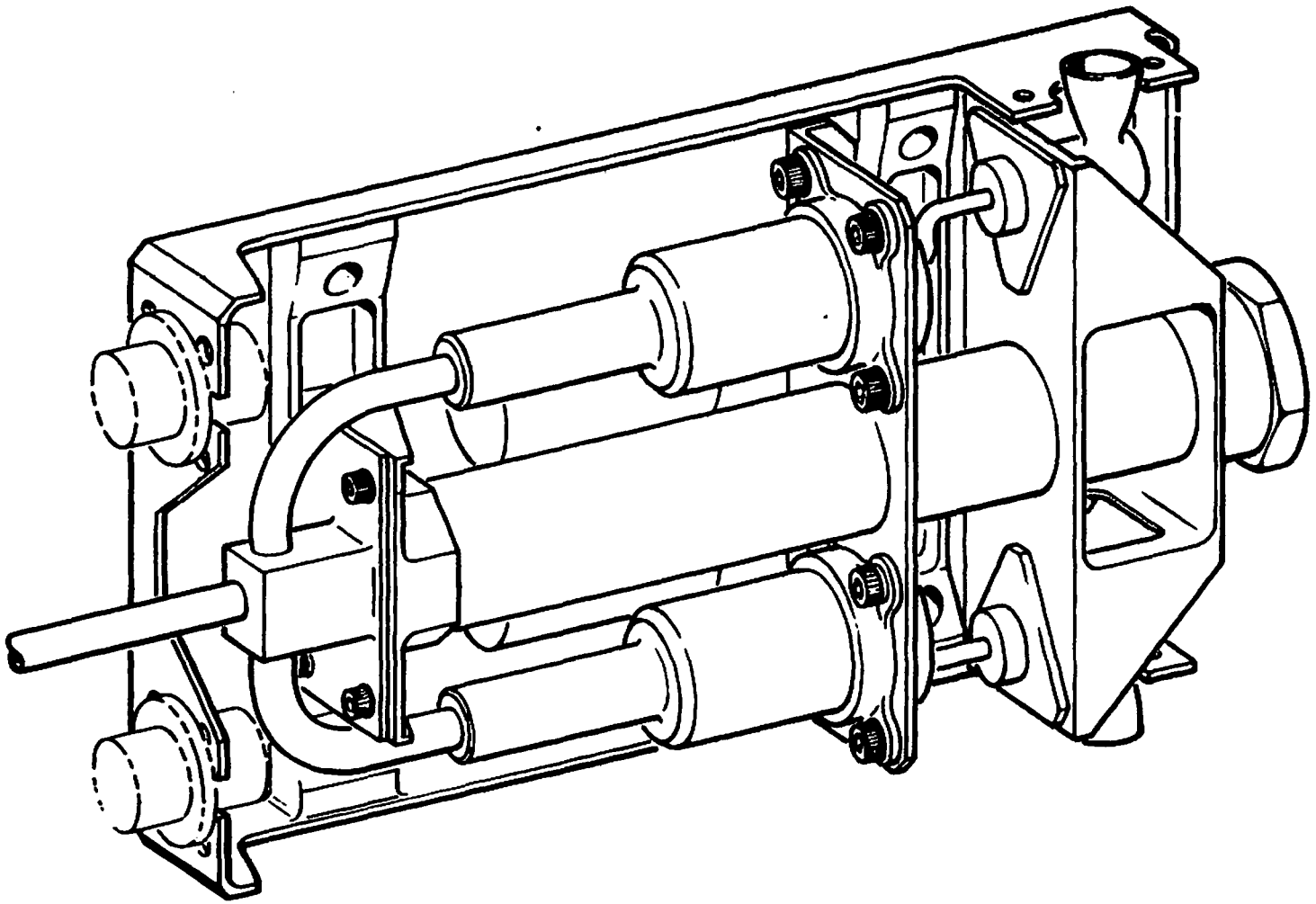


FIGURE 3.4-4 THRUSTER CLUSTER ASSEMBLY

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Figure 3.4-5
Thruster Cluster Assembly

The three radioisotope heater units (RHUs) provide the thermal energy necessary to prevent freezing of the hydrazine in the TCA during periods of non-operation. Two resistance type temperature transducers, one each surface mounted on the thruster bodies near the catalyst chamber, provide telemetry indication of thruster temperatures. The two pressure switches, one per thruster, sense the buildup in chamber pressure at the initiation of a thruster pulse or steady-state firing and close an electrical circuit used to verify thruster operation. The switch returns to the open position upon thrust termination.

1.4.1.1.2.3 Line Heater Assembly. The two propellant line heater assemblies (LHA) have the dual functions of directing propellant flow from the PSA to the TCAs, and also maintaining the propellant about freezing. The LHAs (shown in Figures 6 and 7) consist of the following components:

- Propellant Line
- Two Electrical Heaters
- Four Thermostats

One of the heaters is used as a primary circuit and the other is a secondary redundant circuit. The primary (active) heater element controls the propellant temperature from 55-90°F and the secondary (standby) element is set to control between temperatures of 45-90°F. The heaters are controlled on/off by the four thermostats (two for each circuit) which are mounted on the propellant line near the PSA.

The entire LHA is wrapped with layers of aluminized mylar insulation. Three high thermal resistance standoffs made of fiberglass are provided to attach the LHA to platform struts to minimize thermal losses.

3.4.1.2 General Functional Operation

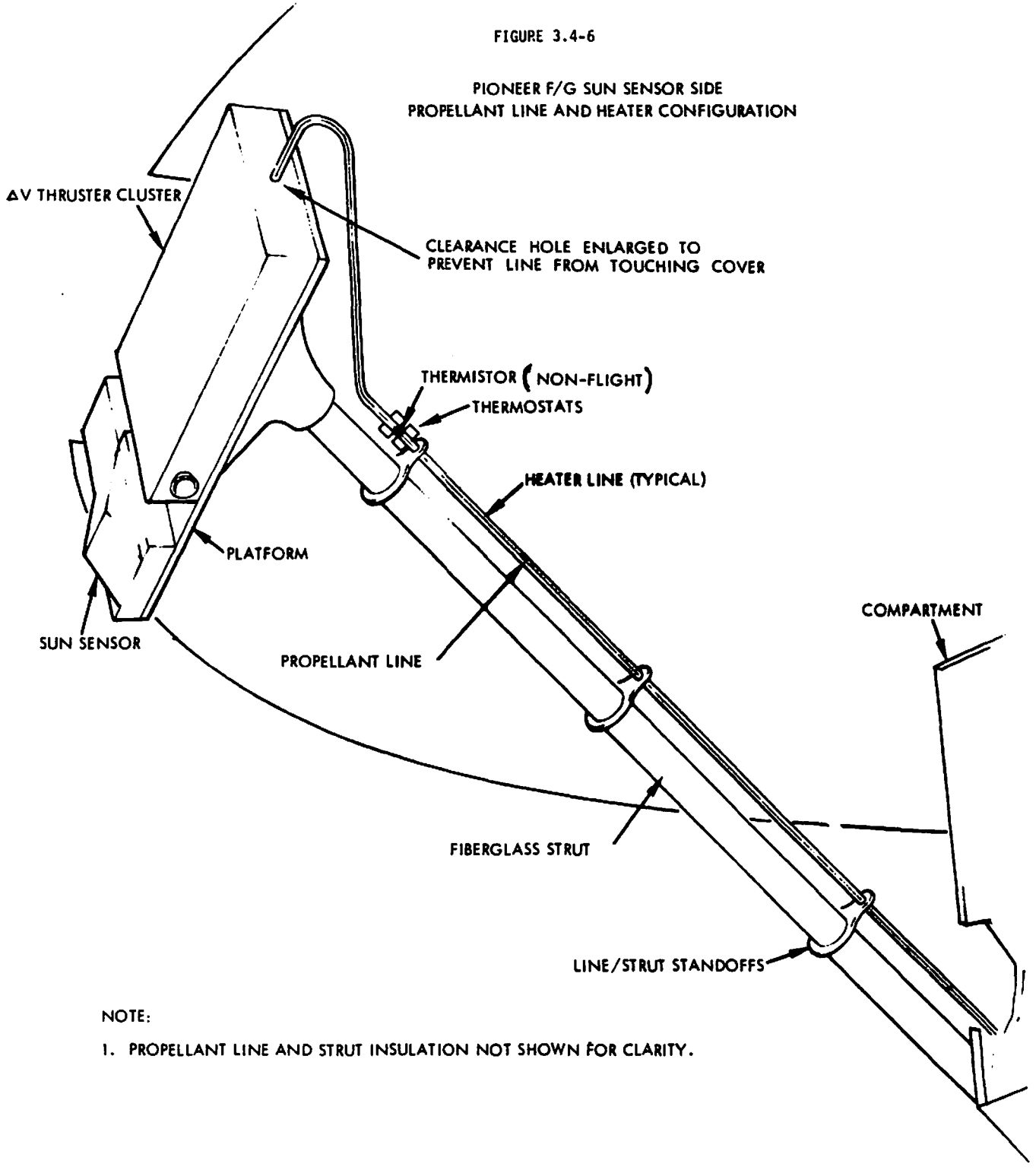
3.4.1.2.1 Operation Principles

3.4.1.2.1.1 Subsystem Mechanical Schematic and Mission Duty Cycle. The subsystem mechanical schematic is shown in Figure 3.4-8, and the nominal duty cycle operation is defined in the following paragraph. Propellant allocations for the various mission phases are summarized in Table 3.4-1.

Immediately following spacecraft separation from the third stage, the on board sequencer initiates a despin maneuver by commanding a steady-state firing of one of the spin/despin thrusters to slow the spacecraft spin rate from 60 $^{+2}_{-3}$ rpm. Upon receipt of the firing command, the energized propellant valve solenoid coil creates a magnetic force lifting the valve plunger off the redundant seats and permits hydrazine flow through the catalyst bed (see Figure 5). The resultant catalytic reaction with the hydrazine produces a hot gas of approximately 2000°F temperature which is expelled through a supersonic nozzle to produce thrust levels in the range of 0.4-1.2 lb depending on the fuel supply pressure. The firing is verified by closure of the pressure switch.

FIGURE 3.4-6

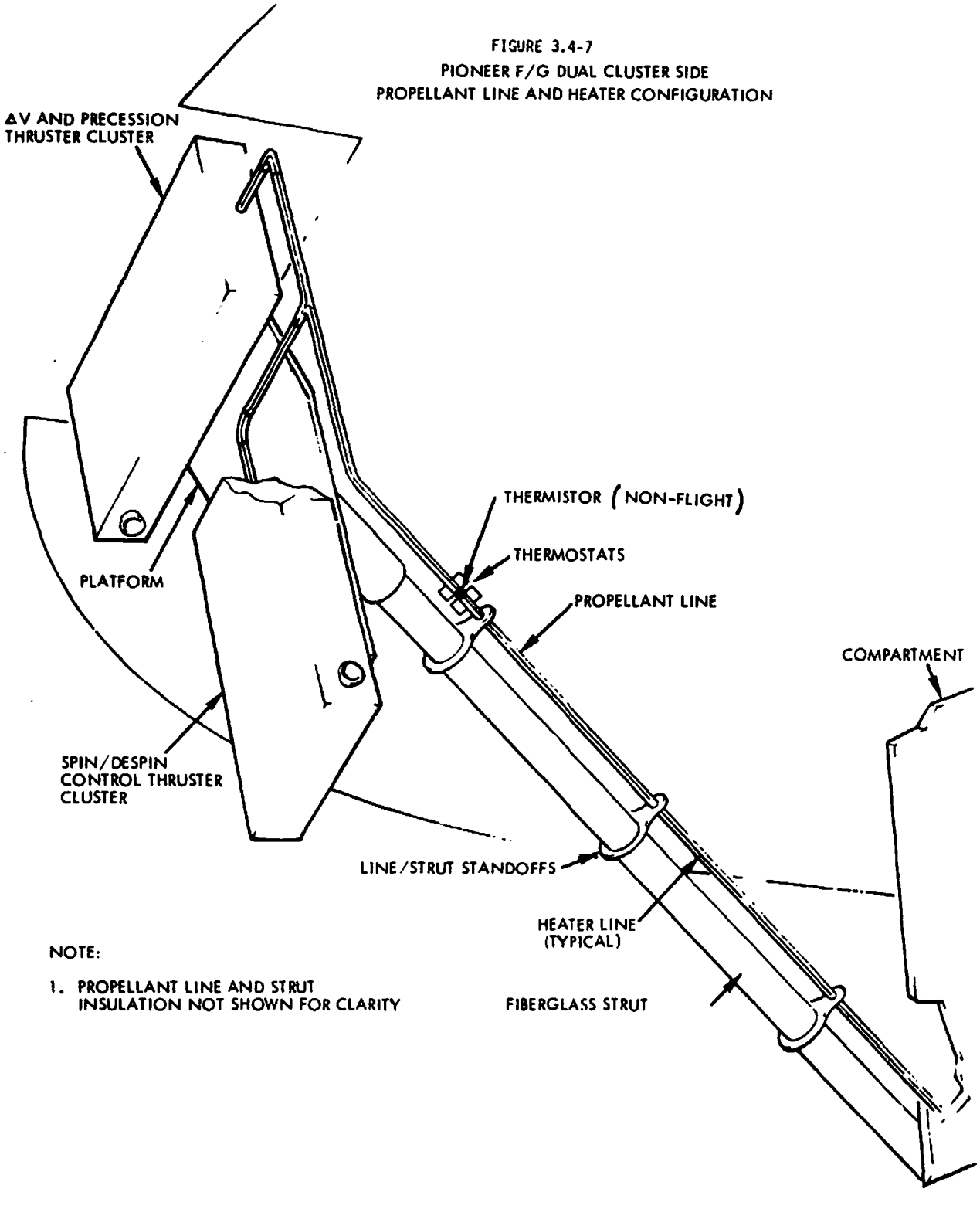
PIONEER F/G SUN SENSOR SIDE
PROPELLANT LINE AND HEATER CONFIGURATION



NOTE:

1. PROPELLANT LINE AND STRUT INSULATION NOT SHOWN FOR CLARITY.

FIGURE 3.4-7
PIONEER F/G DUAL CLUSTER SIDE
PROPELLANT LINE AND HEATER CONFIGURATION



NOTE:

1. PROPELLANT LINE AND STRUT INSULATION NOT SHOWN FOR CLARITY

FIGURE 3.4-8 PROPULSION SUBSYSTEM MECHANICAL SCHEMATIC DIAGRAM

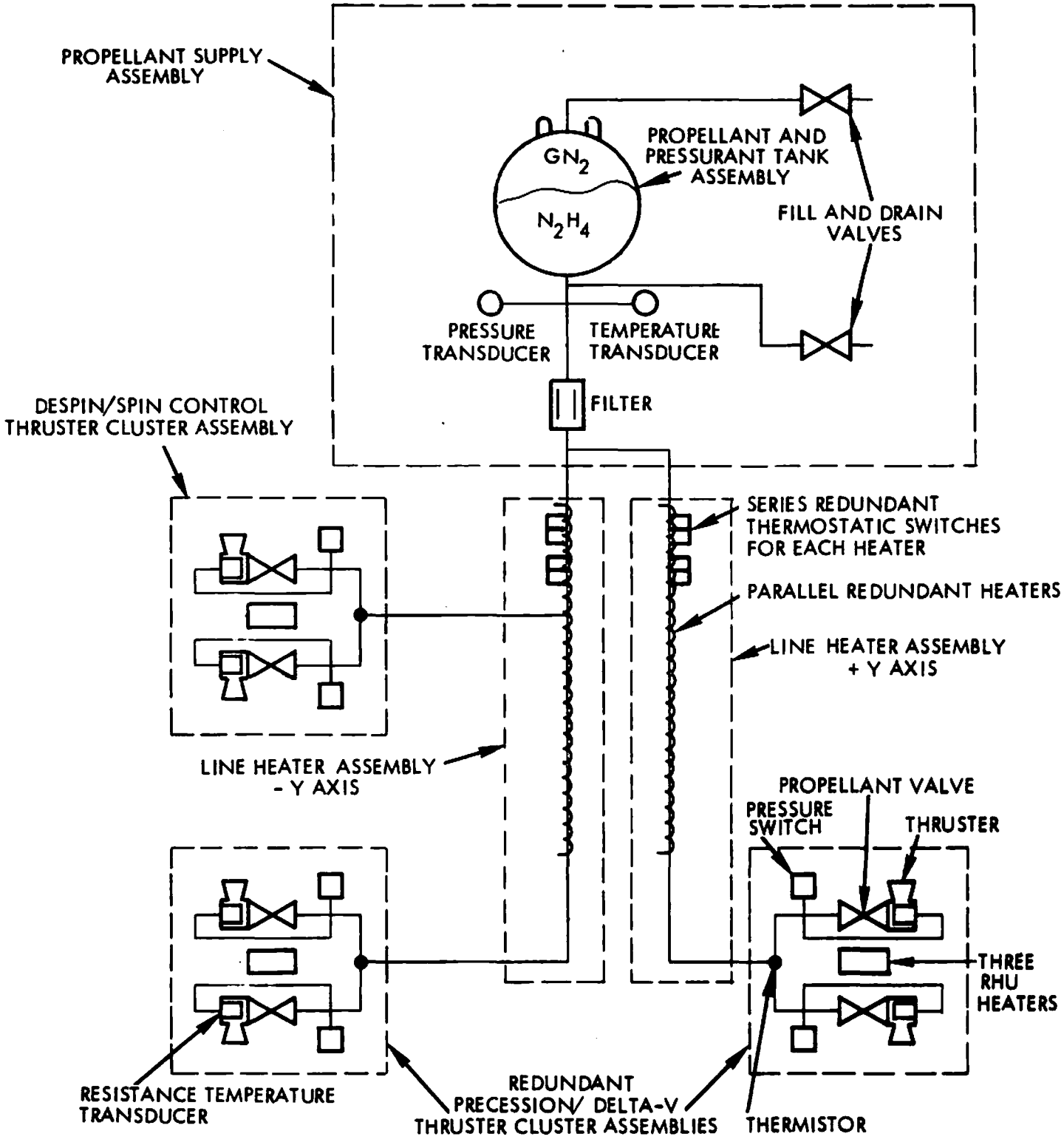


TABLE 3.4-1 PSS Propellant Allocations*

Maneuver	Thruster Mode	Average I_{sp}(sec)	Total Impulses(lb-sec)	Propellant Required (lb)
Despin (59 rpm)	Steady State	215	138	0.64
Precession (1250°)	Pulse	140	1130	8.08
Delta-V (200 m/sec)	Steady State	215	10550	49.08
Spin Control (14 rpm)	Pulse	140	154	1.10
Angular Calibration (90°)	Pulse	140	84	.60
Unavailable Propellant				
Trapped				.80
Leakage				.20
			Total	60.50

*Based upon S/C dry weight = 485 lb.

The despin sensors terminate the thruster firing when the spacecraft spin rate is slowed to 22 ± 1.5 rpm. Deployment of the RTG and magnetometer booms further slows the rate to a nominal 4.8 rpm by increasing the spacecraft inertia about the spin (Z) axis. Throughout the remainder of the mission, the spin rate is controlled within 4.8 ± 0.05 rpm by periodic pulse firings of the spin/despin thrusters. Sufficient propellant is on board to provide a total accumulated spin rate adjustment of 14 rpm (equivalent to 148 lb-sec delivered impulse).

The first spacecraft maneuver which utilizes the precession/delta V thrusters is the sequencer controlled initial orientation maneuver. This maneuver precesses the spacecraft from the initial orbital injection attitude to a near earth pointing position. The precession magnitude is expected to be 165 ± 15 degrees, and is accomplished by pulsing the selected thruster pair for 0.125 seconds duration once each spacecraft revolution (i.e., every 12.5 seconds at 4.8 rpm nominal spin rate) for a total maneuver duration of 104-125 minutes.

A nominal curve of delivered impulse as a function of the number of pulses in the pulse train is shown in Figure 9 at various fuel supply pressures. A typical blowdown curve of supply pressure as a function of temperature and fuel remaining is provided in Figure 10. More precise performance calibration data for each flight subsystem delivered will be provided in a separate document (Reference TBD). This document will characterize propulsion subsystem performance in terms of the performance of a nominal thruster in both steady state and pulsing operation throughout the blow-down pressure range. The blow-down characteristic will define system pressure during a steady state burn and at times subsequent to thruster shutdown until equilibrium pressurant temperature is reached. The characterization will include the performance influence factors of:

a) The deviation of an individual thruster's acceptance test data from the nominal.

b) The deviation of propellant temperature from the nominal (70°F).

c) The accumulation of operating time on the thruster.

In addition, the influence factor of a flight calibration may be incorporated as a modification of the influence effect of a) above.

Propellant consumption will be computed during the mission from initial N_2H_4 and GN_2 mass and by using the telemetry measurements of fuel pressure and temperature in conjunction with the blowdown curve, and an independent calculation is also performed using accumulated pulse count data from the pressure switches and total steady-state burn durations. The impulse delivered by the selected precession thruster pair may be determined from the precession angle obtained with the initial orientation maneuver, and a flight calibration bias factor applied to subsequent precession maneuvers which will use the same thruster pair unless a malfunction forces selection of the redundant pair.

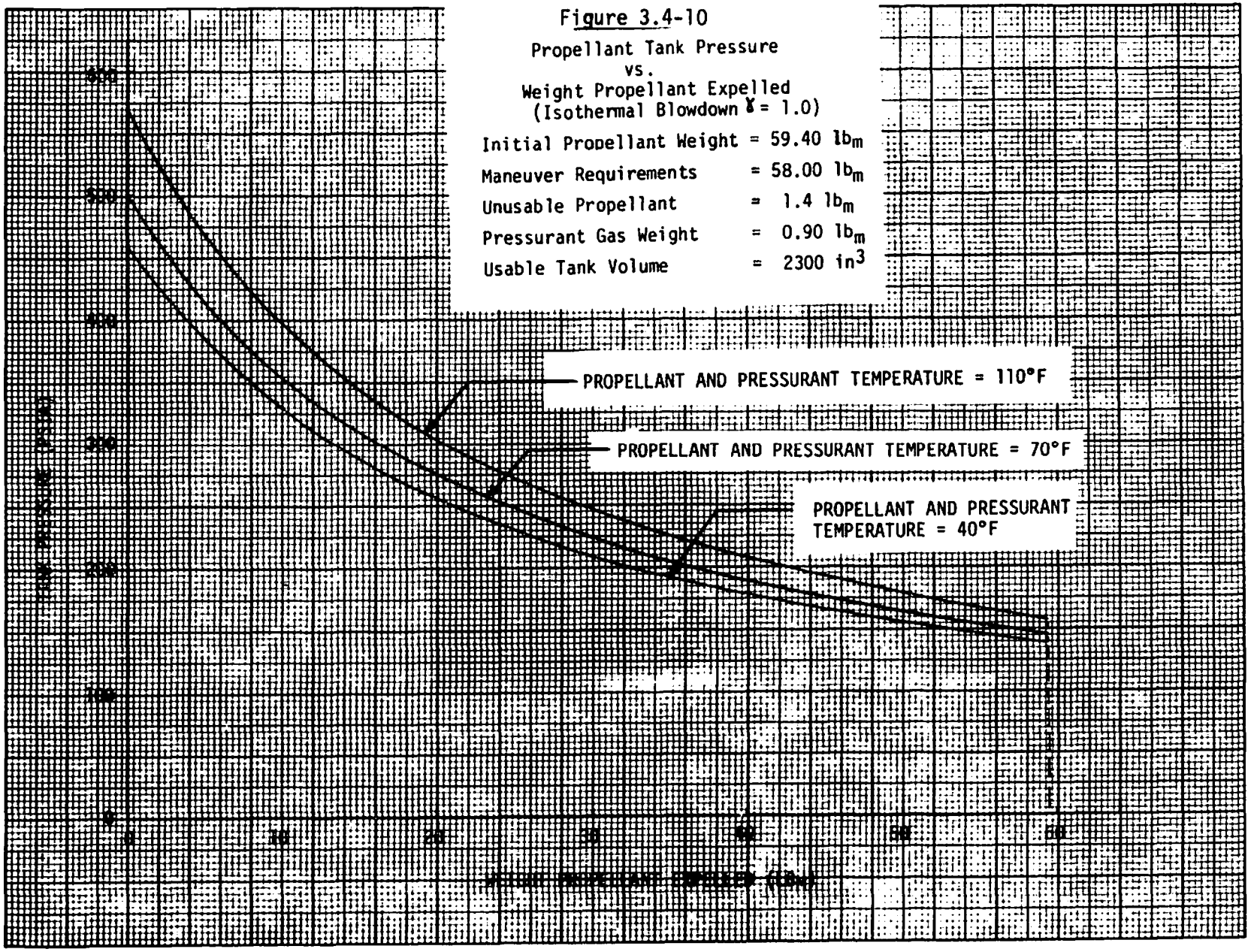
FIGURE 3.4-9 Propulsion Subsystem
Pulse Mode
Nominal Performance

(Final performance data to be supplied at completion of PSS qualification testing.)

Figure 3.4-10

Propellant Tank Pressure
vs.
Weight Propellant Expelled
(Isothermal Blowdown $\gamma = 1.0$)

Initial Propellant Weight = 59.40 lb_m
Maneuver Requirements = 58.00 lb_m
Unusable Propellant = 1.4 lb_m
Pressurant Gas Weight = 0.90 lb_m
Usable Tank Volume = 2300 in^3



After the first earth tracking data is obtained and analyzed, a small additional precession maneuver may be commanded from the ground to correct any errors in the sequencer controlled maneuver to achieve the desired earth pointing. This small maneuver will also be accomplished by 0.125 second precession thruster pulses. While still in this initial earth pointing attitude, a delta V maneuver consisting of a brief steady-state firing of the thruster pair planned for the first midcourse maneuver is tentatively planned. The purpose of this firing is to obtain an accurate inflight calibration through doppler shift to correct any biases in impulse predictions determined from ground testing before committing the midcourse burns. A typical curve of delivered impulse as a function of thruster burn duration and supply pressure is shown in Figure 11.

In order to maintain proper earth pointing attitude, closed loop precessions (conscan) are commanded periodically during flight. Pulse durations of 0.125 seconds will be used if starting from the medium-gain antenna with the pulse durations reduced to 0.03125 seconds for the high gain antenna operation.

A total of at least two, and perhaps three, midcourse maneuvers are planned during flight and will be performed within the first 50 days after launch. The normal maneuver is preceded and followed by the earth tracking maneuver previously described. Based on the desired spacecraft velocity vector addition to achieve proper orbit, the burn attitude and delta V magnitude will be calculated. The required precession angle, spacecraft spin rate and inertia are then used to compute the necessary impulse. The initial precession is expected to be from 0-107 degrees, and will be accomplished by 0.125 second duration pulses from the selected thruster pair once every revolution. The delta V will then be accomplished by a steady-state burn followed by a second precession and orbital reconstruction plus telemetry measurements of fuel pressure and temperature during the maneuvers will be used to determine the actual delivered impulse. A total of 1,136 lb-sec pulsing impulse is allocated to precession maneuvers (equivalent to a total of 1250 degrees), and 10,470 lb-sec steady-state impulse is provided for delta V (equivalent to 200 meters per second).

3.4.1.2.1.2 Subsystem Electrical Schematic. The PSS electrical schematic is shown in Figure 12, and a more detailed LHA thermostat control schematic is provided in Figure 13. The following PSS components are all operated with the 28 $\pm$ 0.56 VDC primary bus power.

a) Line Heater Assemblies

b) Thruster Propellant Valves

c) Propellant Supply Pressure Transducer

d) Thruster Pressure Switches

The thermistors used to monitor propellant supply and TCA inlet temperatures, and the resistance type temperature transducers used to measure individual thruster temperatures are all passive elements in bridge circuits which

FIGURE 3.4-11 Propulsion Subsystem Steady-State Mode Nominal Performance

(Final performance data to be supplied at completion of PSS qualification testing.)

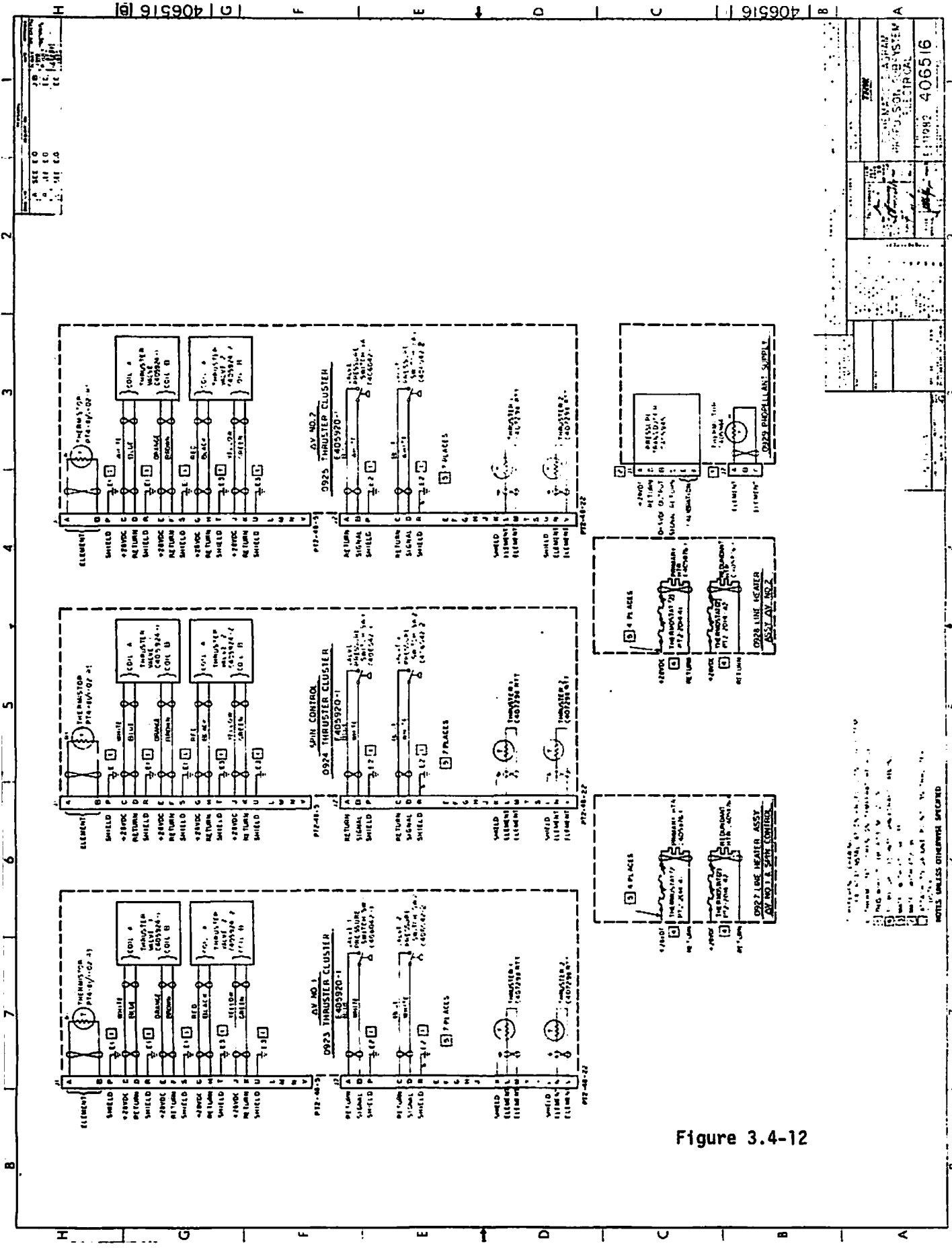


Figure 3.4-12

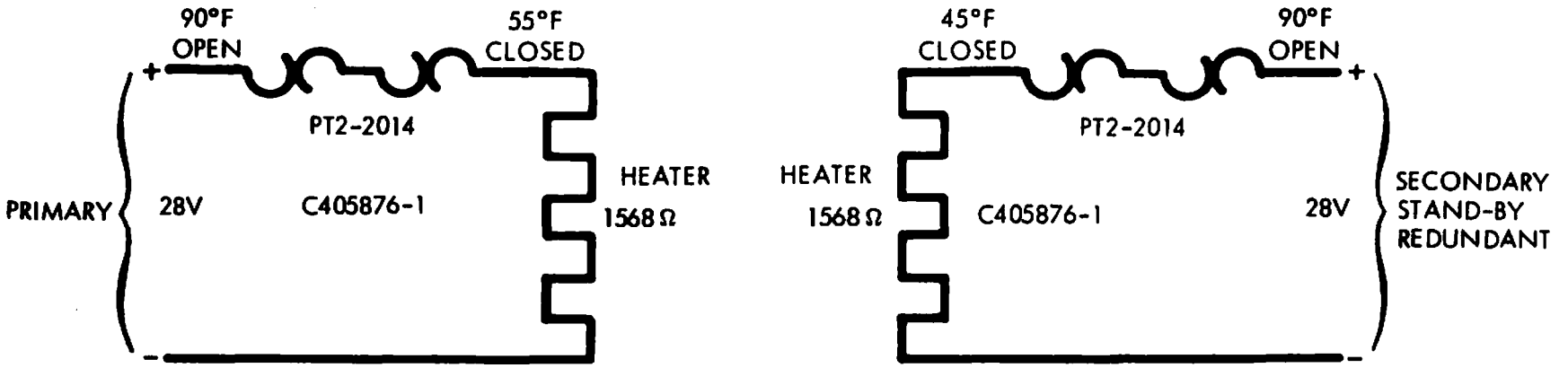


FIGURE 3.4-13 TYPICAL ELECTRICAL CONTROL SCHEMATIC FOR THE LINE HEATER ASSEMBLY.

are completed in the spacecraft CDU. The nominal power budget for these PSS components and the electrical operating limits are summarized in Table 2.

The primary LHA heater circuit is designed such that the control thermostate will close at 55°F and open at 90°F. The secondary redundant circuit is designed to remain in the open condition unless the primary circuit fails permitting the LHA temperature to drop to 45°F. The secondary thermostat control is also set to open when the temperature reaches 90°F.

The thruster propellant valves are signaled open by sequencer or ground command through the CEA and valve drivers. Each valve is designed with redundant solenoid coils, either one of which will cause the valve to reach full open within 15 milliseconds of signal receipt and full closed within 10 milliseconds of signal removal. Each solenoid coil is part of a separate redundant spacecraft command circuit (DSL, see Section 3.3.1.4).

The thruster pressure switches contain a stainless steel diaphrahm which operates with a snap action to close an electrical switching element when the thruster chamber pressure increases to 31 psia and reopens when pressure decays to 16 psia. These pressure switch signals are fed into the spacecraft electronics to provide telemetry indications of individual thruster on/off status and maintain a total pulse count history. The characteristics of the other PSS telemetry transducers is described in Section 3.4.2.1.1.

3.4.1.2.1.3 Subsystem Thermal Control. The PSS operating temperature limits are summarized in Table 3. These temperatures are maintained during the mission by the following methods:

- Resistance type heaters on the propellant supply lines (LHAs previously described).
- Radioisotope heater units (RHUs) installed in the TCA housings.
- Various types of insulation materials including standoffs and heat shields.

The thermostatically controlled LHAs automatically maintain the feedlines above 45°F to prevent freezing of the hydrazine between the PSA and TCAs. The proper operation of these heaters is indicated by telemetry measurements of TCA inlet temperatures. The temperature of the PSA is determined by that of the equipment compartment, and is expected to vary over a maximum range of 40-110°F as indicated by the flight telemetry measurement of PSA fuel outlet temperature.

The TCA is entirely housed in a baseplate and cover assembly. Three one-watt RHUs provide the necessary thermal energy to prevent fuel contained in the TCAs from freezing in the near Jupiter environment. Two of these units are mounted inside tubes in the center of the valve mounting bracket, and the third is located in the center of the thruster bracket (see Figure 5). The RHUs are described in more detail in Section 3.8. Thermal standoffs are used on both the valve and thruster brackets to regulate

TABLE 3.4-2 PSS Nominal Power Budget and Electrical Operating Limits

PSS Component	Power Source	Nominal Power Requirement	Operating Limits	Comments
Propellant Valve (each)				
Pressure Switch (each)				
Pressure Transducer (each)				
Line Heater Assembly (each) Primary Circuit				
Line Heater Assembly (each) Secondary Circuit		(Data to be supplied at completion of PSS qualification testing.)		
Propellant Temperature Transducer				
Thruster Cluster (each) Propellant Inlet Temperature Transducer				
Thruster Case (each) Temperature Transducer				

TABLE 3.4-3 PSS Thermal Operating Limits

PSS Component Measurement	Operating Temperature Limits - °F Non-Firing	Firing	Comments
1. Propellant Supply Temperature			
2. Thruster Cluster Assembly Propellant Inlet Temperature			
3. Velocity Thruster 1A Temperature			
4. Velocity Thruster 1B Temperature	(Data to be supplied at the completion of PSS qualification testing).		
5. Velocity Thruster 2A Temperature			
6. Velocity Thruster 2B Temperature			

propellant valve temperatures between 40-250°F under all anticipated mission conditions. Gold plated foil and aluminized Kapton insulation are utilized to further insulate the thruster bracket assembly and the propellant valves. Second surface mirrors are bonded to the top surface of the TCA cover, and aluminized Kapton insulation covers the remaining exterior surface of the TCA. Excessive heat losses are thus prevented during non-operating cold environments during the mission while preventing excessive heating of the components during the thruster firing hot environment. The single thermistor mounted on the TCA propellant manifold, and the two resistance transducers welded on the thrust chambers provide telemetry indications of the TCA thermal status.

3.4.2 Outputs

3.4.2.1 Telemetry

3.4.2.1.1 Flight Transducer Characteristics. Table 4 summarizes the telemetry measurement list and associated accuracies. These measurements are made by the following types of transducers.

- A strain gage type propellant supply pressure transducer (Figure 14).
- An immersion type thermistor propellant supply temperature transducer (Figure 15).
- A surface mounted thermistor TCA inlet temperature transducer.
- A surface mounted resistance element thruster body temperature transducer.
- A click action thruster chamber pressure switch transducer.

The pressure transducer is vacuum referenced, and when supplied with 28 VDC input will provide a 0-3 VDC linear, directly proportional analog output as the sensed pressure varies from 0 to 600 psia. The propellant temperature transducer provides a nonlinear inverse resistance versus temperature characteristic which is part of a bridge circuit completed in the spacecraft electronics. Propellant temperatures in the 40-140°F range are monitored by this transducer.

The surface mounted thermistors on the TCA propellant inlet manifolds provide an inverse nonlinear resistance characteristic indicating TCA inlet temperatures between 40-200°F. The resistance transducers surface mounted on the individual thrusters yield a directly varying resistance versus temperature output with a 400-1800°F measurement range. The pressure switches have been described in a previous section, and provide an on/off thruster status through bilevel telemetry channels. In addition, the pressure switch actuations are counted in the CDU (Reference para. 3.9) to provide a totalization of thruster pulses. The telemetry calibration curves for each flight PSS will be provided in Reference TBS.

TABLE 3.4-4 PSS Telemetry Summary

Measurement Title	Transducer Location	Range	Accuracy	Telemetry Quantization
1. Propellant Supply Pressure				
2. Propellant Supply Temperature				
3. VPT 1 & 2 (+Y side) Propellant Inlet Temperature				
4. VPT 3 & 4 (-Y side) Propellant Inlet Temperature				
	(Data to be supplied at completion of PSS qualification testing.)			
5. SCT 1 & 2 (-Y side) Propellant Inlet Temperature				
6. VPT 1 Case Temperature				
7. VPT 2 Case Temperature				
8. VPT 3 Case Temperature				
9. VPT 4 Case Temperature				

TABLE 3.4-4 PSS Telemetry Summary (Continued)

	Measurement Title	Transducer Location	Range	Accuracy	Telemetry Quantization
10.	VPT 1 Pulse Count				
11.	VPT 2 Pulse Count				
12.	VPT 3 Pulse Count				
13.	VPT 4 Pulse Count				
14.	SCT 1 Pulse Count				
15.	SCT 2 Pulse Count				
16.	VPT 1 On-Off Status				
17.	VPT 2 On-Off Status				
18.	VPT 3 On-Off Status				
19.	VPT 4 On-Off Status				
20.	SCT 1 On-Off Status				
21.	SCT 2 On-Off Status				

FIGURE 3.4-14

PRESSURE TRANSDUCER

IN 0-600 PSIA
OUT 0-3 VOLTS

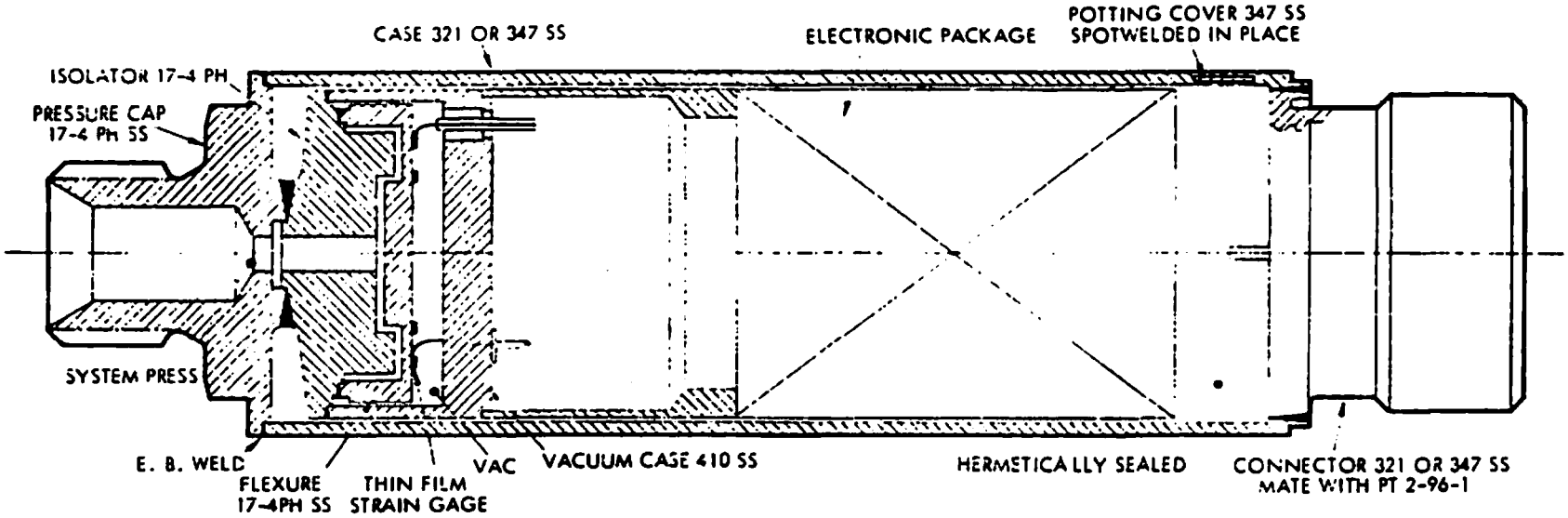
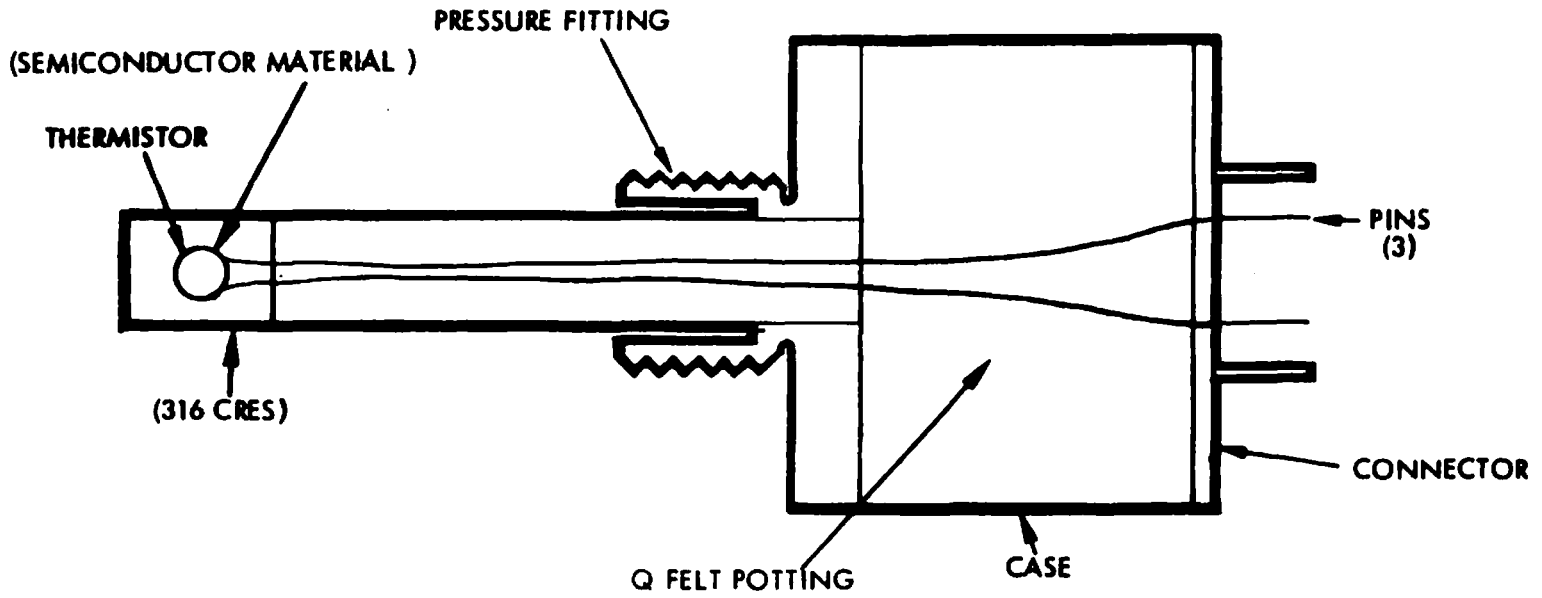


FIGURE 3.4-15

TEMPERATURE TRANSDUCER



3.4.2.1.2 Health Assessment. The normal PSS telemtry limits during the various mission operating phases are summarized in Table 5. These limits are based upon analyses of PSS qualification test data and spacecraft thermal vacuum testing. Measurement outputs beyond these limits may be indicative of either telemetry or subsystem malfunctions, and should prompt further investigation by the propulsion flight support analyst to determine corrective action.

3.4.2.1.3 Signal Responses. Thruster firing signals should result in an immediate "ON" indication by the appropriate bilevel thruster initiation status telemetry channel plus an increase in pulse counter output. Removal of the firing signal should result in an immediate return to "OFF" status.

Secondary PSS telemetry responses to pulse train or steady-state firing signals are:

- A decay in propellant supply pressure following the blowdown curve (Figure 10).
- An increase in thruster temperature to a maximum of approximately 1500°F depending on firing duration.
- An increase in TCA inlet temperature to a maximum of 150°F during firings with a post firing soakback of 230°F, the magnitudes again dependent on firing duration.

Other secondary spacecraft indicators of thruster firings are:

- A change in spacecraft spin rate.
- Spacecraft precession.
- A change in spacecraft orbital parameters.

A failure to observe these normal thruster firing indicators may be indicative of a thruster or thruster command link malfunction, and warrants further investigation by the flight support analyst prior to initiation of the next spacecraft maneuver command sequence.

TABLE 3.4-5 PSS Normal Telemetry Limits

Telemetry Measurement	Nominal Limits	Comments
1. Propellant Supply Pressure		
2. Propellant Supply Temperature		
3. VPT 1 & 2 (+Y side) Propellant Inlet Temperature		
		(Data to be supplied at the completion of PSS qualification testing.)
4. VPT 3 & 4 (-Y side) Propellant Inlet Temperature		
5. SCT 1 & 2 (-Y side) Propellant Inlet Temperature		
6. VPT 1 Case Temperature		
7. VPT 2 Case Temperature		
8. VPT 3 Case Temperature		
9. VPT 4 Case Temperature		

SECTION 3.4 NOTATION

CDU	Command Distribution Unit
DSL	Duration and Steering Logic, Part of Attitude Control Subsystem
EPT-10	Rubber Diaphragm Material
GN_2	Gaseous Nitrogen
LHA	Line Heater Assembly
N_2H_4	Hydrazine Propellant
PSA	Propellant Supply Assembly
PSIA	Pounds per square inch absolute
PSS	Propulsion Subsystem
RHU	Radioisotope Heater Units (Described in Section 3.8)
RTG	Radioisotope Thermoelectric Generator (Described in Section 3.8)
SCT	Spin Control Thruster
TBD	To Be Determined
TBS	To Be Supplied
TCA	Thruster Cluster Assembly
VDC	Volts, Direct Current
VPT	Velocity - Precession Thruster

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3.5 DATA HANDLING SUBSYSTEM

3.5.1 Subsystem Description

The data handling subsystem accepts digital and analog data from other spacecraft subsystems and the scientific instruments, arranges these data into formats based on scientific and engineering requirements, and generates a coded digital data bit stream which modulates a subcarrier signal at a suitable bit rate for the communications subsystem to telemeter back to earth. The data handling subsystem is capable of storing up to 49,152 bits of scientific and engineering data for delayed transmission to earth. The memory system is also used to buffer high rate instrument data (16K BPS) into the prevailing telemetry bit rate. The data handling subsystem provides various timing and control signals to the scientific instruments and to other spacecraft subsystems.

The data handling subsystem consists of the Digital Telemetry Unit (DTU) and the Data Storage Unit (DSU). The DTU measures 5.5"H x 11.0"W x 7.5"D, weighs 6.8 pounds, and uses 3.8 watts of secondary power. The unit consists of nine printed circuit boards of six different types containing some 800 integrated circuits and 1150 discrete components. The DSU measures 3.8"H x 8.0"W x 5.0"D, weighs 3.3 pounds, and uses 0.4 watts in the standby mode, 0.6 watts when operating at 2048 BPS, and 1.2 watts when operating at 16,384 BPS. The DSU consists of a memory core stack assembly totaling 49,152 bits and three printed circuit boards containing approximately 90 integrated circuits and 930 discrete components.

The data handling subsystem interface is illustrated in Figure 3.5-1.

Figure 3.5-2 illustrates the basic DHS formats and format combinations. As shown, there are 10 science formats and 18 format combinations. Any one D format can be telemetered with either A or B; i.e., alternating one frame of format A with one frame of D-1, etc. The science formats A and B contain a 64 word science subcomm and a 128 word engineering subcom.

Five engineering formats are provided. The C format is a combination of C-1 through C-4 transmitted as four 32 word (6 bit per word) mainframes in sequence. The selected C format (C-1 through C-4) is telemetered as a 32 word (6 bit per word) format with C-1 through C-4 and the science subcomm telemetered as 2 subcomm words. For example, if C-1 were selected for acceleration as a mainframe, then both the normal science and engineering subcomms are telemetered as they are normally transmitted in format A and B.

Figure 3.5-3 illustrates the basic DHS modes. The three mutually exclusive DHS modes are real time, telemetry store, and memory readout. When in the real time mode, 23 format combinations may be transmitted. In addition, the buffer mode which is used to store high rate data from the IPP instrument is part of the real time mode.

The telemetry store mode is commanded when formatted data is to be stored. The telemetry store mode contains two submodes, DSU HALT and DSU CONTINUE. This allows data blocks to be stored upon command.

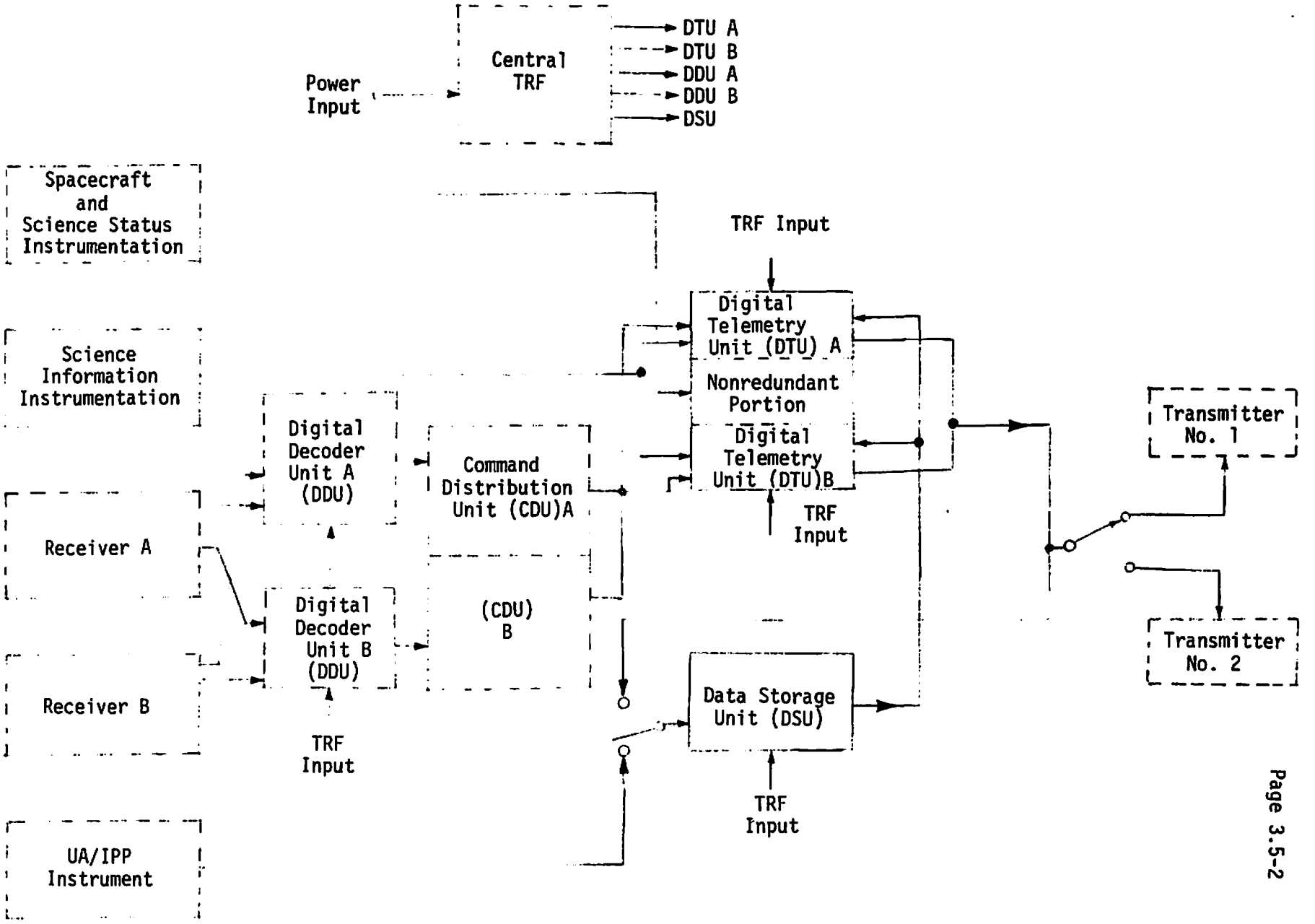
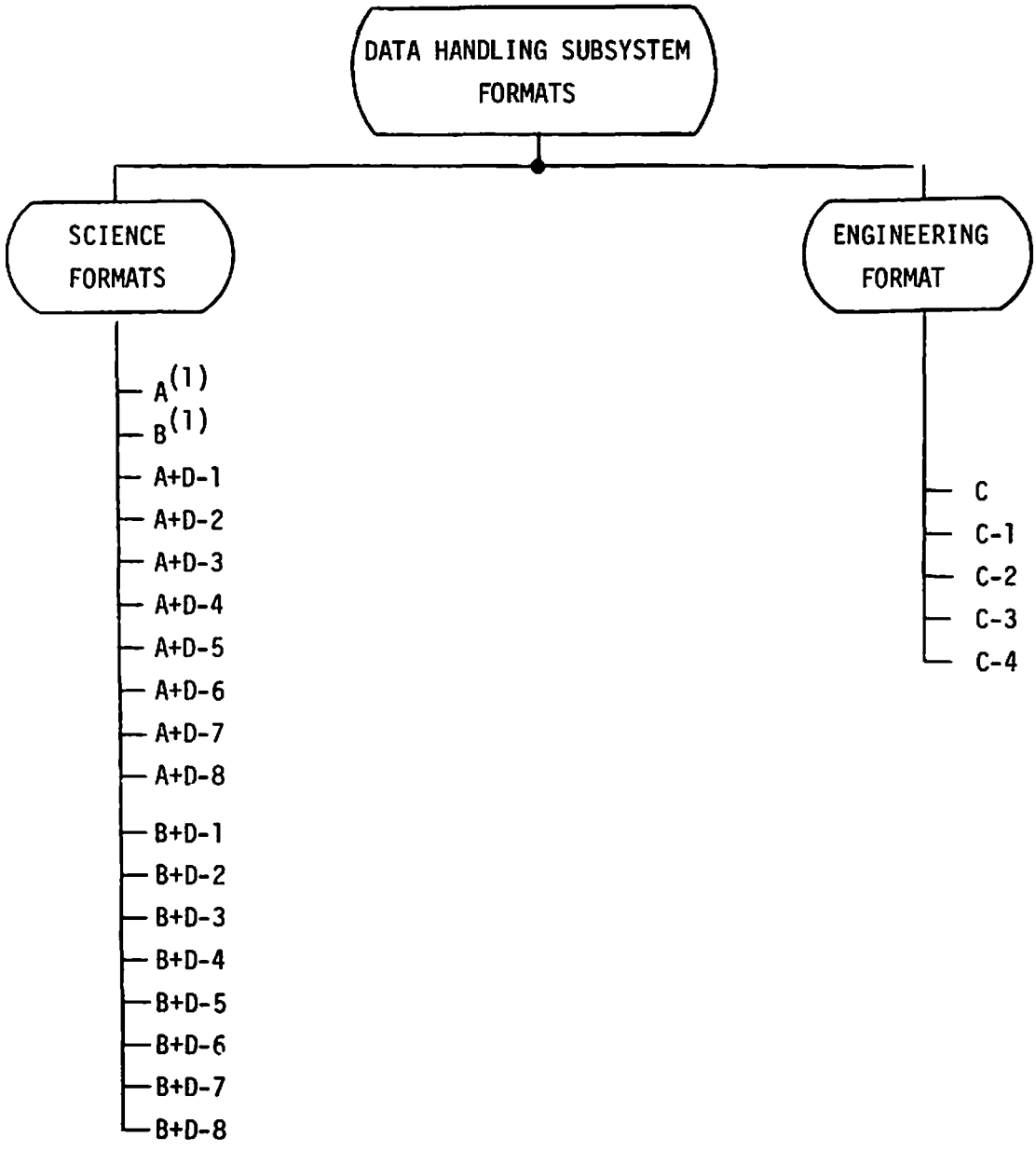


Figure 3.5-1. Data Handling Subsystem Block Diagram



NOTES: (1) Formats A and B contain a 64 word science subcomm and a 128 word engineering subcomm

Figure 3.5-2. DHS Formats

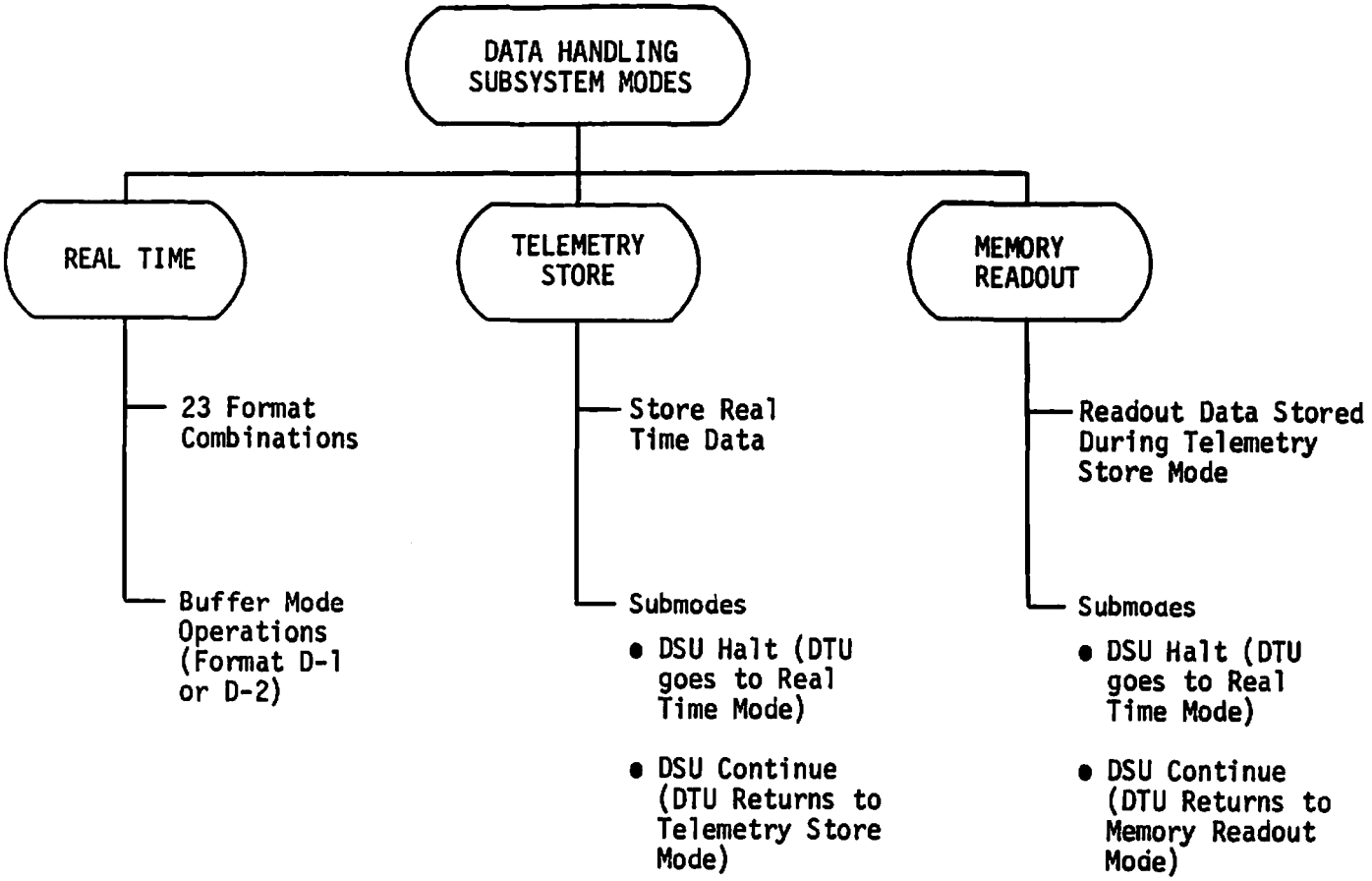


Figure 3.5-3. Data Handling Subsystem Modes

The memory readout mode is commanded when data is to be read from the DSU. The DSU HALT and DSU CONTINUE provides for reading blocks of memory.

Figure 3.5-4 illustrates how the DHS units are connected within the spacecraft system.

Figure 3.1-1 illustrates the physical location of the DHS units within the equipment compartment.

3.5.1.2 Functional Characteristics Summary. Table 1 lists the basic characteristics of the DTU; Table 2 tabulates the sampling rates for main-frame and subframe words versus format and bit rate. Table 3 tabulates the time required to store and readout data from the DSU versus bit rate, and the elapsed extended frame counter time versus bit rate. Table 4 lists the basic characteristics of the DSU.

3.5.2 DTU Operational Description

The DTU processes analog, digital and bi-level (discrete) inputs into a serial time-multiplexed PCM signal which modulates the telemetry transmitter. The DTU operates in eight bit rates, three modes, and 23 format combinations to process 290 data inputs. The DTU provides 20 timing and operational status signals to the instruments. The unit converts analog inputs into a six bit digital code, convolutional codes formatted data and provides a bi-phase modulated 32 KHz square-wave, NRZ-L output to the communication subsystem. The unit is fully redundant except for the extended frame counter, the in-flight calibration voltage generator, and the subframe multiplexer which is partially redundant. The redundant DTU can be selected by ground command.

3.5.2.1 Inputs. The DTU accepts the following types of input signals:

a) Digital and analog data from other spacecraft subsystems and the scientific instruments.

b) Operational status (bi-level) signals from other spacecraft subsystems and the scientific instruments.

c) Analog signals from the spacecraft engineering instrumentation sensors.

d) Formatted digital data from the DSU.

e) An 8-bit serial command from the CDU for changing modes, formats, and bit rates.

f) Spacecraft roll pulses from the attitude control subsystem for the generation of spin period sector pulses.

g) Power from the power subsystem.

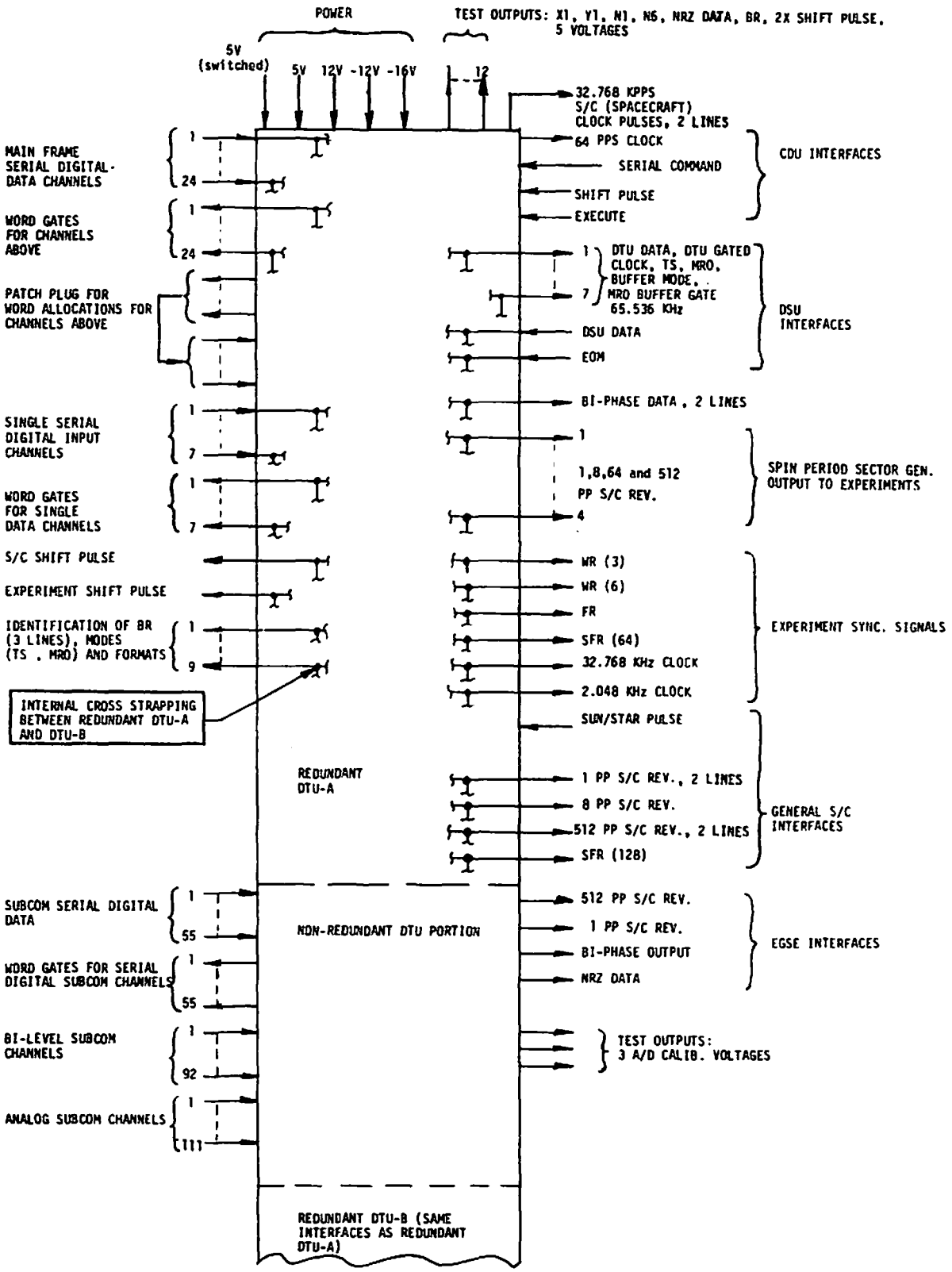


Figure 3.5-4. Data Handling Subsystem Interface with Spacecraft System

Table 1. DTU Characteristics

1.	Basic clock frequency	4.194304 MHz Stability = 0.02%
2.	Type	PCM
3.	Bit rates	2048 BPS 1024 ↕ 512 256 128 64 32 16 BPS
4.	Data input channels Mainframe	32 digital channels 24 basic science format A&B 8 Special science formats D-1 through D-8
	Subframe	55 digital 92 Bi-level 111 Analog
5.	Input voltage range	0 to +3 volts analog Digital and bi-level comparator threshold +2.0 volts
6.	A/D Accuracy	6 bits (± 48 mv)
7.	Data Output	Bi-phase modulated NRZ-L
8.	Subcarrier frequency	32.768 KHz ±0.02% squarewave
9.	Data Coding	Convolutional coded data output upon command
10.	Operating Modes	Real time Telemetry store Memory readout
11.	Data formats, science (selected by ground command)	A (Patchable) B (Patchable) D-1 (Special UA/IPP) D-2 (Special CIT/IR&UA/IPP) D-3 (Special ARC/PA) D-4 } D-5 } D-6 } (Special, Unused) D-7 } D-8 }

TABLE 1 (cont'd)

	E-1 & E-2 64 word science subframe (not selectable by ground command)
Engineering (selectable by ground command)	C - accelerate formats C-1 thru C-4 as four 32 word mainframes in sequence
	C-1 accelerate only C-1 in mainframe
	C-2 accelerate only C-2 in mainframe
	C-3 accelerate only C-3 in mainframe
	C-4 accelerate only C-4 in mainframe

12. Frame size

Mainframe	192 bits, 3 bit word groups
Subframe	Science - 64, 6 bit words Engineering - 128, 6 bit words

13. Mainframe length

Science	Format A&B - 192 Bits Format D-1 thru D-8 384 Bits
Engineering	Format C - ~~768~~ Bits 4-192 bit frames Format C-1 thru C-4 - 192 Bits

14. Extended frame counter — Identifies 8192 unique data frames

15. Spin Period Sector Generator

a) Divide S/C revolution into 512 Sectors to within 150 μs

b) Operate over S/C spin rate from 2 to 7 RPM

c) Outputs

1) 1 pulse per revolution - when in averaging mode time between any two adjacent pulses equal to within 150 μs within the same set of 64 S/C revolutions.

2) 8 pulses per revolution - Accuracy = $\frac{\text{Spin Period}}{8}$ to within 150μs

3) 64 pulses per revolution - Accuracy = $\frac{\text{Spin Period}}{64}$ to within 150 μs

4) 512 pulses per revolution - Accuracy = $\frac{\text{Spin Period}}{512}$ to within 150 μs

d) Modes
Averaging
Non-averaging
ACS

TABLE 1 (cont'd)

e) 1 PPR pulse 0° or 180° upon command

f) Phase error readout - to within 2 ms

g) Configuration at power turn-on

ACS mode
1 PPR Pulse = 0°

16. Power requirements - 3.86 watts

Table 2. Data Sampling Rate

Mainframe Words

(2) 1.048, 3/32 sec for minframe

Bit Rate	Format A, B, C-1, C-2, C-3, C-4	Format D-1 thru D-8	Format C
2048	0.094 sec	0.188 sec	0.375 sec
1024	0.188	0.375	0.750
512	0.375	0.750	1.500
256	0.750	1.500	3.000
128	1.500	3.000	6.000
64	3.000	6.000	12.000
32	6.000	12.000	24.000
16	12.000 sec	24.000 sec	48.000 sec

Science Subcomm Words

Bit Rate	Format A, B, C-1, C-2, C-3, C-4, C	Format D-1 thru D-8	
2048	0.10 min	0.20 min	
1024	0.20	0.40	
512	0.40	0.80	
256	0.80	1.60	
128	1.60	3.20	
64	3.20	6.40	
32	6.40	12.80	
16	12.8 min	25.6 min	

Engineering Subcomm Words

Bit Rate	Format A, B, C-1, C-2, C-3, C-4, C	Format D-1 thru D-8	
2048	0.20 min	0.4 min	
1024	0.40	0.8	
512	0.80	1.6	
256	1.60	3.2	
128	3.20	6.4	
64	6.40	12.8	
32	12.80	25.6	
16	25.6 min	51.2 min	

Table 3. DSU Store and Readout Time and Extended Frame Count Elapsed Time

Bit Rate	Store and Readout Time	Extended Frame Counter Elapsed Time
2048	0.40 min	12 min, 48 sec
1024	0.80	25 min, 36 sec
512	1.60	51 min, 12 sec
256	3.20	1 hr, 42 min, 24 sec
128	6.40	3 hr, 24 min, 48 sec
64	12.80	6 hr, 49 min, 36 sec
32	25.60	13 hr, 39 min, 12 sec
16	51.20 min	27 hr, 18 min, 24 sec

NOTE: DSU capacity = 49,152 bits = 256 frames (192 bits/frame)

Table 4. DSU Characteristics

1. Memory Type	Ferrite core, read-restore
2. Storage Capacity	49, 152 bits
3. Operation	Bit serial (external interface) (6 bit parallel into stack)
4. Operating Modes Telemetry Mode Buffer Mode	Store formatted data from first location to end of memory Store UA/IPP data within last 6144 bits of memory
5. Control	DSU is controlled by DTU in telemetry modes and the UA/IPP instrument in the buffer mode
6. Power Requirements	0.43 watts standby mode 0.54 watts telemetry mode 2048 BPS 1.16 watts in buffer mode with typical duty cycle as follows: 1.16 watts for 0.5 seconds 0.54 watts for 11.5 seconds

3.5.2.2 Outputs. The DTU furnishes the following output signals:

a) 32.768 KHz PCM NRZ-L bi-phase modulated squarewave.

b) 65.536 KHz clock to DSU.

c) 32.768 KHz clock to ACS, DDU, conical scan unit and scientific instruments.

d) 2048 Hz clock to scientific instruments.

e) 64 Hz clock to CDU.

f) Digital word gates to spacecraft subsystems and scientific instruments.

g) Mainframe and subframe rate pulses to scientific instruments.

h) Bit shift pulses to spacecraft and scientific instruments.

i) Format and bit rate status signals to scientific instruments.

j) Roll index pulse to spacecraft and scientific instruments.

k) Spin period sector pulses (512, 64, and 8) to spacecraft and scientific instruments.

l) Commands and control signals to DSU.

m) Digital data to DSU.

3.5.2.3 Circuit Blocks. The DTU contains the following basic circuit blocks:

a) Crystal controlled oscillator

b) Programmer

c) Multiplexers

d) Digital comparator

e) A/D converter

f) Fixed word generator

g) Combiner

h) Convolutional coder

i) Bi-phase modulator

j) Command decoder logic

k) DSU control logic

l) Spin period sector generator

m) Roll attitude timing counter

n) Interface buffers

o) In-flight A/D calibration voltages

p) Extended frame counter

Figure 3.5-5 shows the interconnect between each basic block. An eight bit serial command is received from the command decoder unit; six of these are used to command the DTU. The first two bits shifted into the DTU are not used. The command decoding logic generates outputs to the programmer to control bit rate, format and mode. The command logic sends outputs to the DSU control logic to control DSU operation and to the convolutional coder to select coded or uncoded data.

3.5.2.4 Multiplexers. The data multiplexer selects each data input, one at a time, in accordance with the data format and provides a connection for the selected input to the A/D converter (analog inputs) or digital comparator (digital and bi-level inputs).

The mainframe multiplexer is redundant and accepts only digital inputs. The subframe multiplexer is non-redundant and accepts analog, bi-level and digital inputs.

Each digital input has a corresponding word gate output which is used by the data source to gate data to the DTU. The DTU accepts digital data only during the period in which the corresponding word gate is true.

3.5.2.5 Digital Comparator. The digital comparator compares the serial digital and bi-level channel inputs with a fixed reference point selected midway between the minimum true and the maximum false input levels (approximately 2 volts). If the input level is higher than the threshold voltage, the digital comparator gives a binary one output to the combiner, or a binary zero output, if the data input level is lower than the comparison reference voltage.

3.5.2.6 A/D Converter. The A/D converter encodes the selected analog input to a 6 bit binary word. The maximum encoding error due to all processes associated with the conversion of data from analog to a 6-bit digital word is less than $\pm$1.6% of full scale or $\pm$1 full least significant bit over the full qualification temperature range. The A/D is a ramp type converter (refer to Figure 3.5-6) where the analog input is compared with a linear ramp. Pulses are accumulated within a register at a 1 MHz rate from the time a start signal is received to the time the linear ramp voltage equals the analog input. The accumulated counts are then scaled to a six bit word for telemetry.

Tables 5 and 6 are actual analog to digital calibration charts for each redundant A/D.

The accuracy of the A/D is continually checked by encoding three reference voltages and telemetering the results within three engineering subframe words. Analog data is telemetered most significant bit first.

3.5.2.7 Combiner. The combiner collects selected input data sources, fixed words (frame sync, mode and BR ID, etc.), DSU data and internal DTU status signals and combines these data into one serial output bit stream to the convolutional coder. Figure 3.5.7 illustrates the basic combiner logic and the system for bypassing the convolutional coder when uncoded data is required.

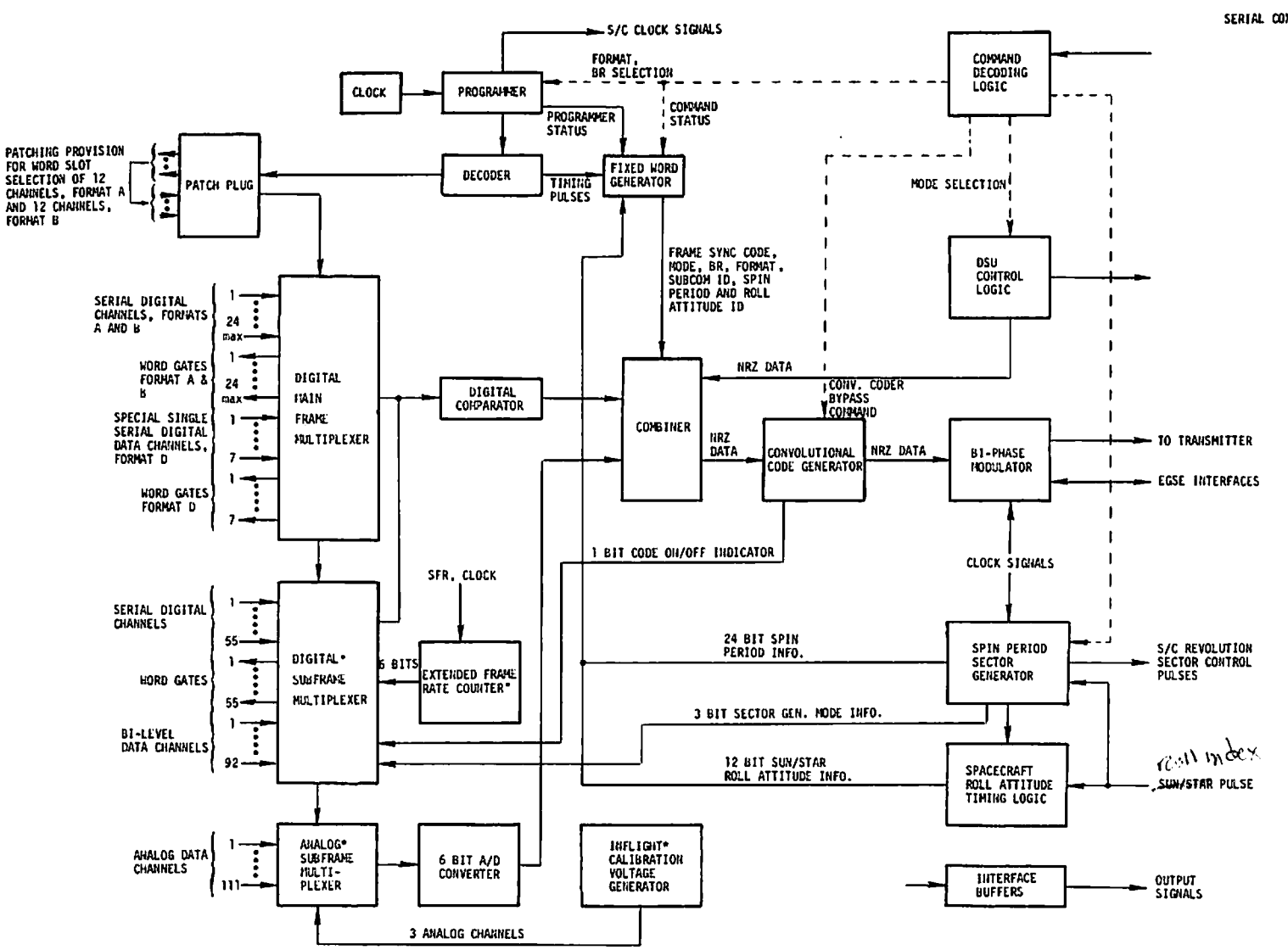
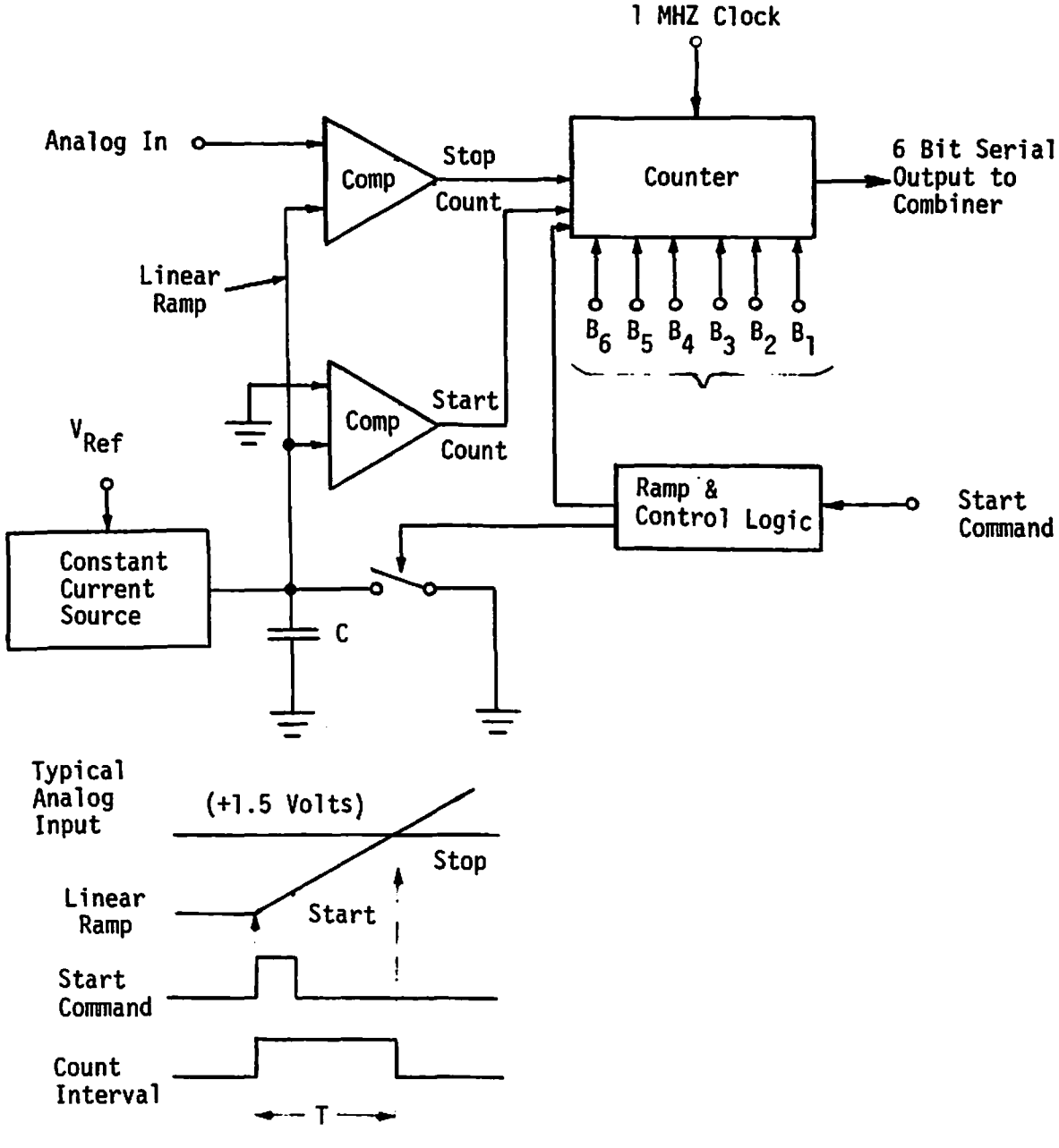


FIGURE 3.5-5. DTU FUNCTIONAL DIAGRAM



Number of Counts
Within Time Period
"T" is Proportional
to Analog Input Voltage

Figure 3.5-6. A/D Block Duagram

Table 5. A/D Conversion Chart (A/D A)

Analog Voltage	Binary Code MSB LSD	Analog Voltage	Binary Code MSB LSB
-10 to 0 v	0 0 0 0 0 0	1.488 to 1.536 v	1 0 0 0 0 0
0 to 48 mv	0 0 0 0 0 1	1.536 to 1.584 v	1 0 0 0 0 1
48 to 96 mv	0 0 0 0 1 0	1.584 to 1.632 v	1 0 0 0 1 0
96 to 144 mv	0 0 0 0 1 1	1.632 to 1.680 v	1 0 0 0 1 1
144 to 192 mv	0 0 0 1 0 0	1.680 to 1.728 v	1 0 0 1 0 0
192 to 240 mv	0 0 0 1 0 1	1.728 to 1.776 v	1 0 0 1 0 1
240 to 288 mv	0 0 0 1 1 0	1.776 to 1.824 v	1 0 0 1 1 0
288 to 336 mv	0 0 0 1 1 1	1.824 to 1.872 v	1 0 0 1 1 1
336 to 384 mv	0 0 1 0 0 0	1.872 to 1.920 v	1 0 1 0 0 0
384 to 432 mv	0 0 1 0 0 1	1.920 to 1.968 v	1 0 1 0 0 1
432 to 480 mv	0 0 1 0 1 0	1.968 to 2.016 v	1 0 1 0 1 0
480 to 528 mv	0 0 1 0 1 1	2.016 to 2.064 v	1 0 1 0 1 1
528 to 576 mv	0 0 1 1 0 0	2.064 to 2.112 v	1 0 1 1 0 0
576 to 624 mv	0 0 1 1 0 1	2.112 to 2.160 v	1 0 1 1 0 1
624 to 672 mv	0 0 1 1 1 0	2.160 to 2.208 v	1 0 1 1 1 0
672 to 720 mv	0 0 1 1 1 1	2.208 to 2.256 v	1 0 1 1 1 1
720 to 768 mv	0 1 0 0 0 0	2.256 to 2.304 v	1 1 0 0 0 0
768 to 816 mv	0 1 0 0 0 1	2.304 to 2.352 v	1 1 0 0 0 1
816 to 864 mv	0 1 0 0 1 0	2.352 to 2.400 v	1 1 0 0 1 0
864 to 912 mv	0 1 0 0 1 1	2.400 to 2.448 v	1 1 0 0 1 1
912 to 960 mv	0 1 0 1 0 0	2.448 to 2.496 v	1 1 0 1 0 0
960 to 1.008 v	0 1 0 1 0 1	2.496 to 2.544 v	1 1 0 1 0 1
1.008 to 1.056 v	0 1 0 1 1 0	2.544 to 2.592 v	1 1 0 1 1 0
1.056 to 1.104 v	0 1 0 1 1 1	2.592 to 2.640 v	1 1 0 1 1 1
1.104 to 1.152 v	0 1 1 0 0 0	2.640 to 2.688 v	1 1 1 0 0 0
1.152 to 1.200 v	0 1 1 0 0 1	2.688 to 2.736 v	1 1 1 0 0 1
1.200 to 1.248 v	0 1 1 0 1 0	2.736 to 2.784 v	1 1 1 0 1 0
1.248 to 1.296 v	0 1 1 0 1 1	2.784 to 2.832 v	1 1 1 0 1 1
1.296 to 1.344 v	0 1 1 1 0 0	2.832 to 2.880 v	1 1 1 1 0 0
1.344 to 1.392 v	0 1 1 1 0 1	2.880 to 2.928 v	1 1 1 1 0 1
1.392 to 1.440 v	0 1 1 1 1 0	2.928 to 2.976 v	1 1 1 1 1 0
1.440 to 1.488	0 1 1 1 1 1	2.976 to 5 v	1 1 1 1 1 1

Table 6. A/D Conversion Chart (A/D B)

Analog Voltage	Binary Code MSB LSB	Analog Voltage	Binary Code MSB LSB
-10 to 0 v	0 0 0 0 0 0	1.488 to 1.536 v	1 0 0 0 0 0
0 to 48 mv	0 0 0 0 0 1	1.536 to 1.584 v	1 0 0 0 0 1
48 to 96 mv	0 0 0 0 1 0	1.584 to 1.632 v	1 0 0 0 1 0
96 to 144 mv	0 0 0 0 1 1	1.632 to 1.680 v	1 0 0 0 1 1
144 to 192 mv	0 0 0 1 0 0	1.680 to 1.728 v	1 0 0 1 0 0
192 to 240 mv	0 0 0 1 0 1	1.728 to 1.776 v	1 0 0 1 0 1
240 to 288 mv	0 0 0 1 1 0	1.776 to 1.824 v	1 0 0 1 1 0
288 to 336 mv	0 0 0 1 1 1	1.824 to 1.872 v	1 0 0 1 1 1
336 to 384 mv	0 0 1 0 0 0	1.872 to 1.920 v	1 0 1 0 0 0
384 to 432 mv	0 0 1 0 0 1	1.920 to 1.968 v	1 0 1 0 0 1
432 to 480 mv	0 0 1 0 1 0	1.968 to 2.016 v	1 0 1 0 1 0
480 to 528 mv	0 0 1 0 1 1	2.016 to 20.64 v	1 0 1 0 1 1
528 to 576 mv	0 0 1 1 0 0	2.064 to 2.112 v	1 0 1 1 0 0
576 to 624 mv	0 0 1 1 0 1	2.112 to 2.160 v	1 0 1 1 0 1
624 to 672 mv	0 0 1 1 1 0	2.160 to 2.208 v	1 0 1 1 1 0
672 to 720 mv	0 0 1 1 1 1	2.208 to 2.256 v	1 0 1 1 1 1
720 to 768 mv	0 1 0 0 0 0	2.256 to 2.304 v	1 1 0 0 0 0
768 to 816 mv	0 1 0 0 0 1	2.304 to 2.352 v	1 1 0 0 0 1
816 to 864 mv	0 1 0 0 1 0	2.352 to 2.400 v	1 1 0 0 1 0
864 to 912 mv	0 1 0 0 1 1	2.400 to 2.448 v	1 1 0 0 1 1
912 to 960 mv	0 1 0 1 0 0	2.448 to 2.496 v	1 1 0 1 0 0
960 to 1.008 v	0 1 0 1 0 1	2.496 to 2.544 v	1 1 0 1 0 1
1.008 to 1.056 v	0 1 0 1 1 0	2.544 to 2.592 v	1 1 0 1 1 0
1.056 to 1.104 v	0 1 0 1 1 1	2.592 to 2.640 v	1 1 0 1 1 1
1.104 to 1.152 v	0 1 1 0 0 0	2.640 to 2.688 v	1 1 1 0 0 0
1.152 to 1.200 v	0 1 1 0 0 1	2.688 to 2.736 v	1 1 1 0 0 1
1.200 to 1.248 v	0 1 1 0 1 0	2.736 to 2.784 v	1 1 1 0 1 0
1.248 to 1.296 v	0 1 1 0 1 1	2.784 to 2.832 v	1 1 1 0 1 1
1.296 to 1.344 v	0 1 1 1 0 0	2.832 to 2.880 v	1 1 1 1 0 0
1.344 to 1.392 v	0 1 1 1 0 1	2.880 to 2.928 v	1 1 1 1 0 1
1.392 to 1.440 v	0 1 1 1 1 0	2.928 to 2.976 v	1 1 1 1 1 0
1.440 to 1.488 v	0 1 1 1 1 1	2.976 to + 5 v	1 1 1 1 1 1

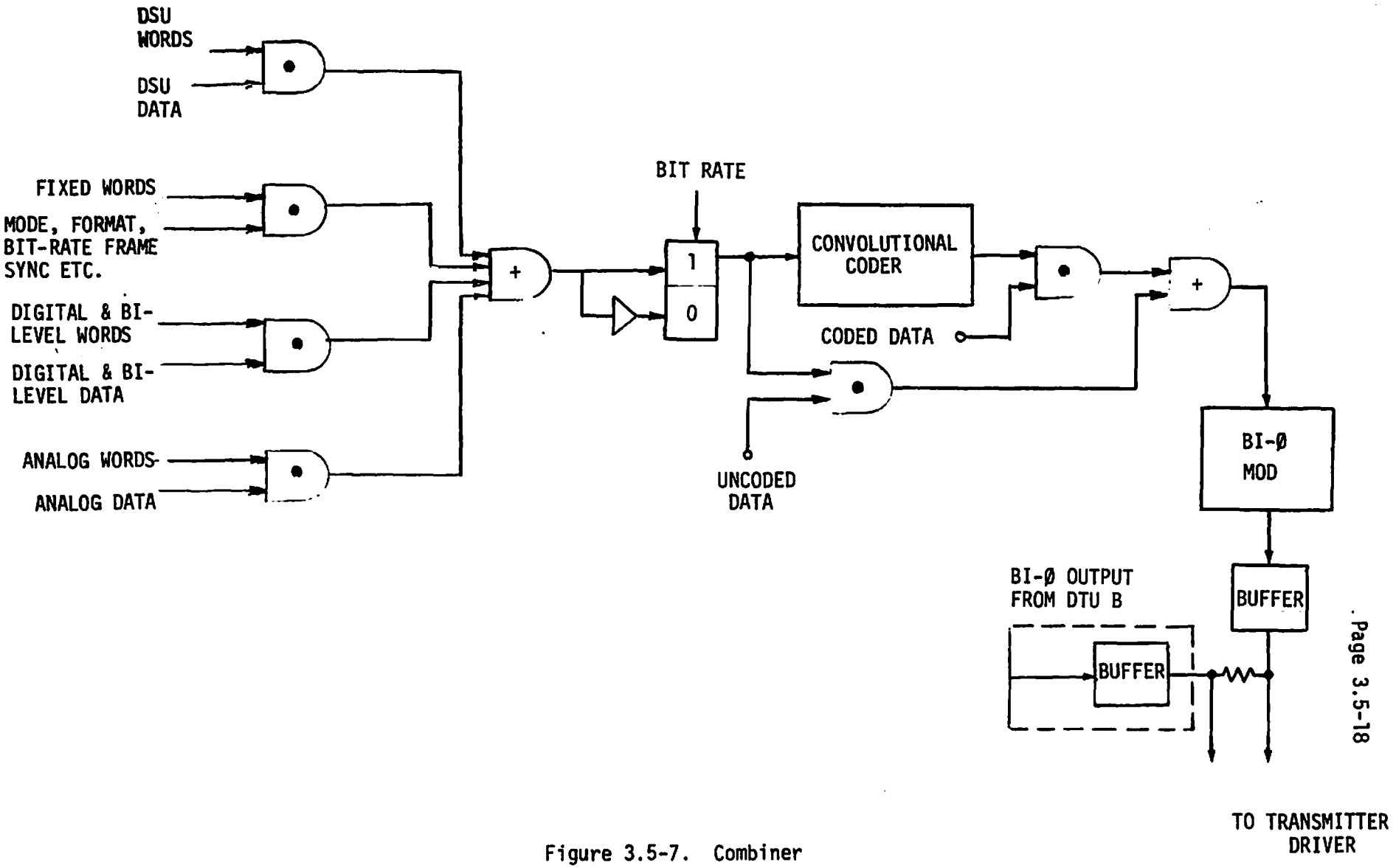


Figure 3.5-7. Combiner

The convolutional coder is bypassed with logic gates as shown in Figure 3.5-7 to provide uncoded data output. As shown, two independent sources (DTU A and DTU B) can modulate the transmitter. Cross-strapping is provided to allow either DTU to drive either transmitter driver for fail-safe operation.

3.5.2.8 Bi-Phase Modulator. This modulator modulates the serial PCM bit stream into a squarewave bi-phase data output. The modulating frequency is 32.768 KHz.

An example of the modulation is given in Figure 3.5-8. The modulated 32 KHz can be in phase for a "1" bit and 180° from phase for a "0" bit or vice versa. The initial phase is determined by the state the bi-phase modulator flip-flop happens to assume during DTU power turn-on. In Figure 3.5-8, an in phase condition represents a "0" and an out of phase condition represents a "1" bit. The phase change is delayed one bit time plus one-half subcarrier cycle as indicated in the figure.

Figure 3.5-9 illustrates the bi-phase output interface with the transmitter driver. The circuit is designed such that either DTU can drive either driver. Furthermore, either output line can be shorted to ground and the other DTU will still drive the unshorted driver.

3.5.2.9 Command Operations. The DTU receives an eight bit binary coded message from the CDU of which only the last six bits are processed. Since the spacecraft command message is eight bits long and the DTU requires only six bits for all of the data handling subsystem commands, each DTU command will respond to at least four 8-bit command numbers.

Figure 3.5-10 illustrates the command decode logic within the DTU. The first two bits (bits 3 and 4) determine the command type and the remaining four bits contain the select message. All commands are processed immediately except the mode commands which are synchronized with frame rate.

The following commands can be detected and processed by the DTU.

1) 8 bit rate commands:

   16, 32, 64, 128, 256, 512, 1024, and 2048 bits per second.

2) 23 format command combinations:

   Formats A, B, C, C1 through C4, D1 through D8 interleaved with Format A and D1 through D8 interleaved with Format B.

3) 3 basic modes:

   Real time, telemetry store and memory readout.

   9 secondary modes:

   Memory halt, memory continue, 0° roll index, 180° roll index, coded data, uncoded data, spin averaging, non-spin averaging. ACS non-spin averaging.

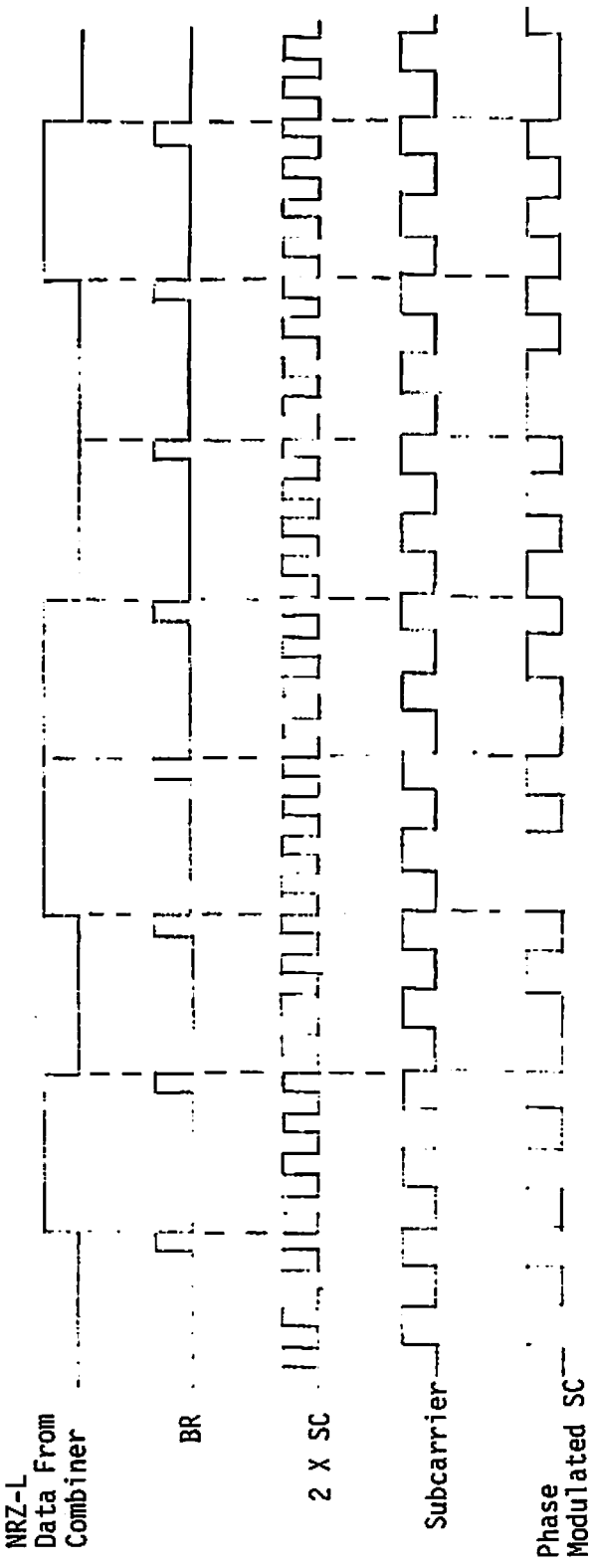


Figure 3.5-8. Bi-Phase Modulator Timing Diagram

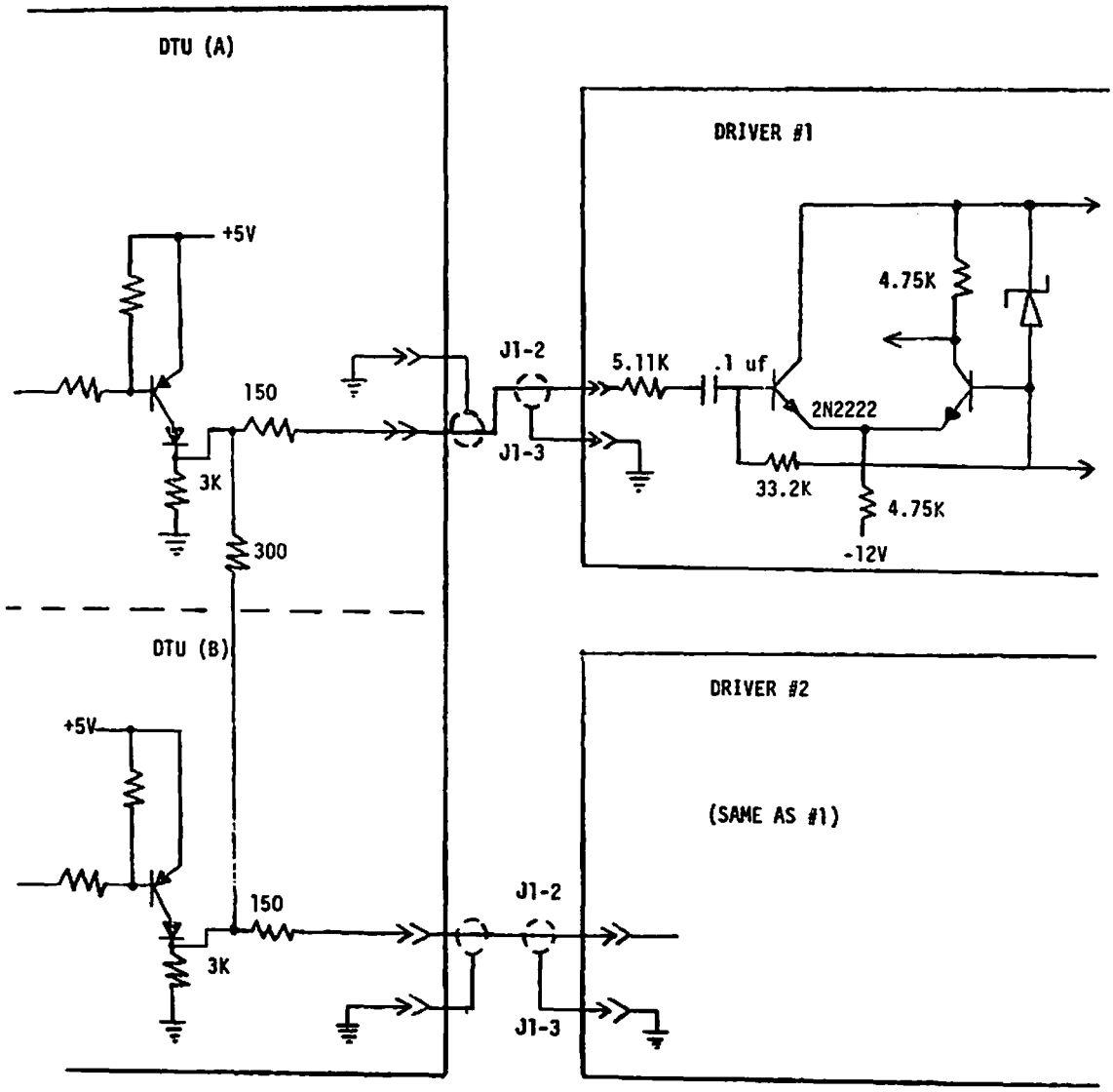


Figure 9. Circuit for Bi-Phase Modulator Output

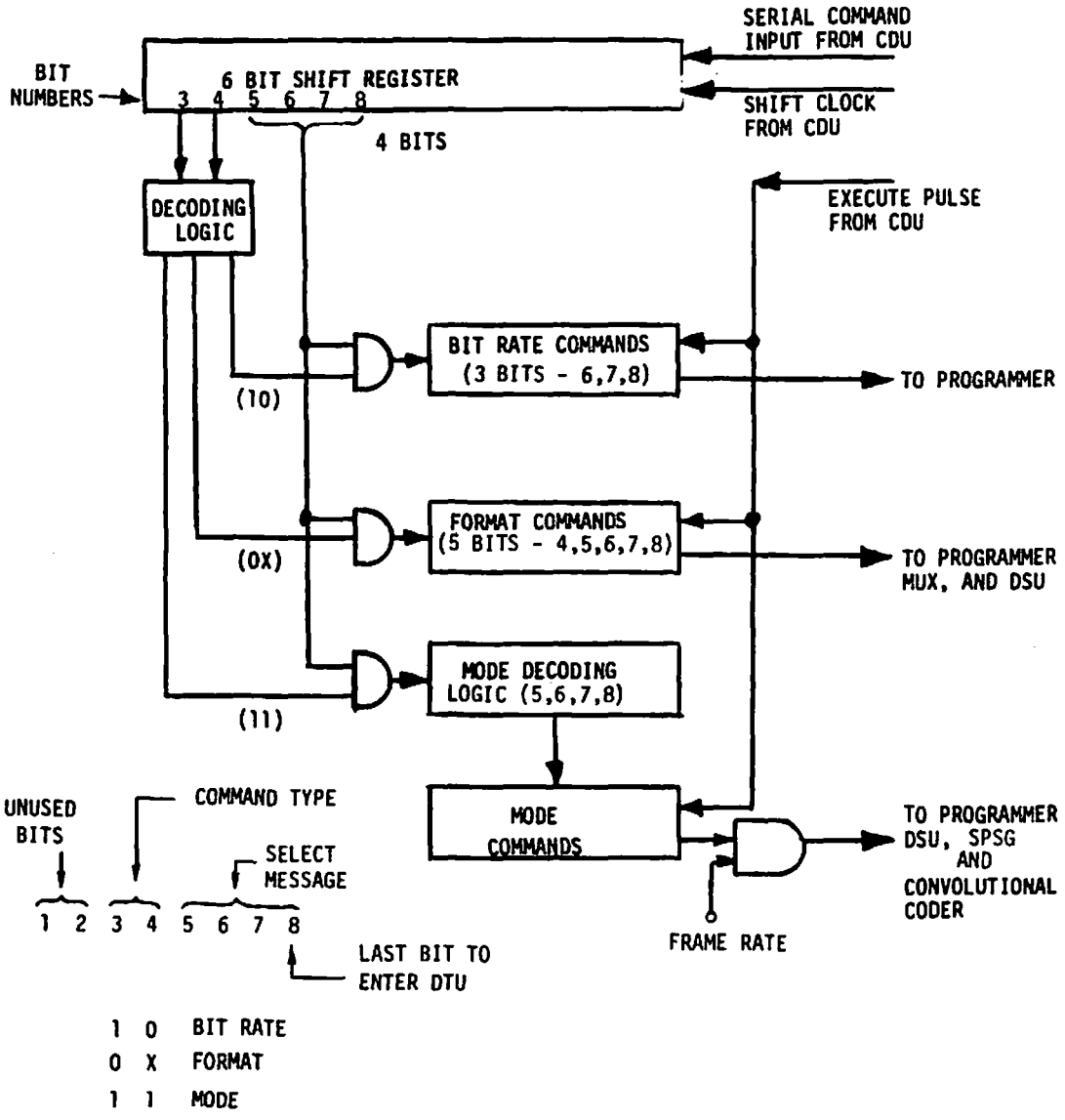


Figure 3.5-10. Command Decode Logic

The commands to select DTU redundancy A and B are decoded directly in the CDU. These commands operate relays within the power subsystem to turn DTU power on. The DTU upon power turn-on will operate in the real time mode, in format C at a bit rate of 512 BPS, with coded data and 0° roll index pulse output and SPSG in ACS mode.

The following is a complete list of all the 8 bit command numbers that the DTU will respond to for the selection of modes, formats, and bit rates.

Table 7. Bit Command Numbers Accepted by the DTU

Bit Rate Commands

Bit Rate	Command Numbers			
	Octal	(Bits 1 & 2 not used by DTU)	Binary 1 2 3 4 5 6 7 8	(Last bit to enter DTU)
16 BPS	0 4 0		0 0 1 0 0 0 0 0	
	1 4 0		0 1 1 0 0 0 0 0	
	2 4 0		1 0 1 0 0 0 0 0	
	3 4 0		1 1 1 0 0 0 0 0	
	0 5 0		0 0 1 0 1 0 0 0	
	1 5 0	250	0 1 1 0 1 0 0 0	
	3 5 0		1 1 1 0 1 0 0 0	
32 BPS	0 4 1		0 0 1 0 0 0 0 1	
	1 4 1		0 1 1 0 0 0 0 1	
	2 4 1		1 0 1 0 0 0 0 1	
	3 4 1		1 1 1 0 0 0 0 1	
	0 5 1		0 0 1 0 1 0 0 1	
	1 5 1		0 1 1 0 1 0 0 1	
	2 5 1		1 0 1 0 1 0 0 1	
	3 5 1		1 1 1 0 1 0 0 1	
64 BPS	0 4 2		0 0 1 0 0 0 1 0	
	1 4 2		0 1 1 0 0 0 1 0	
	2 4 2		1 0 1 0 0 0 1 0	
	3 4 2		1 1 1 0 0 0 1 0	
	0 5 2		0 0 1 0 1 0 1 0	
	1 5 2		0 1 1 0 1 0 1 0	
	2 5 2		1 0 1 0 1 0 1 0	
	3 5 2		1 1 1 0 1 0 1 0	
128 BPS	0 4 3		0 0 1 0 0 0 1 1	
	1 4 3		0 1 1 0 0 0 1 1	
	2 4 3		1 0 1 0 0 0 1 1	
	3 4 3		1 1 1 0 0 0 1 1	

Table 7. (continued)

Bit Rate	Command Numbers											
	Octal	(Bits 1 & 2 not used by DTU)	Binary								(Last bit to enter DTU)	
			1	2	3	4	5	6	7	8		
128 BPS (continued)	0 5 3		0	0	1	0	1	0	1	1		
	1 5 3		0	1	1	0	1	0	1	1		
	2 5 3		1	0	1	0	1	0	1	1		
	3 5 3		1	1	1	0	1	0	1	1		
256 BPS	0 4 4		0	0	1	0	0	1	0	0		
	1 4 4		0	1	1	0	0	1	0	0		
	2 4 4		1	0	1	0	0	1	0	0		
	3 4 4		1	1	1	0	0	1	0	0		
	0 5 4		0	0	1	0	1	1	0	0		
	1 5 4		0	1	1	0	1	1	0	0		
	2 5 4		1	0	1	0	1	1	0	0		
	3 5 4		1	1	1	0	1	1	0	0		
512 BPS	0 4 5		0	0	1	0	0	1	0	1		
	1 4 5		0	1	1	0	0	1	0	1		
	2 4 5		1	0	1	0	0	1	0	1		
	3 4 5		1	1	1	0	0	1	0	1		
	0 5 5		0	0	1	0	1	1	0	1		
	1 5 5		0	1	1	0	1	1	0	1		
	2 5 5		1	0	1	0	1	1	0	1		
	3 5 5		1	1	1	0	1	1	0	1		
1024 BPS	0 4 6		0	0	1	0	0	1	1	0		
	1 4 6		0	1	1	0	0	1	1	0		
	2 4 6		1	0	1	0	0	1	1	0		
	3 4 6		1	1	1	0	0	1	1	0		
	0 5 6		0	0	1	0	1	1	1	0		
	1 5 6		0	1	1	0	1	1	1	0		
	2 5 6		1	0	1	0	1	1	1	0		
	3 5 6		1	1	1	0	1	1	1	0		
2048 BPS	0 4 7		0	0	1	0	0	1	1	1		
	1 4 7		0	1	1	0	0	1	1	1		
	2 4 7		1	0	1	0	0	1	1	1		
	3 4 7		1	1	1	0	0	1	1	1		
	0 5 7		0	0	1	0	1	1	1	1		
	1 5 7		0	1	1	0	1	1	1	1		
	2 5 7		1	0	1	0	1	1	1	1		
	3 5 7		1	1	1	0	1	1	1	1		

Table 7. (continued)

Format Commands

Format	Command Numbers			
	Octal	(Bits 1 & 2 not used by DTU)	Binary	(Last bit to enter DTU)
			1 2 3 4 5 6 7 8	
A	0 1 0		0 0 0 0 1 0 0 0	
	1 1 0		0 1 0 0 1 0 0 0	
	2 1 0		1 0 0 0 1 0 0 0	
	3 1 0		1 1 0 0 1 0 0 0	
	0 1 1		0 0 0 0 1 0 0 1	
	1 1 1		0 1 0 0 1 0 0 1	
	2 1 1		1 0 0 0 1 0 0 1	
	3 1 1		1 1 0 0 1 0 0 1	
B	0 0 0		0 0 0 0 0 0 0 0	
	1 0 0		0 1 0 0 0 0 0 0	
	2 0 0		1 0 0 0 0 0 0 0	
	3 0 0		1 1 0 0 0 0 0 0	
	0 0 1		0 0 0 0 0 0 0 1	
	1 0 1		0 1 0 0 0 0 0 1	
	2 0 1		1 0 0 0 0 0 0 1	
	3 0 1		1 1 0 0 0 0 0 1	
C	0 0 2		0 0 0 0 0 0 1 0	
	1 0 2		0 1 0 0 0 0 1 0	
	2 0 2		1 0 0 0 0 0 1 0	
	3 0 2		1 1 0 0 0 0 1 0	
	0 0 3		0 0 0 0 0 0 1 1	
	1 0 3		0 1 0 0 0 0 1 1	
	2 0 3		1 0 0 0 0 0 1 1	
	3 0 3		1 1 0 0 0 0 1 1	
	0 1 2		0 0 0 0 1 0 1 0	
	1 1 2		0 1 0 0 1 0 1 0	
	2 1 2		1 0 0 0 1 0 1 0	
	3 1 2		1 1 0 0 1 0 1 0	
	0 1 3		0 0 0 0 1 0 1 1	
	1 1 3		0 1 0 0 1 0 1 1	
	2 1 3		1 0 0 0 1 0 1 1	
	3 1 3		1 1 0 0 1 0 1 1	
C-1	0 0 4		0 0 0 0 0 1 0 0	
	1 0 4		0 1 0 0 0 1 0 0	
	2 0 4		1 0 0 0 0 1 0 0	
	3 0 4		1 1 0 0 0 1 0 0	
	0 1 4		0 0 0 0 1 1 0 0	
	1 1 4		0 1 0 0 1 1 0 0	
	2 1 4		1 0 0 0 1 1 0 0	
	3 1 4		1 1 0 0 1 1 0 0	

Table 7. (continued)

Format	Command Numbers			
	Octal	(Bits 1 & 2 not used by DTU)	Binary	(Last bit to enter DTU)
			1 2 3 4 5 6 7 8	
C-2	0 0 5		0 0 0 0 0 1 0 1	
	1 0 5		0 1 0 0 0 1 0 1	
	2 0 5		1 0 0 0 0 1 0 1	
	3 0 5		1 1 0 0 0 1 0 1	
	0 1 5		0 0 0 0 1 1 0 1	
	1 1 5		0 1 0 0 1 1 0 1	
	2 1 5		1 0 0 0 1 1 0 1	
	3 1 5		1 1 0 0 1 1 0 1	
C-3	0 0 6		0 0 0 0 0 1 1 0	
	1 0 6		1 0 0 0 0 1 1 0	
	3 0 6		1 1 0 0 0 1 1 0	
	0 1 6		0 0 0 0 1 1 1 0	
	1 1 6		0 1 0 0 1 1 1 0	
	2 1 6		1 0 0 0 1 1 1 0	
	3 1 6		1 1 0 0 1 1 1 0	
C-4	0 0 7		0 0 0 0 0 1 1 1	
	1 0 7		0 1 0 0 0 1 1 1	
	2 0 7		1 0 0 0 0 1 1 1	
	3 0 7		1 1 0 0 0 1 1 1	
	0 1 7		0 0 0 0 1 1 1 1	
	1 1 7		0 1 0 0 1 1 1 1	
	2 1 7		1 0 0 0 1 1 1 1	
	3 1 7		1 1 0 0 1 1 1 1	
A/D-1	0 3 0		0 0 0 1 1 0 0 0	
	1 3 0		0 1 0 1 1 0 0 0	
	2 3 0		1 0 0 1 1 0 0 0	
	3 3 0		1 1 0 1 1 0 0 0	
A/D-2	0 3 1		0 0 0 1 1 0 0 1	
	1 3 1		1 0 0 1 1 0 0 1	
	2 3 1		1 0 0 1 1 0 0 1	
	3 3 1		1 1 0 1 1 0 0 1	
A/D-3	0 3 2		0 0 0 1 1 0 1 0	
	1 3 2		0 1 0 1 1 0 1 0	
	2 3 2		1 0 0 1 1 0 1 0	
	3 3 2		1 1 0 1 1 0 1 0	

Table 7. (continued)

Format	Command Numbers			
	Octal	(Bits 1 & 2 not used by DTU)	Binary	(Last bit to enter DTU)
			1 2 3 4 5 6 7 8	
A/D-4*	0 3 3		0 0 0 1 1 0 1 1	
	1 3 3		0 1 0 1 1 0 1 1	
	2 3 3		1 0 0 1 1 0 1 1	
	3 3 3		1 1 0 1 1 0 1 1	
A/D-5*	0 3 4		0 0 0 1 1 1 0 0	
	1 3 4		0 1 0 1 1 1 0 0	
	2 3 4		1 0 0 1 1 1 0 0	
	3 3 4		1 1 0 1 1 1 0 0	
A/D-6*	0 3 5		0 0 0 1 1 1 0 1	
	1 3 5		0 1 0 1 1 1 0 1	
	2 3 5		1 0 0 1 1 1 0 1	
	3 3 5		1 1 0 1 1 1 0 1	
A/D-7*	0 3 6		0 0 0 1 1 1 1 0	
	1 3 6		0 1 0 1 1 1 1 0	
	2 3 6		1 0 0 1 1 1 1 0	
	3 3 6		1 1 0 1 1 1 1 0	
A/D-8*	0 3 7		0 0 0 1 1 1 1 1	
	1 3 7		0 1 0 1 1 1 1 1	
	2 3 7		1 0 0 1 1 1 1 1	
	3 3 7		1 1 0 1 1 1 1 1	
B/D-1	0 2 0		0 0 0 1 0 0 0 0	
	1 2 0		0 1 0 1 0 0 0 0	
	2 2 0		1 0 0 1 0 0 0 0	
	3 2 0		1 1 0 1 0 0 0 0	
B/D-2	0 2 1		0 0 0 1 0 0 0 1	
	1 2 1		0 1 0 1 0 0 0 1	
	2 2 1		1 0 0 1 0 0 0 1	
	3 2 1		1 1 0 1 0 0 0 1	
B/D-3	0 2 2		0 0 0 1 0 0 1 0	
	1 2 2		0 1 0 1 0 0 1 0	
	2 2 2		1 0 0 1 0 0 1 0	
	3 2 2		1 1 0 1 0 0 1 0	

* Note that formats D-4 to D-8 can be commanded, however, there are no words assigned to these formats.

Table 7. (continued)

Format	Command Numbers			
	Octal	(Bits 1 & 2 not used by DTU)	Binary 1 2 3 4 5 6 7 8	(Last bit to enter DTU)
B/D-4*	0 2 3		0 0 0 1 0 0 1 1	
	1 2 3		0 1 0 1 0 0 1 1	
	2 2 3		1 0 0 1 0 0 1 1	
	3 2 3		1 1 0 1 0 0 1 1	
B/D-5*	0 2 4		0 0 0 1 0 1 0 0	
	1 2 4		0 1 0 1 0 1 0 0	
	2 2 4		1 0 0 1 0 1 0 0	
	3 2 4		1 1 0 1 0 1 0 0	
B/D-6*	0 2 5		0 0 0 1 0 1 0 1	
	1 2 5		0 1 0 1 0 1 0 1	
	2 2 5		1 0 0 1 0 1 0 1	
	3 2 5		1 1 0 1 0 1 0 1	
B/D-7*	0 2 6		0 0 0 1 0 1 1 0	
	1 2 6		0 1 0 1 0 1 1 0	
	2 2 6		1 0 0 1 0 1 1 0	
	3 2 6		1 1 0 1 0 1 1 0	
B/D-8*	0 2 7		0 0 0 1 0 1 1 1	
	1 2 7		0 1 0 1 0 1 1 1	
	2 2 7		1 0 0 1 0 1 1 1	
	3 2 7		1 1 0 1 0 1 1 1	
		Mode Commands		
Mode	Octal		1 2 3 4 5 6 7 8	
Real Time	0 6 0		0 0 1 1 0 0 0 0	
	1 6 0		0 1 1 1 0 0 0 0	
	2 6 0		1 0 1 1 0 0 0 0	
	3 6 0		1 1 1 1 0 0 0 0	
Memory Readout	0 6 1		0 0 1 1 0 0 0 1	
	1 6 1		0 1 1 1 0 0 0 1	
	2 6 1		1 0 1 1 0 0 0 1	
	3 6 1		1 1 1 1 0 0 0 1	

* Note that formats D-4 to D-8 can be commanded, however, there are no words assigned to these formats.

Table 7. (continued)

Mode	Command Numbers										
	Octal	(Bits 1 & 2 not used by DTU)	Binary								(Last bit to enter DTU)
			1	2	3	4	5	6	7	8	
Telemetry Store	0 6 2		0	0	1	1	0	0	1	0	
	1 6 2		0	1	1	1	0	0	1	0	
	2 6 2		1	0	1	1	0	0	1	0	
	3 6 2		1	1	1	1	0	0	1	0	
Memory Halt	0 6 4		0	0	1	1	0	1	0	0	
	1 6 4		0	1	1	1	0	1	0	0	
	2 6 4		1	0	1	1	0	1	0	0	
	3 6 4		1	1	1	1	0	1	0	0	
Memory Continue	0 6 5		0	0	1	1	0	1	0	1	
	1 6 5		0	1	1	1	0	1	0	1	
	2 6 5		1	0	1	1	0	1	0	1	
	3 6 5		1	1	1	1	0	1	0	1	
Roll Index 0°	0 6 6		0	0	1	1	0	1	1	0	
	1 6 6		0	1	1	1	0	1	1	0	
	2 6 6		1	0	1	1	0	1	1	0	
	3 6 6		1	1	1	1	0	1	1	0	
Roll Index 180°	0 6 7		0	0	1	1	0	1	1	1	
	1 6 7		0	1	1	1	0	1	1	1	
	2 6 7		1	0	1	1	0	1	1	1	
	3 6 7		1	1	1	1	0	1	1	1	
Coded Data	0 7 0		0	0	1	1	1	0	0	0	
	1 7 0		0	1	1	1	1	0	0	0	
	2 7 0		1	0	1	1	1	0	0	0	
	3 7 0		1	1	1	1	1	0	0	0	
Uncoded Data	0 7 1		0	0	1	1	1	0	0	1	
	1 7 1		0	1	1	1	1	0	0	1	
	2 7 1		1	0	1	1	1	0	0	1	
	3 7 1		1	1	1	1	1	0	0	1	
Spin Averaging	0 7 7		0	0	1	1	1	1	1	1	
	1 7 7		0	1	1	1	1	1	1	1	
	2 7 7		1	0	1	1	1	1	1	1	
	3 7 7		1	1	1	1	1	1	1	1	
Non-Spin Averaging	0 7 6		0	0	1	1	1	1	1	0	
	1 7 6		0	1	1	1	1	1	1	0	
	2 7 6		1	0	1	1	1	1	1	0	
	3 7 6		1	1	1	1	1	1	1	0	
ACS Non-Spin Averaging	0 7 4		0	0	1	1	1	1	0	0	
	1 7 4		0	1	1	1	1	1	0	0	
	2 7 4		1	0	1	1	1	1	0	0	
	3 7 4		1	1	1	1	1	1	0	0	

3.5.2.10 Operating Modes. The data handling subsystem will operate in three basic modes:

Real Time
Telemetry Store
Memory Readout

All mode changes are synchronous with the start of the next mainframe after the receipt of a mode command. If modes are changed in less than 12 seconds, the DTU will assume an indeterminate state when operating at 16 bits/sec.

a) Real Time Mode. In the real time mode, data are transmitted directly without intermediate storage at a bit rate selected by ground command.

b) Telemetry Store Mode. In the telemetry store mode, formatted NRZ data are presented to the DSU for storage at one of the selected bit rates. Consecutive frames of data are simultaneously transmitted and stored on a 100% duty cycle basis until either the DSU is full or the DTU is commanded to the real time mode. Upon receipt of the telemetry store command, the DSU memory address will reset to zero. When the memory has been filled, the DTU will automatically switch to the real time mode and remain in the format and bit rate used during telemetry store. The DSU memory can be partially filled and data stored at a later time beginning at the memory location of the last previously stored data by the use of the memory halt and memory continue commands.

c) Memory Readout Mode. In the memory readout mode, formatted NRZ data from the DSU are presented to the DTU for convolutional coding and transmission to earth at a bit rate selected by ground command. Upon receipt of the memory readout command, the DSU memory address will reset to zero and sequentially unload the stored data in the same sequence as stored. When the memory has been completely readout, the DTU will automatically switch to the real time mode in the format used before memory readout and remain in the bit rate used during memory readout. The DSU memory can be partially readout, return to real time mode operation, and then continue the readout of the memory by using the memory halt and memory continue commands.

3.5.2.11 Bit Rates. The DTU will operate in one of the following eight data bit rates.

a) 16 bits/sec

b) 32 bits/sec

c) 64 bits/sec

d) 128 bits/sec

e) 256 bits/sec

f) 512 bits/sec

g) 1024 bits/sec

h) 2048 bits/sec

The operating data bit rate is the rate at which input data to the DTU will be sampled and formatted PCM data will be transmitted.

The coded bit rate is twice the operating data bit rate. The data bit rate is selectable by ground command and is independent of format and mode. Changes in bit rate occur within one bit period after the DTU receives a bit rate command and, therefore, can occur at any time with respect to the data being telemetered. Tables 2 and 3 tabulate the time required to telemeter the various formats, store and readout data from memory for each operating bit rate.

3.5.2.12 Word Organization. The basic word size is 3 bits for mainframe words and 6 bits for subframes. Status or bi-level data (subframe only) is accepted as single bits only (one per channel) and formatted into groups of six bits. All analog data (subframe only) is encoded to 6-bit words. Spacecraft generated words are transmitted with the most significant bit first.

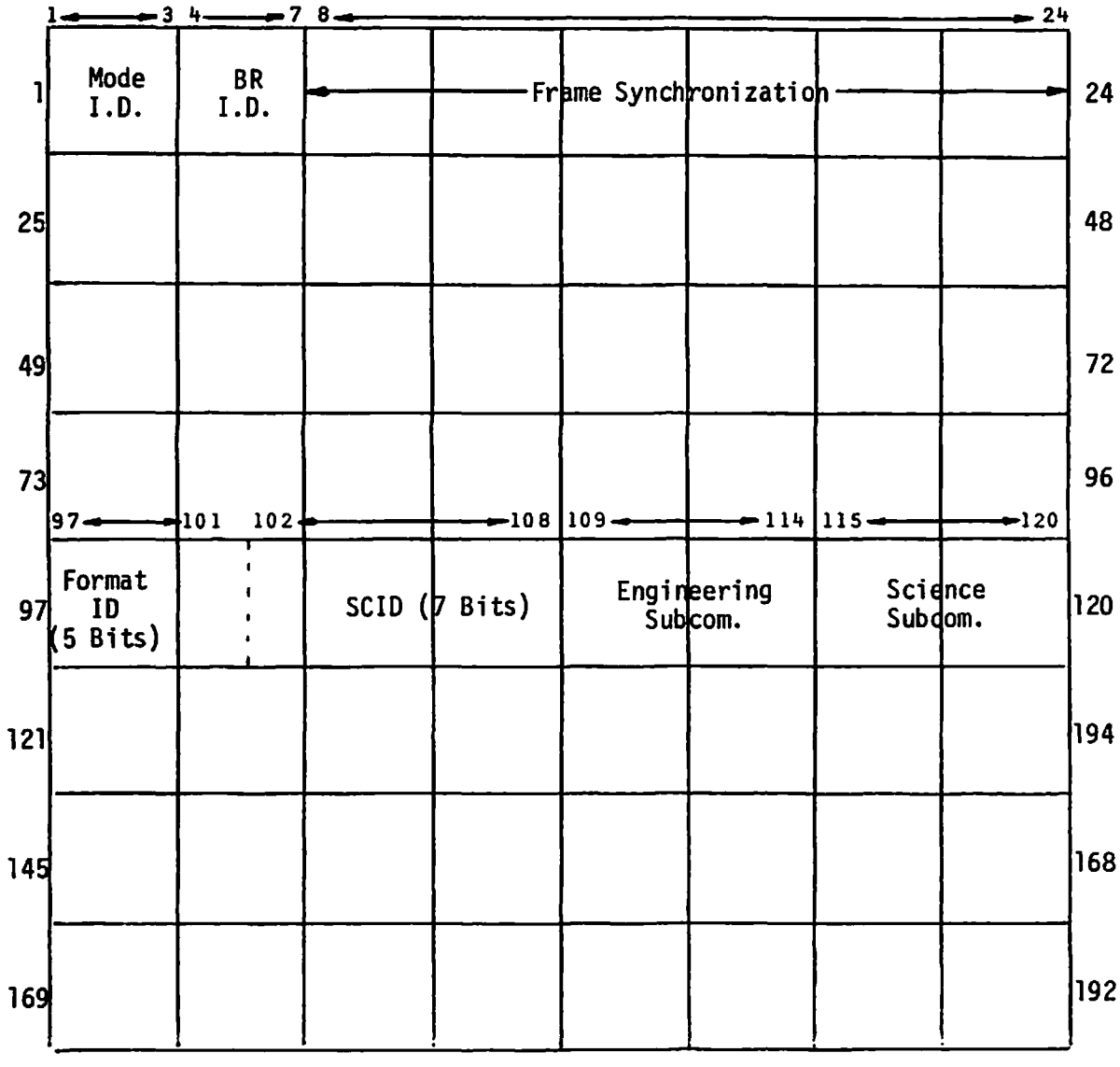
3.5.2.13 Mainframe Organization. The DTU assembles input information into frames composed of a series of 192 bits. Formats are arrangements of a frame. The mainframe format arrangement is shown in Figure 3.5-11. Bits 1 through 24 and 97 through 120 are reserved for fixed words (except for the D Formats). A total of 48-3 bit words can be assigned to a maximum of 12 serial digital data channels for each of two mainframe Formats A and B. Each redundant DTU has a patch plug which allows for the rearrangement of the science formats A and B without disassembling and rewiring the DTU. In the case of the D formats, all 192 bits are assigned to a single scientific instrument on a single data channel, with the exception of Format D-2 where two instruments share the same format.

3.5.2.14 Fixed Mainframe Words. Fixed words are included in each mainframe for purposes of synchronization and DTU mode, bit rate, format and subcommutator identification.

a) Frame Synchronization. Frame synchronization is an 18-bit code which occurs in bit positions 7 through 24 of each mainframe. The code is as follows:

Mainframe Word No.	3	4	5	6	7	8
	111	100	110	101	000	000

b) Mode Identification. The operational modes of the DTU are identified in the first 3 bits of each mainframe as follows:



NOTES:
1. Formats A and B are the basic scientific formats with a word size of 3 bits.
2. SCID = Subcommutator Identification
3. Each blank word slot denotes a serial digital word slot. Formats A and B are both independently patchable in the data portion of the mainframe. Each of these two formats handle a maximum of 12 digital input channels.
4. Each mainframe word slot is sampled at a rate of BR/192, where BR = bit rate.

Figure 3.5-11. Mainframe Organization

	DTU-A ON			DTU-B ON		
	Bit 1		Bit 3	Bit 1		Bit 3
Real Time	0	0	0	0	0	1
Memory Readout	0	1	0	0	1	1
Telemetry Store	1	0	0	1	0	1

c) Bit Rate Identification. The DTU operating data bit rate is identified in bits 4 through 6 of each mainframe as follows:

	Bit 4		Bit 6
16 BPS	0	0	0
32 BPS	0	0	1
64 BPS	0	1	0
128 BPS	0	1	1
256 BPS	1	0	0
512 BPS	1	0	1
1024 BPS	1	1	0
2048 BPS	1	1	1

d) Format Identification. The formats are identified in bits 97 through 101 of each mainframe as follows:

Format	Bit 97				Bit 101
A	0	1	0	0	0
A	0	1	0	0	1
B	0	0	0	0	0
B	0	0	0	0	1
C-1	0	0	1	0	0
C-1	0	1	1	0	0
C-2	0	0	1	0	1
C-2	0	1	1	0	1
C-3	0	0	1	1	0
C-3	0	1	1	1	0
C-4	0	0	1	1	1
C-4	0	1	1	1	1
A/D-1	1	1	0	0	0
A/D-2	1	1	0	0	1
A/D-3	1	1	0	1	0
B/D-1	1	0	0	0	0
B/D-2	1	0	0	0	1
B/D-3	1	0	0	1	0

Format ID for the C format is identical to the format ID for C-1 through C-4. That is, bit Nos. 100 and 101 will sequence through the binary count 00 through 11 as the C format sequences through formats C-1 through C-4.

Although the DTU is implemented to operate in Formats A/D-1 through A/D-8 and B/D-1 through B/D-8, only formats D-1 through D-3 are being used on the Pioneer F mission. The format identification codes are derived from the format

commands. The five bit format ID code corresponds directly bit by bit, with the least significant five bits of each format command number.

e) Subcommutator Identification. The subcommutator position is identified by bits 102 through 108 of each mainframe. The total of 7 bits is derived from a divide-by-128 subframe counter, which is advanced by one count per subframe for a 128 word subcommutator. Bit 102 is the most significant bit for the 128 word engineering subcommutator. Bits 103 through 108 also identify the 64 word science subcommutator with the most significant bit first.

3.5.2.15 Engineering Subcommutator. The DTU provides in each mainframe an engineering subcommutator consisting of 128-6 bit words. The engineering subcommutator appears in bits 109 through 114 of the mainframe. Information telemetered in this subcommutator is obtained from various spacecraft engineering instrumentation such as thermistors, voltage and current monitors, switch positions, etc. Analog, digital, and status (bi-level) information is accepted by the DTU for telemetering in the engineering subcom. The assignment of analog, digital, and bi-level channels within the subcom is shown in Figure 3.5-12. The word slots marked by an X are permanently assigned (hard wired) to measurements made within the DTU. Bi-level bit 1 of word 32 is also internally wired. The word assignment for the engineering subcomm is listed in Figure 3.5-13 sheets 1 through 11.

3.5.2.16 Roll Attitude Timing. The DTU contains a redundant 12-bit counter which is driven by a 128 BPS clock and controlled by the roll reference pulse. The 12 bits of the counter are read out with the most significant bit first during engineering subcomm words C-112 and C-116. The counter is reset with each occurrence of the roll reference pulse, except during telemetry readout, and advanced by the 128 BPS clock. The clock is inhibited until the end of subcomm word C-125. The count stops on word C-112, is reset on the first roll pulse after the end of word gate C-125. Counting is then resumed. The 12-bit count represents the time between the occurrence of a roll ~~reference~~ index pulse and the start of word C-112 of the engineering subcomm. This telemetered time will permit correlating the attitude of the roll-index reference line with given telemetered science and engineering data. The redundant roll attitude timer can be used only by commanding the redundant DTU. Whenever Format C or C1 are commanded, this information is telemetered at the mainframe rate only and engineering subcomm words C-112 and C-116 will contain zeros. Each count within the register represents 0.0078125 seconds.

3.5.2.17 Definition of Specific Telemetry Words within Engineering Subcomm.

a) Stored Command Identification. Engineering words C-305, C-306, and C-307 contain the stored execute delay time and the command number of one of five stored commands within the CDU. To identify which of the five commands and its delay time are being telemetered in any given engineering subcom bits 4, 5, and 6 of word C-307 are coded to represent the five commands.

1	A/D Low	Calib. Med.	Volt High	Ext. SCID	A	A	A	B	8	C1
9	A	A	A	X	A	A	A	X	16	
17	A	A	A	B	A	A	A	B	24	
25	A	A	A	B	A	A	A	B	32	
33	A	A	A	A	A	A	A	A	40	C2
41	A	A	A	A	A	A	A	A	48	
49	A	A	A	A	A	A	A	A	56	
57	A	A	A	A	A	A	A	A	64	
65	A	A	A	A	D	D	D	B	72	C3
73	A	A	A	A	D	D	D	B	80	
81	A	A	A	A	D	D	D	B	88	
89	A	A	A	A	D	D	D	B	96	
97	B	A	D	D	X	X	X	X	104	C4
105	B	D	D	D	D	D	D	D	112	
113	B	A	D	D	D	D	D	D	120	
121	D	D	D	D	D	D	D	D	128	

Notes:

1. The engineering subcommutator is located in word slot Nos. 37 and 38 of the main frame, Format A or B, with a 6-bit word size.
2. Each subframe is sampled at a nominal rate of BR/24, 576 where BR = bit rate (bit/sec), when this subcommutator is used in Format A or B.
3. A = analog data input.
4. D = 6-bit serial digital word.
5. B = six bilevel inputs.
6. X = words supplied by the fixed word generator:
   a. Roll attitude timing: Word Nos. 12 and 16, 12 bits.
   b. Spin rate and phase indicator: Word Nos. 101 thru 104, 24 bits.
   The first 5 bits of word 104 contains phase and bit 6 contains the sign bit.
7. Word Nos. 1 through 4 have internally hardwired multiplexed data information as indicated (three analog and 6 bilevel channels).
8. Bit No. 1, word No. 32,represents an internally wired bilevel channel for convolutional coder ON/OFF indication.
9. Bit Nos. 1 thru 3, word 113 contain spin period sector generator status bits.

Figure 3.5-12. Engineering Subcommutator

Word	Bit	Measurement	Type
C-101		DTU A/D Calibration Voltage (Low) 168 mVdc	A
C-102		DTU A/D Calibration Voltage (Med) 1.512 Vdc	A
C-103		DTU A/D Calibration Voltage (High) 2.424 Vdc	A
C-104	1	Extended SCID-MSB (2^{12})	B
	2	Extended SCID (2^{11})	B
	3	Extended SCID (2^{10})	B
	4	Extended SCID (2^{9})	B
	5	Extended SCID (2^{8})	B
	6	Extended SCID-LSB (2^{7})	B
C-105		RTG 2 Current 0-11A	A
C-106		Battery Voltage 0-15 Vdc	A
C-107		DC Bus Voltage 26-30 Vdc	A
C-108	1	JPL/HVM Power On/Off (On=1)	B
	2	ARC/PA Power On/Off (On=1)	B
	3	UC/CPI Power On/Off (On=1)	B
	4	UI/GTT Power On/Off (On=1)	B
	5	GSFC/CRT Power On/Off (On=1)	B
	6		B
C-109		Battery Charge Current 0-0.3A	A
C-110		RTG 1 Voltage 0-6 Vdc	A
C-111		Receiver A AGC Conscan +4 dB AM to -4 dB AM	A
C-112	1	Roll Attitude Timer-MSB (2^{13})	B
	2	Roll Attitude Timer (2^{12})	B
	3	Roll Attitude Timer (2^{9})	B
	4	Roll Attitude Timer (2^{8})	B
	5	Roll Attitude Timer (2^{7})	B
	6	Roll Attitude Timer (2^{6})	B
C-113		RTG 4 Voltage 0-6 Vdc	A
C-114		RTG 3 Current 0-11A	A
C-115		Battery Temperature -20°F to +120°F	A
C-116	1	Roll Attitude Timer (2^{5})	B
	2	Roll Attitude Timer (2^{4})	B
	3	Roll Attitude Timer (2^{3})	B
	4	Roll Attitude Timer (2^{2})	B
	5	Roll Attitude Timer (2^{1})	B
	6	Roll Attitude Timer-LSB (2^{0})	B
C-117		TRF +5 Volt Output CDU Bus A 0-6 Vdc	A
C-118		TRF +5 Volt Output CDU Bus B 0-6 Vdc	A
C-119		DC Bus Voltage 0-30 Vdc	A
C-120	1	JPL/HVM Boom Released (Released=0)	B
	2	RTG 1/2 Deployed (Deployed=0)	B
	3	RTG 3/4 Deployed (Deployed=0)	B
	4	S/C Separation Status (Separation=0)	B
	5	Decoder A Addressed (Addressed=1)	B
	6	Decoder B Addressed (Addressed=1)	B

Figure 3.5-13. Word Assignments for Format C (Page 1 of 11)

Word	Bit	Measurement	Type
C-121		Receiver B AGC Conscan +4 dB AM to -4 dB AM	A
C-122		Shunt Bus Current 0-3A	A
C-123		RTG 4 Current 0-11A	A
C-124	1	UCSD/TRD Power On/Off (On=1)	B
	2	USC/UV Power On/Off (On=1)	B
	3	UA/IPP Power On/Off (On=1)	B
	4	CIT/IR Power On/Off (On=1)	B
	5	GE/AMD Power On/Off (On=1)	B
	6	LaRC/MD Power On/Off (On=1)	B
C-125		RTG 2 Voltage 0-6 Vdc	A
C-126		Battery Discharge Current 0-10A	A
C-127		RTG 1 Current 0-11A	A
C-128	1	Battery Charge Status (Auto=0, Float=1)	B
	2	Battery Discharge Status (Enable=0)	B
	3		B
	4		B
	5	Ordnance Relay Status Prime (Arm=1)	B
	6	Ordnance Relay Status Redundant (Arm=1)	B
C-129		DC Bus Current 0-6A	A
C-130			A
C-131		RTG 3 Voltage 0-6 Vdc	A
C-132	1	Coded/Uncoded Data (Coded=1)	B
	2	RTG 1/2 Ordnance Status (Safe=0)	B
	3	RTG 3/4 Ordnance Status (Safe=0)	B
	4	RTG 1/2 Redundant Ordnance Status (Safe=0)	B
	5	RTG 3/4 Redundant Ordnance Status (Safe=0)	B
	6	CDU Sequencer Status (Enabled=1)	B

Figure 3.5-13. (Page 2 of 11)

Word	Bit	Measurement	Type
C-201		RTG 1 Fin Root Temperature 160°F to 360°F	A
C-202		RTG 2 Fin Root Temperature 160°F to 360°F	A
C-203		RTG 3 Fin Root Temperature 160°F to 360°F	A
C-204		RTG 4 Fin Root Temperature 160°F to 360°F	A
C-205		TWT A Temperature 40°F to 125°F	A
C-206		Driver A Temperature 20°F to 110°F	A
C-207		TWT A Converter Temperature 40°F to 125°F	A
C-208		TWT A Cathode Current 24 to 30 mA	A
C-209		Shunt Bus Current 0-3A	A
C-210		Propellant Supply Pressure 0-600 PSIA	A
C-211		TWT A Helix Current 0 to 10 mA	A
C-212		Receiver A Loop Stress -100 kHz to +100 kHz	A
C-213		Receiver B Signal Strength -149 dBm to -63 dBm	A
C-214		TWT B RF Output Power 26 to 40.4 dBm	A
C-215		TWT B Cathode Current 24 to 30 mA	A
C-216		TWT B Helix Current 0 to 10 mA	A
C-217		RTG 4 Hot Junction Temperature 880°F to 1200°F	A
C-218		RTG 3 Hot Junction Temperature 880°F to 1200°F	A
C-219		RTG 2 Hot Junction Temperature 880°F to 1200°F	A
C-220		RTG 1 Hot Junction Temperature 880°F to 1200°F	A
C-221		TWT B Converter Temperature 40°F to 125°F	A
C-222		Receiver A VCO Temperature 20°F to 110°F	A
C-223		Driver B Temperature 20°F to 110°F	A
C-224		TWT A Reference Voltage 0-28 Vdc	A
C-225			A
C-226			A
C-227		Receiver B VCO Temperature 20°F to 110°F	A
C-228		TWT B Temperature 40°F to 125°F	A
C-229		Receiver B Loop Stress -100 kHz to +100 kHz	A
C-230		TWT B Reference Voltage 0-28 Vdc	A
C-231		TWT A RF Output Power 26 to 40.4 dBm	A
C-232		Receiver A Signal Strength -149 dBm to -63 dBm	A

Figure 3.5-13. (Page 3 of 11)

Word	Bit	Measurement	Type
C-301		S/C Platform Temperature 1 0°F to 140°F	A
C-302		S/C Platform Temperature 2 0°F to 140°F	A
C-303		SRA Temperature -10°F to +95°F	A
C-304		S/C Platform Temperature 3 0°F to 140°F	A
C-305	1	Stored Command Time Register-MSB (2^6)	D
	2	Stored Command Time Register (2^5)	D
	3	Stored Command Time Register (2^4)	D
	4	Stored Command Time Register (2^3)	D
	5	Stored Command Time Register (2^2)	D
	6	Stored Command Time Register (2^1)	D
C-306	1	Stored Command Time Register-LSB (2^0)	D
	2	Stored Command Register-MSB (2^7)	D
	3	Stored Command Register (2^6)	D
	4	Stored Command Register (2^5)	D
	5	Stored Command Register (2^4)	D
	6	Stored Command Register (2^3)	D
C-307	1	Stored Command Register (2^2)	D
	2	Stored Command Register (2^1)	D
	3	Stored Command Register-LSB (2^0)	D
	4	Stored Command Identification-Bit 1	D
	5	Stored Command Identification-Bit 2	D
	6	Stored Command Identification-Bit 3	D
C-308	1	Receiver A Signal Present (Signal Present=1)	B
	2	Receiver B Signal Present (Signal Present=1)	B
	3	Receiver A Oscillator Enabled/Disabled (Enabled=1)	B
	4	Receiver B Oscillator Enabled/Disabled (Enabled=1)	B
	5	Spin Thruster B Pulse Count	B
	6	Spin Thruster A Pulse Count	B
C-309		Velocity Thruster Cluster 1 Temperature 40°F to 200°F	A
C-310		Spin Thruster Cluster Temperature 40°F to 200°F	A
C-311		VPT 1 Thruster Temperature 400°F to 1800°F	A
C-312		VPT 2 Thruster Temperature 400°F to 1800°F	A
C-313	1	Conscan-Thruster Phase Output Status (0°=0, 180°=1)	D
	2	Conscan Threshold Mode Status (Hi=1, Med=0)	D
	3	Conscan A Sin θ - LSB (2^0)	D
	4	Conscan A Sin θ (2^1)	D
	5	Conscan A Sin θ (2^2)	D
	6	Conscan A Sin θ (2^3)	D
C-314	1	Conscan A Sin θ (2^4)	D
	2	Conscan A Sin θ (2^5)	D
	3	Conscan A Sin θ (2^6)MSB	D
	4	Conscan A Sin θ - Sign Bit	D
	5	Conscan A Cos θ - LSB (2^0)	D
	6	Conscan A Cos θ (2^1)	D

Figure 3.5-13. (Page 4 of 11)

Word	Bit	Measurement	Type
C-315	1	Conscan A Cos θ (2^2)	D
	2	Conscan A Cos θ (2^3)	D
	3	Conscan A Cos θ (2^4)	D
	4	Conscan A Cos θ (2^5)	D
	5	Conscan A Cos θ (2^6)MSB	D
	6	Conscan A Cos θ - Sign Bit	D
C-316	1	Conscan Power On/Off (On=1)	B
	2	Conscan Threshold (Above=1)	B
	3		B
	4	Receiver Switch Status (A/B=Hi/Med=1),(A/B=Med/Hi=0)	B
	5	Transmitter Switch Status (A/B=Med/Hi=1),(A/B=Hi/Med=0)	B
	6	Antenna Feed Switch Status (Norm=0, Offset=1)	
C-317		SSA Temperature -30°F to +194°F	A
C-318		S/C Platform Temperature 4 0°F to 140°F	A
C-319		S/C Platform Temperature 5 -20°F to +110°F	A
C-320		S/C Platform Temperature 6 -20°F to +110°F	A
C-321	1	Velocity Thruster 2 (1B) Pulse Count - MSB (2^5)	D
	2	Velocity Thruster 2 (1B) Pulse Count (2^4)	D
	3	Velocity Thruster 2 (1B) Pulse Count (2^3)	D
	4	Velocity Thruster 2 (1B) Pulse Count (2^2)	D
	5	Velocity Thruster 2 (1B) Pulse Count (2^1)	D
	6	Velocity Thruster 2 (1B) Pulse Count - LSB (2^0)	D
C-322	1	Velocity Thruster 4 (2A) Pulse Count - MSB (2^5)	D
	2	Velocity Thruster 4 (2A) Pulse Count (2^4)	D
	3	Velocity Thruster 4 (2A) Pulse Count (2^3)	D
	4	Velocity Thruster 4 (2A) Pulse Count (2^2)	D
	5	Velocity Thruster 4 (2A) Pulse Count (2^1)	D
	6	Velocity Thruster 4 (2A) Pulse Count - LSB (2^0)	D
C-323	1		D
	2		D
	3		D
	4		D
	5		D
	6		D
C-324	1	Command Memory Status (Processing=0, Standby=1)	B
	2		B
	3		B
	4		B
	5		B
	6	Command Execute (Toggle=1 or 0)	B
C-325		VPT 4 Thruster Temperature 400°F to 1800°F	A
C-326		Velocity Thruster Cluster 2 Temperature 40°F to 200°F	A
C-327		Propellant Supply Temperature 40°F to 160°F	A
C-328		VPT 3 Thruster Temperature 400°F to 1800°F	A

Figure 3.5-13. (Page 5 of 11)

Word	Bit	Measurement	Type
C-329	1	Velocity Thruster 1 (1A) Pulse Count - MSB (2^5)	D
	2	Velocity Thruster 1 (1A) Pulse Count (2^4)	D
	3	Velocity Thruster 1 (1A) Pulse Count (2^3)	D
	4	Velocity Thruster 1 (1A) Pulse Count (2^2)	D
	5	Velocity Thruster 1 (1A) Pulse Count (2^1)	D
	6	Velocity Thruster 1 (1A) Pulse Count - LSB (2^0)	D
C-330	1	Velocity Thruster 3 (2B) Pulse Count - MSB (2^5)	D
	2	Velocity Thruster 3 (2B) Pulse Count (2^4)	D
	3	Velocity Thruster 3 (2B) Pulse Count (2^3)	D
	4	Velocity Thruster 3 (2B) Pulse Count (2^2)	D
	5	Velocity Thruster 3 (2B) Pulse Count (2^1)	D
	6	Velocity Thruster 3 (2B) Pulse Count - LSB (2^0)	D
C-331	1		D
	2		D
	3		D
	4		D
	5		D
	6		D
C-332	1	Sequencer Power On/Off (On=1)	B
	2	Overload Protection On/Off (On=0)	B
	3	Receiver Reverse Inhibit (Inhibit=1)	B
	4	Command Processor Memory A Select/B Select (A=1)	B
	5	Command Memory DTU Identification (DTU=1)	B
	6	CDU +5V Bus Status A/B (Bus A=1)	B

Figure 3.5-13. (Page 6 of 11)

Word	Bit	Measurement	Type
C-401	1		B
	2		B
	3		B
	4		B
	5		B
	6		B
C-402			A
C-403	1	Precession Pair: VPT 1&4/VPT 2&3 (VPT 1&4 = 1)	D
	2	Pulse Length - Bit 1	D
	3	Pulse Length - Bit 2	D
	4	Pulse Length - Bit 3	D
	5	Delta V Pair: VPT 1&3/VPT 2&4 (VPT 1&3 = 1)	D
	6	Spin Control Direction Up/Down (Up=0)	D
C-404	1	Star Time Gate - LSB (2^0)	D
	2	Star Time Gate (2^1)	D
	3	Star Time Gate (2^2)	D
	4	Star Time Gate (2^3)	D
	5	Star Time Gate - MSB (2^4)	D
	6	Delta V/SCT Mode Enabled/Disabled (Enabled=1)	D
C-405	1	Spin Period - MSB (2^{17})	D
	2	Spin Period (2^{16})	D
	3	Spin Period (2^{15})	D
	4	Spin Period (2^{14})	D
	5	Spin Period (2^{13})	D
	6	Spin Period (2^{12})	D
C-406	1	Spin Period (2^{11})	D
	2	Spin Period (2^{10})	D
	3	Spin Period (2^9)	D
	4	Spin Period (2^8)	D
	5	Spin Period (2^7)	D
	6	Spin Period (2^6)	D
C-407	1	Spin Period (2^5)	D
	2	Spin Period (2^4)	D
	3	Spin Period (2^3)	D
	4	Spin Period (2^2)	D
	5	Spin Period (2^1)	D
	6	Spin Period - LSB (2^0)	D
C-408	1	Roll Pulse/Roll-Index Pulse Phase Error - MSB (2^4)	D
	2	Roll Pulse/Roll-Index Pulse Phase Error (2^3)	D
	3	Roll Pulse/Roll-Index Pulse Phase Error (2^2)	D
	4	Roll Pulse/Roll-Index Pulse Phase Error (2^1)	D
	5	Roll Pulse/Roll-Index Pulse Phase Error - LSB (2^0)	D
	6	Phase Error Sign - (Roll Pulse before Index Pulse = 1)	D

Figure 3.5-13. (Page 7 of 11)

Word	Bit	Measurement	Type
C-409	1	VPT 1 Firing Status (Firing =1)	B
	2	VPT 2 Firing Status (Firing =1)	B
	3	VPT 4 Firing Status (Firing =1)	B
	4	VPT 3 Firing Status (Firing =1)	B
	5	SCT 1 Firing Status (Firing =1)	B
	6	SCT 2 Firing Status (Firing =1)	B
C-410	1	Despin On/Off (On=1)	D
	2	Conscan Enabled/Disabled (Enabled=1)	D
	3	Clock Select A (Selected =1)	D
	4	Clock Select B (Selected =1)	D
	5	Star Angle Gate 45°/360° (45° =1)	D
	6	Star Level >180% Canopus Yes/No (Yes=1)	D
C-411	1	No. 1 Precession Magnitude - LSB (2^0)	D
	2	No. 1 Precession Magnitude (2^1)	D
	3	No. 1 Precession Magnitude (2^2)	D
	4	No. 1 Precession Magnitude (2^3)	D
	5	No. 1 Precession Magnitude (2^4)	D
	6	No. 1 Precession Magnitude (2^5)	D
C-412	1	No. 1 Precession Magnitude (2^6)	D
	2	No. 1 Precession Magnitude (2^7)	D
	3	No. 1 Precession Magnitude (2^8)	D
	4	No. 1 Precession Magnitude (2^9)	D
	5	No. 1 Precession Magnitude - MSB (2^{10})	D
	6	Delta V Magnitude - LSB (2^0)	D
C-413	1	Delta V Magnitude (2^1)	D
	2	Delta V Magnitude (2^2)	D
	3	Delta V Magnitude (2^3)	D
	4	Delta V Magnitude (2^4)	D
	5	Delta V Magnitude (2^5)	D
	6	Delta V Magnitude (2^6)	D
C-414	1	Delta V Magnitude (2^7)	D
	2	Delta V Magnitude (2^8)	D
	3	Delta V Magnitude (2^9)	D
	4	Delta V Magnitude (2^{10})	D
	5	Delta V Magnitude (2^{11})	D
	6	Delta V Magnitude - MSB (2^{12})	D
C-415	1	No. 2 Precession Magnitude - LSB (2^0)	D
	2	No. 2 Precession Magnitude (2^1)	D
	3	No. 2 Precession Magnitude (2^2)	D
	4	No. 2 Precession Magnitude (2^3)	D
	5	No. 2 Precession Magnitude (2^4)	D
	6	No. 2 Precession Magnitude (2^5)	D

Figure 3.5-13. (Page 8 of 11)

Word	Bit	Measurement	Type
C-416	1	No. 2 Precession Magnitude (2^6)	D
	2	No. 2 Precession Magnitude (2^7)	D
	3	No. 2 Precession Magnitude (2^8)	D
	4	No. 2 Precession Magnitude (2^9)	D
	5	No. 2 Precession Magnitude - MSB (2^{10})	D
	6	Star Coincidence Yes/No (Yes=1)	D
C-417	1	SPSG Roll Reference 0°/180° (0°=0)	B
	2	SPSG Mode - Bit 1	B
	3	SPSG Mode - Bit 2	B
	4		B
	5		B
	6		B
C-418			A
C-419	1	Star Delay - LSB (2^0)	D
	2	Star Delay (2^1)	D
	3	Star Delay (2^2)	D
	4	Star Delay (2^3)	D
	5	Star Delay (2^4)	D
	6	Star Delay (2^5)	D
C-420	1	Star Delay (2^6)	D
	2	Star Delay (2^7)	D
	3	Star Delay (2^8)	D
	4	Star Delay (2^9)	D
	5	Star Delay (2^{10})	D
	6	Star Delay - MSB (2^{11})	D
C-421	1	Star Count - LSB (2^0)	D
	2	Star Count (2^1)	D
	3	Star Count - MSB (2^2)	D
	4	CEA Power Status - Bit 1 (DSLA On = 1)	D
	5	CEA Power Status - Bit 2 (DSLB On = 1)	D
	6	CEA Power Status - Bit 3 (PSE On = 1)	D
C-422	1	Spacing Bit	D
	2	Spacing Bit	D
	3	Spacing Bit	D
	4	Spacing Bit	D
	5	Spacing Bit	D
	6	No. 2 Precession Redundant Magnitude - LSB (2^0)	D
C-423	1	No. 2 Precession Redundant Magnitude (2^1)	D
	2	No. 2 Precession Redundant Magnitude (2^2)	D
	3	No. 2 Precession Redundant Magnitude (2^3)	D
	4	No. 2 Precession Redundant Magnitude (2^4)	D
	5	No. 2 Precession Redundant Magnitude (2^5)	D
	6	No. 2 Precession Redundant Magnitude (2^6)	D

Figure 3.5-13. (Page 9 of 11)

Word	Bit	Measurement	Type
C-424	1	No. 2 Precession Redundant Magnitude - MSB (2^7)	D
	2	No. 2 Precession Angle - LSB (2^0)	D
	3	No. 2 Precession Angle (2^1)	D
	4	No. 2 Precession Angle (2^2)	D
	5	No. 2 Precession Angle (2^3)	D
	6	No. 2 Precession Angle (2^4)	D
C-425	1	No. 2 Precession Angle (2^5)	D
	2	No. 2 Precession Angle (2^6)	D
	3	No. 2 Precession Angle (2^7)	D
	4	No. 2 Precession Angle (2^8)	D
	5	Delta V Redundant Magnitude - LSB (2^0)	D
	6	Delta V Redundant Magnitude (2^1)	D
C-426	1	Delta V Redundant Magnitude (2^2)	D
	2	Delta V Redundant Magnitude (2^3)	D
	3	Delta V Redundant Magnitude (2^4)	D
	4	Delta V Redundant Magnitude (2^5)	D
	5	Delta V Redundant Magnitude (2^6)	D
	6	Delta V Redundant Magnitude - MSB (2^7)	D
C-427	1	Time Delay - LSB (2^0)	D
	2	Time Delay (2^1)	D
	3	Time Delay (2^2)	D
	4	Time Delay (2^3)	D
	5	Time Delay (2^4)	D
	6	Time Delay (2^5)	D
C-428	1	Time Delay - MSB (2^6)	D
	2	No. 1 Precession Redundant Magnitude - LSB (2^0)	D
	3	No. 1 Precession Redundant Magnitude (2^1)	D
	4	No. 1 Precession Redundant Magnitude (2^2)	D
	5	No. 1 Precession Redundant Magnitude (2^3)	D
	6	No. 1 Precession Redundant Magnitude (2^4)	D
C-429	1	No. 1 Precession Redundant Magnitude (2^5)	D
	2	No. 1 Precession Redundant Magnitude (2^6)	D
	3	No. 1 Precession Redundant Magnitude - MSB (2^7)	D
	4	No. 1 Precession Angle - LSB (2^0)	D
	5	No. 1 Precession Angle (2^1)	D
	6	No. 1 Precession Angle (2^2)	D
C-430	1	No. 1 Precession Angle (2^3)	D
	2	No. 1 Precession Angle (2^4)	D
	3	No. 1 Precession Angle (2^5)	D
	4	No. 1 Precession Angle (2^6)	D
	5	No. 1 Precession Angle (2^7)	D
	6	No. 1 Precession Angle - MSB (2^8)	D

Figure 3.5-13. (Page 10 of 11)

Word	Bit	Measurement	Type
C-431	1	ACS Sequence Status - Bit 1	D
	2	ACS Sequence Status - Bit 2	D
	3	ACS Sequence Status - Bit 3	D
	4	Star Time Gate Enabled/Disabled (Enabled = 1)	D
	5	Reference Select - Bit 1	D
	6	Reference Select - Bit 2	D
C-432	1	Star Location - Octant 1 Absent/Present (Absent = 0)	D
	2	Star Location - Octant 8 Absent/Present (Present = 1)	D
	3	ACS Registers Inhibit Status (Normal = 0)	D
	4	Precession Register 1 Arm/Disarm (Arm = 1)	D
	5	Delta V Register Arm/Disarm (Arm = 1)	D
	6	Precession Register 2 Arm/Disarm (Arm = 1)	D

Figure 3.5-13. (Page 11 of 11)

Word C-307 Bits 4 5 6	Stored Command Register No.
0 0 0	1
0 0 1	2
0 1 0	3
0 1 1	4
1 0 0	5

Whenever Format C or C3 are commanded, this information is telemetered at the mainframe rate only and bits 4, 5, and 6 of engineering subcomm word C-307 will contain all zeros or all ones.

In general, whenever an engineering digital word is accelerated to the mainframe rate, all zeros or all ones (whichever is on the output line from the source to the DTU) will be transmitted in the engineering subcomm position for that particular word.

b) Sequence Status. Bits 1, 2, and 3 of engineering word C-431 identify the sequence status for the open-loop series mode of the stored program operation of the ACS.

Word C-431 Bits 1 2 3	Sequence Status
0 0 0	Reset--State 0
1 0 0	No. 1 Delay--State 1
0 1 0	No. 1 Precession Mode--State 2
1 1 0	Delta V Delay--State 3
0 0 1	Delta V Mode--State 4
1 0 1	No. 2 Delay--State 5
0 1 1	No. 2 Precession Mode--State 6
1 1 1	Program Complete--State 7

Whenever Format C or C4 are commanded, this information is telemetered at the mainframe rate only and bits 1, 2, and 3 of engineering subcomm word C-431 will contain all zeros or all ones.

c) Reference Select. The roll reference source for the ACS and scientific instrument roll index pulse is identified in bits 5 and 6 of engineering word C-431.

Word C-431 Bits 5 6	Reference
1 1	Sun A
1 0	Sun B
0 1	Star

Whenever Format C or C4 are commanded, this information is telemetered at the mainframe rate only and bits 5 and 6 of engineering subcomm word C-431 will contain all zeros or all ones.

d) Star Location. Engineering word C-432 bits 1 and 2 determine where in an octant oriented ecliptic system the delayed star occurs.

Word C-432 Bits 1	2	Star Location
1	0	Sector 1
0	1	Sector 8

Whenever Format C or C4 are commanded, this information is telemetered at the mainframe rate only, and bits 1 and 2 of engineering subcomm word C-432 will contain all zeros or all ones.

e) Pulse Length. Bits 2, 3, and 4 of engineering word C-403 identify the time duration of thruster impulses.

Word C-403 Bits 2	3	4	Pulse Length
0	0	1	No. 1 - 31.25 msec
0	1	1	No. 2 - 125 msec
1	1	1	No. 3 - 500 msec
1	1	0	No. 4 - 1.0 sec
1	0	0	No. 5 - 2.0 sec

This information can only be telemetered at the engineering subcomm rate.

f) Spin Period Sector Generator Modes. The operating mode of the SPSG is determined by bits 2 and 3 of engineering word C-417.

Word C-417 Bits 2	3	SPSG Modes
0	0	Non-Spin Averaging
0	1	ACS
1	0	Spin Averaging

This information can only be telemetered at the engineering subcomm rate.

g) CEA Power Status. Engineering word C-421, bits 4, 5, and 6 identify the DC power to the four CEA modular slices.

Word C-421 Bits 4 5 6	CEA Power Status
0 0 0	Standby only (SPC On)
0 0 1	Storage PSE (SPC, PSE On)
1 0 0	DSL A (SPC, DSL A On)
1 0 1	PSE/DSL A (SPC, PSE, DSL A On)
0 1 1	PSE/DSL B (SPC, PSE, DSL B On)
0 1 0	DSL B (SPC, DSL B On)

Whenever Format C or C4 are commanded, this information is telemetered at the mainframe rate only and bits 4, 5, and 6 of engineering subcomm word C-421 will contain all zeros or all ones. For further information, refer to Section 3.3.3.2.

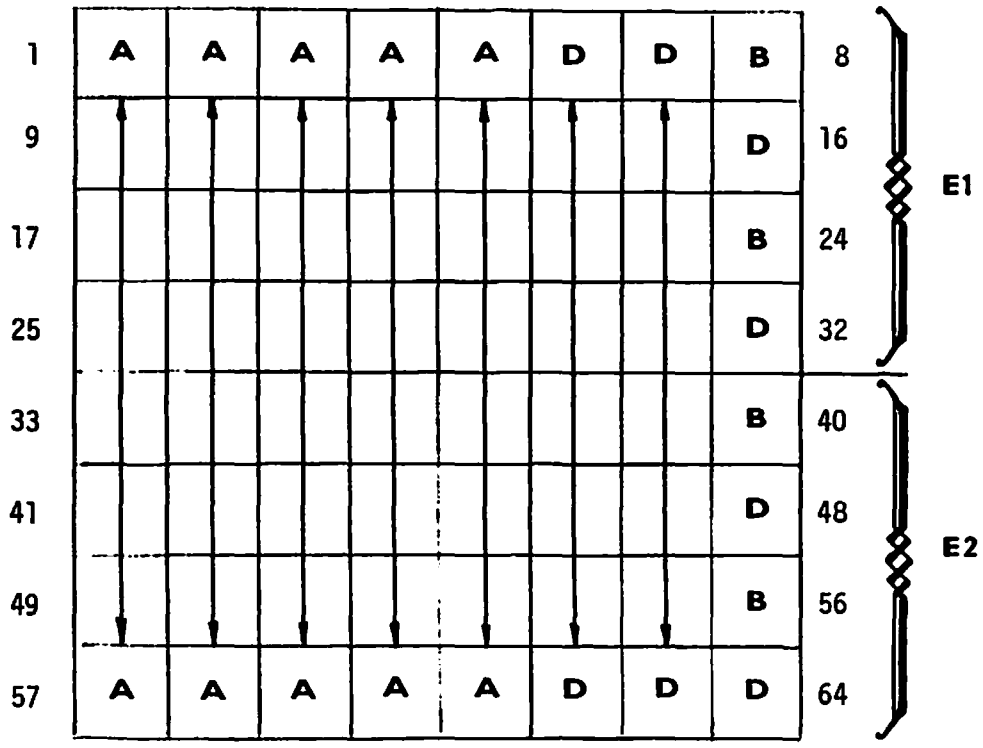
h) DTU Clock Select for CEA. Engineering word C-410, bits 3 and 4 identify the clocks selected for timing in the DSL of the CEA. For details, see Section 3.3.1.4.2, clock selection logic.

Word C-410 Bits 3 4	Clock Select Status
1 1	Clock Reset; both clocks used
0 1	Clock A select; only A used
0 0	Clock B select; clocks reversed
1 1	Clock Reset
1 0	Clock B select; only B used
0 0	Clock A select; clocks reversed

Normally, clock A drives clock countdown #1 and B drives clock countdown #2 (command ACR1). If both clocks are selected, this order will be reversed.

3.5.2.18 Science Subcommutator. The DTU also provides in each mainframe a science subcommutator consisting of 64-6 bit words. The science subcommutator appears in bits 115 through 120 of the mainframe. Analog, digital, and status (bi-level) information is accepted by the DTU from the scientific instruments for telemetering in the science subcomm. The assignment of analog, digital, and bi-level channels within the subcomm is shown in Figure 3.5-14. The word assignment for the Pioneer F mission is listed in Figure 3.5-15, Sheets 1 and 2.

3.5.2.19 Formats. The DTU is capable of operating in one of 23 format combinations. All formats are selectable by ground command, however, only 13 of the 23 format combinations are being used for Pioneer F. Changes in format occur within one bit period upon receipt of a format command by the DTU and, therefore, can occur at any time with respect to the data being telemetered. The buffer mode formats D1 and D2 are restricted to the real-time mode operation. All other formats can be operated in real-time, telemetry store, and memory readout modes, as required.



NOTES:

1. The science subcommutator is located in word slot #39 and #40 of the mainframe, Format A or B, with a 6-bit word size.
2. Each subframe is sampled at a nominal rate of BR/12,288, where BR = bit rate (bit/sec), when this subcommutator is used in Format A or B.
3. A = analog data input.
4. D = 6-bit serial digital word.
5. B = six bilevel inputs.

Figure 3.5-14. Science Subcommutator

Word	Bit	Measurement	Type
E-101		ARC/PA Detectors Temperature	A
E-102		ARC/PA Electronics Temperature	A
E-103		JPL/HVM Spectrum Analyzer/X-Axis Output	A
E-104		JPL/HVM Spectrum Analyzer/Y-Axis Output	A
E-105		JPL/HVM Spectrum Analyzer/Z-Axis Output	A
E-106		JPL/HVM Status	D
E-107		LaRC/MD Event Count	D
E-108	1	UC/CPI Detector D1 Status	B
	2	UC/CPI Detector D2 Status	B
	3	UC/CPI Detector D7 Status	B
	4	UC/CPI Priority Mode Status	B
	5	UC/CPI Calibrate Status	B
	6	UC/CPI Calibrate Status	B
E-109		USC/UV Electronics Temperature	A
E-110		UC/CPI Electronics Temperature	A
E-111		UC/CPI Egg Current Range 1	A
E-112		UC/CPI Egg Current Range 2	A
E-113		UC/CPI Egg Current Range 3	A
E-114		UC/CPI Fission Detector	D
E-115		UC/CPI Fission Detector	D
E-116		UC/CPI Fission Detector	D
E-117		CIT/IR Low Range Temperature	A
E-118		GE/AMD Preamp Temperature	A
E-119		GE/AMD Secondary Voltage	A
E-120			A
E-121			A
E-122		GE/AMD Event Data	D
E-123		GE/AMD Event Data	D
E-124	1	UI/GTT Logic Status	B
	2	GSFC/CRT Status	B
	3	GE/AMD Star Exclusion Status	B
	4	GE/AMD Data Readout Status	B
	5	USC/UV Channel Status	B
	6	USC/UV Roll Status	B
E-125		GSFC/CRT Electronics Temperature	A
E-126		GSFC/CRT Analog Data D1	A
E-127		GSFC/CRT Analog Data D2	A
E-128		GSFC/CRT Detector Temperature	A
E-129		GSFC/CRT Secondary Voltage	A
E-130		GSFC/CRT Identification Data	D
E-131		CIT/IR Command Register--Part 1	D
E-132		CIT/IR Command Register--Part 2	D
E-201		CIT/IR High Range Temperature	A
E-202		JPL/HVM Commutated Housekeeping Data	A

Figure 3.5-14. Science Subcomm Word Assignment (Sheet 1 of 2)

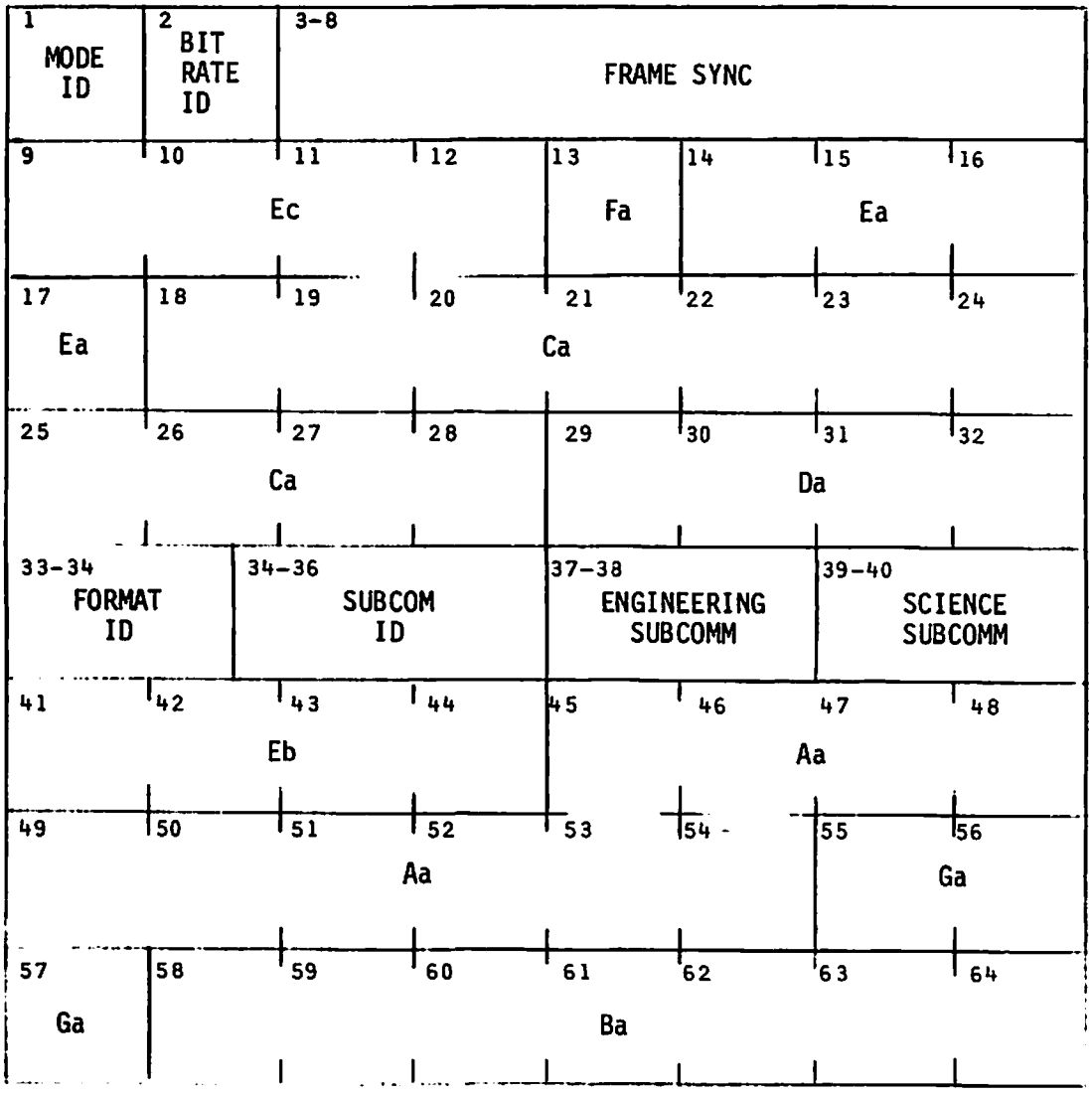
Word	Bit	Measurement	Type
E-203		JPL/HVM Spectrum Analyzer/X-Axis Output	A
E-204		JPL/HVM Spectrum Analyzer/Y-Axis Output	A
E-205		JPL/HVM Spectrum Analyzer/Z-Axis Output	A
E-206		UC/CPI LI PHA	D
E-207		LaRC/MD Event Count	D
E-208	1	GE/AMD Threshold Level Status	B
	2	GE/AMD Bandwidth Status	B
	3		B
	4		B
	5		B
	6		B
E-209		UCSD/TRD Electronics Temperature	A
E-210		UCSD/TRD High Voltage Monitor	A
E-211		UCSD/TRD PCU Monitor	A
E-212		UC/CPI D7 Count Rate	A
E-213		UC/CPI Egg Temperature	A
E-214		UC/CPI L1 L2 Coincidence Count Rate	D
E-215		UC/CPI D1 D2 $\bar{S}$ D3 $\bar{D7}$ Count Rate	
E-216		UC/CPI D2 D4 D5 D6 $\bar{D7}$ Count Rate	D
E-217		UA/IPP High Voltage Monitor	A
E-218			A
E-219			A
E-220		UI/GTT 7.75 vdc Monitor	A
E-221		UI/GTT Electronics Temperature	A
E-222		GE/AMD Event Data	D
E-223		GE/AMD Event Data	D
E-224	1		B
	2	UCTD/TRD Timing Status	B
	3	UCSD/TRD High Voltage Status	B
	4	UC/CPI GSE Stimulus Status	B
	5	USC/UV Channel Status	B
	6	USC/UV Roll Status	B
E-225			A
E-226			A
E-227			A
E-228			A
E-229			A
E-230			D
E-231			D
E-232			D

Figure 3.5-15. Science Subcomm Word Assignment (Sheet 2 of 2)

a) Format A. Format A is the first science format and is arranged to meet the scientific requirements during interplanetary cruising. All 48-3 bit words available are assigned to the scientific instruments. The DTU will accept only digital information in Format A. For the Pioneer F mission 7 scientific instruments share the format. Word assignments for Format A are shown in Figure 3.5-16. Individual instrument word assignments are identified by means of a letter code. The lower case letters represent separate data lines from the instruments to the DTU as well as corresponding separate word gate lines from the DTU to the instruments. In the case of the GSFC Cosmic Ray Telescope, the instrument data is assigned in three locations of the format and three data lines (tied together in the harness) and three word gate lines are being used. Each instrument will receive from the DTU a word gate signal at the appropriate location in Format A as the DTU mainframe multiplexer cycles through the format. The word gate signal informs the instrument that digital information is to be shifted into the DTU for the duration of the word gate. The duration of the word gate varies from instrument to instrument depending on their data requirements.

b) Format B. Format B is the second science format and is arranged to meet the scientific requirements during Jupiter encounter. The requirements and operation of Format B are identical to Format A. The same 7 scientific instruments share the format, which has a slight rearrangement of the words assigned to the GSFC Cosmic Ray Telescope instrument and the UCSD Trapped Radiation Detector instrument. Word assignments for Format B are shown in Figure 3.5-17. As in Format A, the lower case letters denote separate data lines and in every instance where the lower case letter is the same as in Format A, this does refer to the same data line. For the Pioneer F mission, each of the 7 scientific instruments sharing both Formats A and B are using the same data line for both formats.

d) Format C. Format C consists of the engineering subcommutator accelerated at the mainframe rate, resulting in a 32:1 sampling increase of the measurements. The format arrangement is illustrated in Figure 3.5-18. The subformats C1 through C4 are sampled in sequence at the mainframe rate. Eight 6-bit words per subformat are inhibited by the mainframe fixed words as indicated. Both the engineering (with exceptions) and science subcommutators are telemetered in their normal mainframe word positions whenever Format C or one of the engineering subformats C1 through C4 is commanded to the mainframe rate. Not all engineering words that are accelerated to the mainframe rate can be simultaneously transmitted in



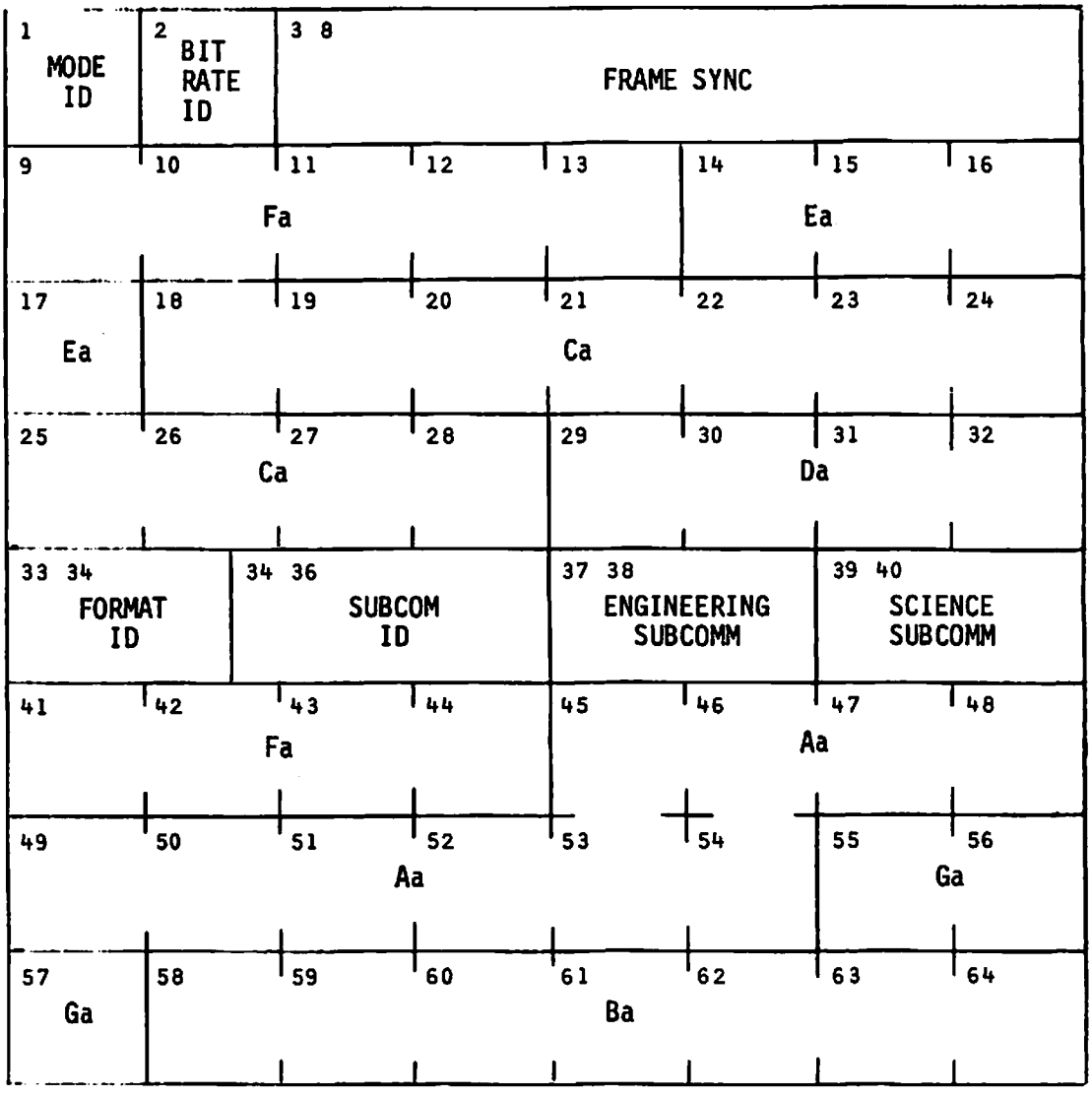
A - JPL Helium Vector Magnetometer
B - ARC Plasma Analyzer
C - U/Chicago Charged Particle Instr.
D - U/Iowa Geiger Tube Telescope
E - GSFC Cosmic Ray Telescope
F - UCSD Trapped Radiation Detector
G - USC Ultraviolet Photometer
H - U/Arizona Imaging Photopolarimeter
I - CIT Infrared
J - GE Asteroid/Meteoroid Det.
K - LaRC Meteoroid Detector

Phantom Word Gates

Da to Instrument G
Ea to Instrument B

Lower case letters denote separate data lines.

Figure 3.5-16. Format A Word Assignments



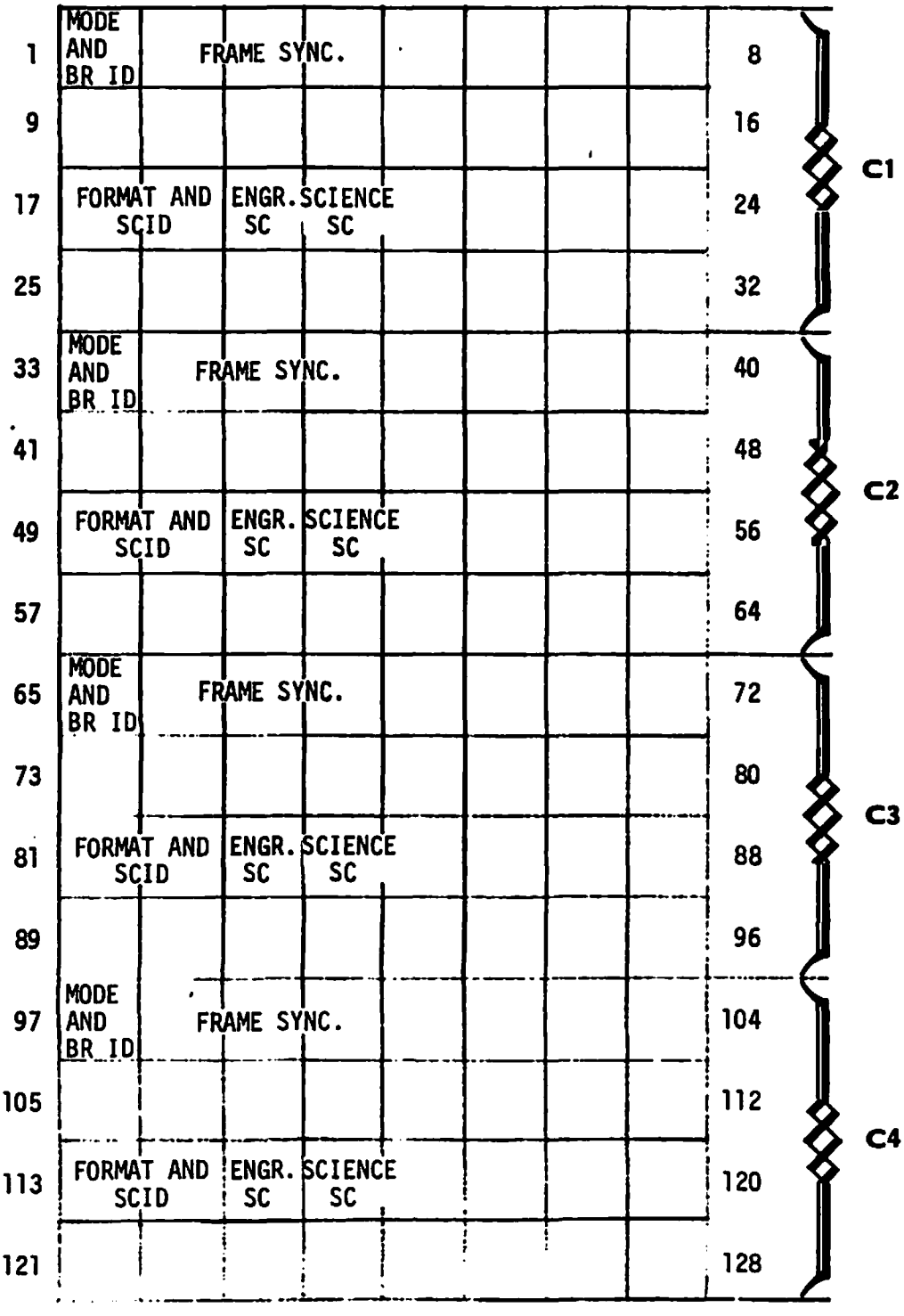
A - JPL Helium Vector Magnetometer
B - ARC Plasma Analyzer
C - U/Chicago Charged Particle Instr.
D - U/Iowa Geiger Tube Telescope
E - GSFC Cosmic Ray Telescope
F - UCSD Trapped Radiation Detector
G - USC Ultraviolet Photometer
H - U/Arizona Imaging Photopolarimeter
I - CIT Infrared Radiometer
J - GE Asteroid/Meteoroid Det.
K - LaRC Meteoroid Detector

Phantom Word Gates

Da to Instrument G
Ea to Instrument B

Lower case letters denote separate data lines.

Figure 3.5-17. Format B Word Assignments



NOTES:
1. The engineering subcommutator is accelerated to one-fourth of the mainframe rate.
2. Blank word slots denote data words accelerated to a nominal rate of BR/768, where BR = bit rate (bits per second). Thirty-two of the 6 bit data words are inhibited by the mainframe fixed words as indicated.

Figure 3.5-18. Accelerated Subcommutator Format C

the engineering subcomm word. Figure 3.5-19 indicates for each engineering subcomm word whether it can be telemetered at a subcomm rate only, both mainframe and subcomm rates simultaneously (Format C commanded), or mainframe rate only (Format C commanded). All zeros or all ones will be telemetered in the engineering subcomm word positions for those words that can be telemetered at the mainframe rate only when Format C is commanded.

d) Formats C1 through C4. Each of the four 32 word subformats designated C1 through C4 may be commanded to the mainframe rate, resulting in a 128:1 sampling increase of the measurements. As indicated in Figure 3.5-20, eight 6-bit words, 1-4 and 17-20, will be inhibited by the mainframe fixed words. The inhibited words will appear in the engineering subcomm word slot 19 only. The entire engineering subcomm will be telemetered in word 19 except for those words that can only be telemetered at the main frame rate (Figure 3.5-19), e.g., words 69, 70, 71, 77, 78, 79, 85, 86, 87, 93, 94, and 95 will not be transmitted in word 19 whenever format C3 is commanded to the mainframe rate. The words of the four formats have generally been grouped to accommodate specific spacecraft subsystems, i.e., C1-Power, C2-Communications, C3-Electrical Distribution/Propulsion, and C4-Attitude Control.

e) Formats D1 through D8. These formats are special formats in which all 192 bits of a mainframe are assigned to a single instrument with the exception of Format D2 in which two instruments share the format. Only one format, D, can be commanded to the mainframe rate at a time, i.e., no combinations of D formats can be commanded. Any of the D formats will be telemetered only on an alternating frame-for-frame basis with either Format A or Format B. Since all 192 bits of a D format contain science data, the mainframe fixed words will be telemetered once every 384 bits instead of once every 192 bits. For the Pioneer F mission, only Formats D1, D2, and D3 will be used. Formats D1 and D2 are referred to as buffer mode formats in that a special buffer storage in the DTU is used by the UA/IPP instrument rather than interfacing directly with the DTU. The buffer mode formats are restricted to the real-time mode of operation.

f) Format D1. Format D1 contains science data from the UA/IPP instrument. Since this instrument generates up to 6000 bits of information in a fraction of a spacecraft revolution, it is necessary to store this data until it can be transmitted at the prevailing spacecraft data rate. This is accomplished by allowing the IPP instrument to store data in the last 6144 bit portion of the DSU at a 16.384 KHz rate. During the time that the IPP instrument is shifting data to the buffer storage of the DSU, the

Engineering Word	Subcomm Rate Only	Main Frame and Subcomm Rates	Main Frame Rate Only
C-101	X	-	-
C-102	X	-	-
C-103	X	-	-
C-104	X	-	-
C-105	-	X	-
C-106	-	X	-
C-107	-	X	-
C-108	-	X	-
C-109	-	X	-
C-110	-	X	-
C-111	-	X	-
C-112	-	-	X
C-113	-	X	-
C-114	-	X	-
C-115	-	X	-
C-116	-	-	X
C-117	X	-	-
C-118	X	-	-
C-119	X	-	-
C-120	X	-	-
C-121	-	X	-
C-122	-	X	-
C-123	-	X	-
C-124	-	X	-
C-125	-	X	-
C-126	-	X	-
C-127	-	X	-
C-128	-	X	-
C-129	-	X	-
C-130	-	X	-
C-131	-	X	-
C-132	-	X	-
C-201	X	-	-
C-202	X	-	-
C-203	X	-	-
C-204	X	-	-
C-205	-	X	-
C-206	-	X	-
C-207	-	X	-
C-208	-	X	-
C-209	-	X	-
C-210	-	X	-
C-211	-	X	-
C-212	-	X	-
C-213	-	X	-
C-214	-	X	-

X denotes words with characteristic shown in column heading.

Figure 3.5-19. Engineering Word Telemetry Rate When Operating in Format C (Page 1 of 3)

Engineering Word	Subcomm Rate Only	Main Frame and Subcomm Rates	Main Frame Rate Only
C-215	-	X	-
C-216	-	X	-
C-217	X	-	-
C-218	X	-	-
C-219	X	-	-
C-220	X	-	-
C-221	-	X	-
C-222	-	X	-
C-223	-	X	-
C-224	-	X	-
C-225	-	X	-
C-226	-	X	-
C-227	-	X	-
C-228	-	X	-
C-229	-	X	-
C-230	-	X	-
C-231	-	X	-
C-232	-	X	-
C-301	X	-	-
C-302	X	-	-
C-303	X	-	-
C-304	X	-	-
C-305	-	-	X
C-306	-	-	X
C-307	-	-	X
C-308	-	X	-
C-309	-	X	-
C-310	-	X	-
C-311	-	X	-
C-312	-	X	-
C-313	-	-	X
C-314	-	-	X
C-315	-	-	X
C-316	-	X	-
C-317	X	-	-
C-318	X	-	-
C-319	X	-	-
C-320	X	-	-
C-321	-	-	X
C-322	-	-	X
C-323	-	-	X
C-324	-	X	-
C-325	-	X	-
C-326	-	X	-
C-327	-	X	-

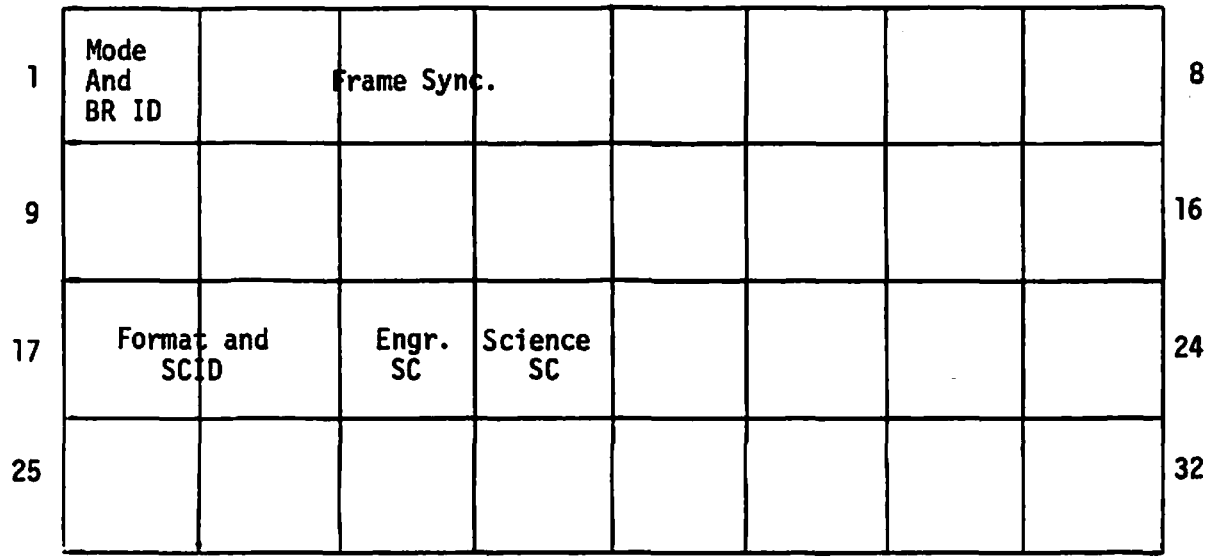
X denotes words with characteristic shown in column heading.

Figure 3.5-19. Engineering Word Telemetry Rate When Operating in Format C (Page 2 of 3)

Engineering Word	Subcomm Rate Only	Main Frame and Subcomm Rates	Main Frame Rate Only
C-328	-	X	-
C-329	-	-	X
C-330	-	-	X
C-331	-	-	X
C-332	-	X	-
C-401	X	-	-
C-402	X	-	-
C-403	X	-	-
C-404	X	-	X
C-405	-	-	X
C-406	-	-	X
C-407	-	-	X
C-408	-	-	X
C-409	-	X	-
C-410	-	-	X
C-411	-	-	X
C-412	-	-	X
C-413	-	-	X
C-414	-	-	X
C-415	-	-	X
C-416	-	-	X
C-417	X	-	-
C-418	X	-	-
C-419	X	-	-
C-420	X	-	-
C-421	-	-	X
C-422	-	-	X
C-423	-	-	X
C-424	-	-	X
C-425	-	-	X
C-426	-	-	X
C-427	-	-	X
C-428	-	-	X
C-429	-	-	X
C-430	-	-	X
C-431	-	-	X
C-432	-	-	X

X denotes words with characteristic shown in column heading.

Figure 3.5-19. Engineering Word Telemetry Rate When Operating in Format C (Page 3 of 3)



NOTES:

1. The following portions of the engineering subcommutator are accelerated to the main frame rate.

C1	=	# 1	32
C2	=	#33	64
C3	=	#65	96
C4	=	#97	128

2. Blank word slots denote data words accelerated to a nominal rate of BR/192, where BR = bit rate (bits per second). Eight of the 6 bit data words are inhibited by the main frame fixed words as indicated.

Figure 3.5-20. Accelerated Subcommutator Format C1, C2, C3, or C4

DTU will telemeter zeros in the D1 format. The telemetering of Formats A or B is not affected by either read-in or read-out of the buffer storage. As soon as the IPP instrument is finished shifting data to the DSU, the DTU will commence read-out of this data in the D1 Format as it alternates with Formats A or B at the operating data bit rate. The DSU furnishes the IPP instrument with an End of Memory pulse whenever the last bit of the buffer storage is either read-in or read-out. This end of memory pulse has no effect on the DTU operating modes as does the end of memory pulse associated with the telemetry store and memory readout modes described in Sections 3.5.2.10(B) and 3.5.2.10(C). Since the DSU memory is of the nondestructive readout type, new IPP data will write over old data and old data can be read-out and telemetered when the IPP instrument stores much less than 6000 bits and the DTU data bit rate is high.

g) Format D2. Format D2 contains science data from both the CIT/IR and UA/IPP instruments. The first 24 bits of each D2 format are assigned to the CIT/IR instrument. The data readout from the CIT/IR instrument is direct without any intermediate storage; i.e., the same as Format A or B. The operation of the UA/IPP instrument with respect to the DSU and readout of its data in format D2 is identical to Format D1.

h) Format D3. Format D3 contains science data from the ARC/PA instrument. The readout of data from the ARC/PA is the same as for Format A or B.

i) Formats E1 and E2. Format E1 is the first 32 words of the science subcomm and Format E2 is the second or last 32 words of the science subcomm. As such, these formats are telemetered only in the science subcomm words of the mainframe, at all times and sequentially. Word assignments are listed in Figure 3.5-15.

3.5.2.20 Power Turn-On. Power to DTU A and B is controlled by a latching relay within the CTRF. Upon spacecraft power turn-on, the last commanded DTU will be powered on as determined by the magnetic latching relay in the CTRF. Upon spacecraft power turn-on or upon commanding DTU redundance A and B, the DHS configuration will be as follows:

1) Last commanded DTU A or B

2) Format C

3) 512 bits/sec bit rate

4) Convolutional coded data

5) Real-time mode

6) SPSG in ACS mode

7) SPSG in zero degree roll reference

The following example indicates the commands required to place the DTU in the following operational configuration:

1) DTU A on

2) Format A

3) 2048 BPS bit rate

4) Convolutional coder off

5) Telemetry store mode

6) SPSG in averaging mode

7) SPSG in 180° roll reference

Transmit the following seven commands to place the DTU into the above configuration.

Command Name	Octal Code	Designator
Step 1 - DTU A On		
DHS will be in the following power turn-on configuration a) Format C b) 512 BPS c) Coded data d) Real-time mode e) SPSG--ACS mode f) SPSG--zero deg roll reference	(044)	DTA
Step 2 - Format A	(010)	FMA1
Step 3 - 2048 BPS Bit Rate	(047)	BIT8
Step 4 - Non-Coded Mode	(071)	UCM1
Step 5 - Telemetry Store Mode	(062)	TSM1
Step 6 - SPSG--Averaging Mode	(077)	RIP3
Step 7 - SPSG--180° Roll Reference	(067)	RIP2

The DTU is now in the desired configuration and ready to sample and format data.

3.5.2.21 Calibration Voltages. Three in-flight calibration voltages are provided to measure the analog/digital conversion accuracy. These voltages are telemetered in engineering subcomm words 1, 2, and 3 at the subcomm rate only, irrespective of Format C or C1 being commanded to the mainframe rate. The calibration voltage generator is nonredundant.

Voltage	Word No	PCM Output Codes Min	Max
Low 168 MV ± 10 MV	1	000011	000101
Med 1.512 V ± 10 MV	2	011111	100001
High 2.424 V ± 10 MV	3	110010	110100

3.5.2.22 Spin Period Sector Generator (SPSG).

a) Definition of terms.

1) P_R Pulse -- Roll pulse - A pulse generated by the attitude control subsystem each time the spacecraft spin axis crosses the ecliptic plane.

2) P_f Pulse -- Roll index pulse - A filtered roll pulse generated by the SPSG which occurs once per spacecraft revolution.

3) P_8 Pulse -- Spin sector pulse - A pulse generated by the SPSG which occurs 8 times between each P_f pulse.

4) P_{64} Pulse -- Spin sector pulse - A pulse generated by the SPSG which occurs 64 times between each P_f pulse.

5) P_{512} Pulse -- Spin sector pulse - A pulse generated by the SPSG which occurs 512 times between each P_f pulse.

b) Basic Operation. From the roll reference pulse provided by the attitude control subsystem the spin period sector generator generates for the scientific instruments and spacecraft subsystems a roll-index pulse and spin period sector pulses corresponding to 512, 64, and 8 sectors per spacecraft revolution. The SPSG is fully redundant but requires commanding to the redundant DTU to operate the redundant SPSG. The function of the SPSG is to produce for roll reference a roll-index pulse which has

less jitter than the roll pulse (up to ±1/2° with Canopus as the source and up to ±1-1/4° with the sun as the source) and provide accurate spin period sector pulses. The SPSG generates an 18-bit spin period measurement which is telemetered in engineering subcomm words C-405, C-406, and C-407 with the MSB first. The SPSG also generates a phase error measurement between the roll pulse and the roll-index pulse with up to a maximum of 60 msec of phase error. This information is telemetered in bits 1 through 5 of engineering subcomm word C-408 with the MSB first. Bit 6 of word C-408 contains the sign bit where a one means the roll pulse has occurred before the roll-index pulse and a zero means the roll index pulse has occurred before the roll pulse. The counters and registers of the SPSG are sized to allow normal operation with spin periods as long as 30 seconds.

c) Operating Modes. In establishing the period between roll-index pulses, the SPSG can operate in one of three modes:

1. Non-spin-averaging
2. Spin-averaging
3. ACS

At power turn-on the SPSG will automatically enter the ACS mode. The three modes are mutually exclusive and are selectable by ground command only. The commands that select the modes also synchronize the roll-index pulse following receipt of the command by the SPSG with the roll pulse irrespective of the operating mode in effect at the time of command reception; e.g., if the system is operating in the spin-averaging mode and it is determined that the phase error between the roll pulse and the roll index pulse is too large and that it is desirable to synchronize the two pulses (0° phase error), a spin-averaging mode command can be transmitted to the spacecraft which will synchronize the roll-index pulse with the roll pulse without interrupting the existing SPSG operating mode. The operating mode of the SPSG is identified in bits 2 and 3 of engineering subcomm word C-417. The code is described in Section 3.5.2.17(F). The operational concepts associated with each SPSG mode are as follows:

1. Non-Spin Averaging Mode. In the non-spin-averaging mode, the period between a pair of roll-index pulses, P_F, (at the beginning and end of a spacecraft revolution) is equal to the period between the pair of roll pulses, P_R, occurring during the previous revolution to within 150 μsec. In this mode a phase error can build up between the roll pulse and the roll-index pulse because of changes in the spacecraft spin rate.

If there are no P_R pulses for an extended period of time, the generator will continue to provide P_F pulses based upon the period of the last two P_R pulses; however, upon receipt of the first P_R after the extended period of no P_R pulses, the P_F period is undefined until receipt of the second P_R pulse.

2. Spin-Averaging Mode. The spin-averaging mode operates in a similar fashion as the non-spin-averaging mode except that the SPSG measures the period of a block of 64 spacecraft revolutions instead of each revolution. In this mode, the period between a pair of roll-index pulses is equal (to within 150 μsec) to the average period between pairs of roll pulses measured during the previous block of 64 revolutions. At constant spin rates, the spin-averaging mode is better than the non-spin-averaging mode since the change in phase error from revolution to revolution is smaller and is constant during a block of 64 revolutions. For changing spin rates, the non-spin-averaging mode is better than the spin-averaging mode. The same technique is used in the spin-averaging mode to make the period between any adjacent pair of roll-index pulses or sector pulses the same as that between any other pair within the same block of 64 revolutions to within 150 μsec. If there are no P_R pulses for an extended period of time, the generator will continue to provide P_F pulses based upon this previous block of 64 P_R pulses; however, upon reinstatement of P_R pulses and completion of a block of 64 P_R pulses, the P_F period will be undefined for the next block of 64 P_R pulses.

3. ACS Mode. The ACS mode is provided to accommodate the constraints of the attitude control subsystem during times of high rate of change of spin rate, such as precession or velocity changes, when the acquisition of scientific data is not the primary spacecraft mode of operation. The ACS mode is exactly like the non-spin-averaging mode in operation except that there will be no phase error buildup between the roll pulse and the roll-index pulse since they will be essentially resynchronized with each roll pulse. In this mode, greater or fewer than 512-sector pulses can occur depending upon the phase difference between the roll pulse and the roll-index pulse from revolution to revolution. If there are no P_R pulses for an

extended period of time, the generator will continue to provide P_F pulses based upon the period of the last two P_R pulses; however, upon receipt of the first P_R pulse after the extended period of no P_R pulses, the P_F period is undefined until receipt of the second P_R pulse.

4. Roll-Index Phase Change. Since the spin axis of the spacecraft can point to either side of the sun during a year, the roll pulse can occur either when the reference line ascends through the reference plane or when it descends through the reference plane. Therefore, the SPSG can be commanded to change the phase of the roll-index pulse by 180° at those time when the roll pulse occurs when the reference line descends through the reference plane. Bit 1 of engineering subcomm word C-417 identifies whether the roll-index pulse is in phase with the roll pulse (a zero) or whether the roll-index pulse is 180° out of phase with the roll pulse (a one).

3.5.2.23 Convolutional Code Generator. The DTU has the capability of convolutional coding the output data bit stream (before bi-phase modulation) by means of redundant encoders. The coding or noncoding of the digital output data is controllable by ground command only. However, the DTU will assume the coded mode when electrical power is applied to the DTU. When the data subsystem is operating in the coded mode, the encoder will replace each data bit generated with two parity bits designated P and $\bar{Q}$. The value of each parity bit is based on the values of selected data bits previously generated in a 32-bit shift register. The encoding cycle begins at the end of the last bit of each frame synchronization word at which time the shift register is reset to zero. The encoder shift register is reset for each 192 bit frame except when the DTU is operating in one of the D formats, in which case the shift register is reset every 384 bits since the frame synchronization word only occurs in the interleaved Format A or B. If the convolutional coder is inputted with more than 33 consecutive zeros, its output will be a series of alternating ones and zeros. Conversely, if the input is more than 33 consecutive ones, the output will also be a series of all ones. The convolutional coding ON/OFF condition is telemetered in bit 1 of the engineering subcomm word C-132 where a one indicates that the coding is on. To operate the redundant coder requires that the redundant DTU be commanded on.

It is possible to reconstruct the information without going through the sequential decoding process. Since the taps for the 2 parity bits P and $\bar{Q}$ are identical except for the second stage, the original data sequence, D, can be reconstructed by module-two adding the two parity streams except for a 1-bit delay, i.e., $D_n = P_{n+1} + \bar{Q}_{n+1}$. This process will recover the complete information stream except for the second to the last data bit received just before resetting the registers. The state of this bit cannot be predicted. However, since the bit in question is the second to the

last bit of the frame synchronization word, this bit of frame sync can be ignored when operating in the convolutional coded mode.

3.5.3 DSU Operational Description

The data storage unit consists of 49,152 bit magnetic core assembly and associated electronics. The DSU is non-redundant and is used to store formatted NRZ-L data from the DTU and to buffer high rate data from the University of Arizona Imaging Photopolarimeter instrument for later transmission at bit rates compatible with ground station requirements. The DSU is controlled by the DTU except when buffering UA/IPP data, in which case the instrument controls the store operations and the DTU controls the readout operations. The memory can be partially stored or readout, by the use of ground commands through the DTU, with later storage or readout occurring at the DSU with the unit remaining in the standby mode until commanded to store or readout data.

3.5.3.1 Inputs. The DSU accepts the following input signals from the DTU and UA/IPP:

a) Formatted digital data from the DTU

b) 65.536 KHz DTU clock

c) Operating mode commands from the DTU

d) MRO buffer gate and gated clocks from the DTU

e) UA/IPP digital data

f) Data store gate and bit shift clock from UA/IPP

3.5.3.2 Output Signals to the DTU and the UA/IPP.

a) Stored formatted DTU digital data

b) Buffered unformatted UA/IPP digital data

c) End of memory signals to the DTU and UA/IPP

3.5.3.3 Operating Modes. The DSU has three basic operating modes, namely:

a) Telemetry store

b) Buffer

c) Memory readout

The three modes are mutually exclusive and if none of the mode commands from the DTU are being received or if end of memory has been reached, the DSU will automatically go into the standby mode. Since the DSU mode commands are controlled by the DTU, mode changes will be frame synchronous.

The memory assembly is of the read/restore type and therefore a memory clear or reset condition is not available. During storing operations the new data will write over old data. In the MRO mode the data is not destroyed and may be read out more than once.

a) Telemetry Store Mode. Upon receipt of a telemetry store command from the DTU, the DSU memory address will reset to zero and formatted data from the DTU is stored at the DTU data bit rate until the memory is full. At the completion of the store cycle, the DSU will generate an end of memory pulse which automatically puts the DTU in the real time mode at the same bit rate and format used during telemetry store. Frame synchronization is maintained during this automatic mode change since the memory stores exactly 256 frames (256 frames x 192 bits/frame equals 49,152 bits). The telemetry store mode can be terminated prior to a memory full condition by either a real time mode, memory readout mode, or a memory halt command. The memory halt command stops and returns the DTU to the real time mode. The telemetry store signal to the DSU remains in the on state but the bit shift pulses to the DSU are inhibited. With the telemetry store signal remaining in the on state, the DSU will not reset the memory address to zero but will retain the last address at which storage took place. By retaining the last address, the DSU is then able to continue storing data at a later time beginning at the address where storage had ended. This is accomplished by commanding the DTU with the memory continue command which returns the bit shift pulse train to the DSU thereby allowing data to be shifted into the DSU.

b) Buffer Mode. Receipt of a buffer mode signal from the DTU commands the DSU to buffer data from the UA/IPP instrument. The buffer mode is defined as the operation of the data handling subsystem in Formats D1 or D2 in the real time mode. This mode is achieved by commanding the DTU to either Format A/D1, A/D2, B/D1, or B/D2. In the buffer mode, up to 6144 bits of instrument data may be stored under the control of the instrument. Data is shifted out of the instrument in bursts of 6 or 10 bits at a rate of 16,384 BPS. The UA/IPP has several modes of operation requiring different burst rates and storage times. Section 11 of PC-220.02 defines the various modes of the UA/IPP. When storing the maximum 6144 bits, the instrument data burst rate is such that the buffer will be filled in two seconds or less. These data are then readout and formatted in Formats D1 or D2 under the control of the DTU in the remaining portion of the spacecraft revolution. In this mode the UA/IPP instrument has priority in the control of the DSU; i.e., readout of the buffer by the DTU may be interrupted at any time by the instrument when new data is to be buffered. Only the last 6144 bits of the DSU memory are used for the buffer mode. During the buffer mode the DSU will accept instrument data only when the instrument data store gate goes

to the high level. The leading edge of each data store gate presents the memory address counter to 43,008. The DSU will continue loading data at the burst rate until the data store gate goes to the low level or the end of memory is reached. After storing 6143 bits, the DSU sends an end of memory (EOM) signal to the instrument signaling that the next bit to be stored will be the last data bit. When the instrument data store gate goes to the low level, The DTU will commence unloading the data at the prevailing bit rate. Should the data store gate return to the high level before completion of readout, the data not yet unloaded will be lost. The DSU will output zeros should the DTU complete the unloading of data prior to receipt of a new data store gate. It will also output zeros while instrument data is being loaded. The buffer mode can be terminated only by a format command other than one involving Formats D1 or D2.

c) Memory Readout Mode. In the memory readout mode, the stored data is read out of the DSU by the DTU wherein it is convolutional coded and bi-phase modulated prior to transmission. Receipt of a memory readout signal from the DTU will result in the memory resetting to address zero and the sequential unloading of the DTU stored data in the same sequence as stored. Readout of data occurs at the DTU bit rate and continues until either the real time mode, telemetry store mode, or memory halt commands are received by the DTU or the end of memory is reached. When the end of memory is reached, the DSU goes into the standby mode until further commands are received. The read-out of data is nondestructive. The memory readout mode is not associated with the buffer mode which automatically reads out stored instrument data. Completion of readout or termination of readout by command will automatically place the DTU in the real time mode at the bit rate used during memory readout and in the format used prior to memory readout. In a manner similar to the telemetry store mode, the data readout can be interrupted by transmitting a memory halt command to the DTU which places the DTU in the real time mode and maintains the memory readout signal to the DSU (bit shift pulses from DTU are inhibited) such that a subsequent memory continue command will continue data readout from the last address. As with all mode changes, frame synchronization is maintained at all times. The time at which data was stored can be determined with the use of the frame (SCID) and extended frame counter words within the stored format.

3.5.3.4 DTU/DSU Control Command Response. Figure 3.5-21 shows the relationships between DTU modes as they affect DSU operation with respect to various command inputs. The real time mode without shift pulses to the DSU is the logic implementation of the memory halt mode which allows the DSU to continue storage at a later time from the last previously stored memory locations; i.e., the DSU logically remains in the telemetry store

or memory readout mode, but without DTU bit shift pulses the DSU is idling (standby mode). The condition where the original mode is RT*/TS or RT*MR and the EOM puts the DTU in the RT* mode is a highly improbable situation as the EOM would have to occur simultaneously with the memory halt command implementation in the DSU. However, logically this is a legitimate state and is therefore listed.

3.5.4 DHS Operational Considerations and Constraints

3.5.4.1 DTU Operational Considerations.

a) When operating at 16 BPS bit rate, a minimum of 12 seconds is required between mode commands. This is required since the DTU processes mode commands synchronously with the frame rate pulse, which has a period of 12 seconds at the lowest bit rate.

   Should another command be transmitted before the occurrence of the frame rate pulse, an erroneous command may be accepted by DTU. This would result in the uncommanded change of mode, bit rate or format.

b) Change of format can cause loss of frame sync when changing from any format to a "D" format. In the "D" format, frame sync occurs only every 384 bits, not every 192 bits.

c) When operating in format "C," format ID cycles through all four ID codes as the format runs through the four parts of format C.

d) The subframe multiplexer is powered partly by TRF A and partly by TRF B. Should one TRF power output be terminated, certain subframe multiplexer data will still be retrieved.

   The following words are retrieved even with one TRF OUT:

	TRF A OUT
C-201	C-41
C-204	C-401 Bits 4, 5, 6
C-205	C-409 Bits 4, 5, 6
C-207	E-201
C-209	E-204
C-212	E-205
C-215	E-209
C-404	E-212
C-412	E-213
C-413	E-207
	E-215

TRB B OUT

C-110	C-124 Bits 1, 2, 3
C-111	C-132 Bits 1, 2, 3
C-122	C-324 Bits 1, 2, 3
C-126	C-332 Bits 1, 2, 3
C-127	E-118
C-130	E-119
C-318	E-126
C-319	E-127
C-326	E-122
C-327	E-130
C-322	E-132
C-330	E-124 Bits 1, 2, 3

COMMAND INPUT	NEXT MODE WHEN ORIGINAL MODE IS:						
	RT	TS	MRO	RT*/TS	RT*/MRO	BMF(RT)	RT*
RT	NC	RT	RT	RT*	RT*	NC	NC
TS	TS_o	NC	TS_o	TS_c	TS_o	NC	TS_o
MRO	MRO_o	MRO_o	NC	MRO_o	MRO_c	NC	MRO_o
Memory Halt	NC	RT*/TS	RT*/MRO	NC	NC	NC	NC
Memory Continue	NC	NC	NC	TS_c	MRO_c	NC	RT
BMF(Buffer Mode Format D1 or D2)	BMF(RT)	BMF(RT)	BMF(RT)	BMF(RT)	BMF(RT)	NC	BMF(RT)
BMF (Any other Format	NC	NC	NC	NC	NC	RT	NC
EOM	NC	RT	RT	RT*	RT*	NC	NC

Notes:

1. NC = No change in basic operational condition.
2. RT = Real time operation when shift pulses are supplied to the DSU.
   RT* = Real time operation without shift pulses to the DSU.
3. RT*/TS or RT*/MRO designates that the DTU has returned to RT* after a memory halt command with the previous TS or MRO memorized.
4. TS_o and MRO_o: Start store or readout from memory address zero.

   TS_c and MRO_c: Reinstate the mode from the last memory location.

Figure 3.5-21. DTU/DSU Control Command Response

e) When accelerating any "C" format, all digital words will be readout only in the mainframe word positions and not in the corresponding subframe positions. Such subframe digital words will read either all ones or all zeros depending upon what state is sitting at the digital input at the time the DTU samples that line.

The above is true except for words C-402, C-403, C-419, and C-420. These words are read out only in the subcomm positions. Refer to Figure 3.5-19 for a listing of such words.

f) It is possible to transmit 192 or more consecutive ones or zeros in a "D" format. For example, in format D-1, if the UA/IPP instrument is loading data into the DSU, the DSU will output continuous zero data for one frame period.

g) Roll attitude timing and 24 bits of spin period sector generator data are completely redundant and will not be lost with failure of either TRF.

h) The SPSG command response will be delayed 65 or 66 spacecraft revolution periods after receipt of an averaging mode command by the DTU.

i) The phase difference between the roll pulse from ACS and the roll index pulse generated by the SPSG is reset to zero with the processing of each averaging mode command.

j) The SPSG response will be delayed two or three spacecraft revolution periods after the receipt of a non-averaging mode command by the DTU.

k) The phase difference between the roll pulse and the roll index pulse is reset to zero with the processing of each non-averaging mode command.

l) The SPSG response will be delayed two or three spacecraft revolution periods after the receipt of an ACS mode command by the DTU.

m) The phase difference between the roll pulse and the roll index pulse is reset to zero with the processing of each ACS mode command.

n) Continuous roll pulses are required from the attitude control system for proper SPSG operation. More than one roll pulse in a single spacecraft revolution will cause erroneous SPSG operation.

If roll pulses are terminated, the SPSG will continue to generate roll index pulses and sector pulses based upon the period between the last two roll pulses.

SECTION 3.5 NOTATION

A	Analog
ACS	Attitude Control
A/D	Analog to Digital
AGC	Automatic Gain Control
ARC/PA	Ames Research Center/Plasma Analyzer
B	Bi-Level
BMF	Buffer Mode Format
BPS	Bits per second
BR	Bit Rate
BRID	Bit Rate Identification
CDU	Command Distribution Unit (Described in Section 3.9)
CEA	Control Electronics Assembly (Described in Section 3.3)
CIT/IR	California Institute of Technology/Infrared Radiometer
CTRF	Central Transformer-Rectifier-Filter
D	Digital
DDU	Digital Decoder Unit (Described in Section 3.9)
DHS	Data Handling Subsystem
DSL	Duration & Steering Logic (Part of CEA)
DSU	Data Storage Unit
DTU	Digital Telemetry Unit
EGSE	Electronic Ground Support Equipment
EOM	End of Memory
FR	Frame Rate
HVM	Helium Vector Magnetometer
GE/AMD	General Electric/Asteroid Meteoroid Detector
GSFC/CRT	Goddard Spacecraft Flight Center/Cosmic Ray Telescope
ID	Identification
IPP	U/Arizona/Imaging Photopolarimeter
JPL/HVM	Jet Propulsion Laboratory/Helium Vector Magnetometer
LSB	Least Significant Bit
LaRC/MD	Langley Research Center/Meteroid Detector
MRO	Memory Readout
MHz	Mega Hertz
ms	millisecond
MSB	Most Significant Bit
MUX	Multiplexer
NRZ-L	Coding of bilevel-serially transmitted signals where a given level has fixed designation; lower level represents 0, higher level represents 1.
NC	No Change
PCM	Pulse Code Modulation
P_f	Roll Index Pulse filtered by SPSG, in phase with P_r
PPS	Pulse Per Second
PPR	Pulse Per Revolution
P_r	Roll pulse from either the sun sensor or Stellar Reference Assembly
PSE	Program Storage & Execution (in CEA)
PSIA	Pounds per square inch absolute

SECTION 3.5 NOTATION (CONT'D)

RIP	Roll Index Pulse
RPM	Revolutions Per Minute
RT	Real Time
RTG	Radioisotope Thermoelectric Generator (Described in Section 3.8)
RT/MR	Real Time/Memory Readout
SC	Subcarrier
S/C	Spacecraft
SCT	Spin Control Thruster
SCID	Subcommutator Identification
SFR	Subframe Rate
SPC	Sensor & Power Control (Part of CEA)
SPSG	Spin Period Sector Generator
SRA	Stellar Reference Assembly
SSA	Sun Sensor Assembly
TRF	Same as CTRF
TS	Telemetry Store (Same as TSM)
TSM	Telemetry Store Mode
TWT	Travelling Wave Tube
UA/IPP	Univeristy of Arizona/Imaging Photopolarimeter
UC/CPI	University of Chicago/ Charged Particle Instrument
USCD/TRD	University of California at San Diego/Trapped Radiation Detector
UI/GIT	University of Iowa/Geiger Tube Telescope
USC/UV	Univerity of Southern California/Ultra-Violet Photometer
V	Volts
VCO	Voltage Controlled Oscillator
VPT	Velocity-Precession Thruster
ΔV/SCT	Mode in ACS for lateral ΔV execution using SCT's

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3.6 COMMUNICATIONS SUBSYSTEM

3.6.1 Communications Subsystem Description

3.6.1.1 Functional Definition. The spacecraft communications subsystem performs the following functions:

a) Radiates a noncoherent RF signal with no uplink signal present to permit acquisition of the spacecraft by the Deep Space Stations (DSS).

b) Provides a phase coherent retransmission after acquisition of an uplink signal such that two-way doppler measurements can be made at the DSS. The frequency translation ratio is 240/221.

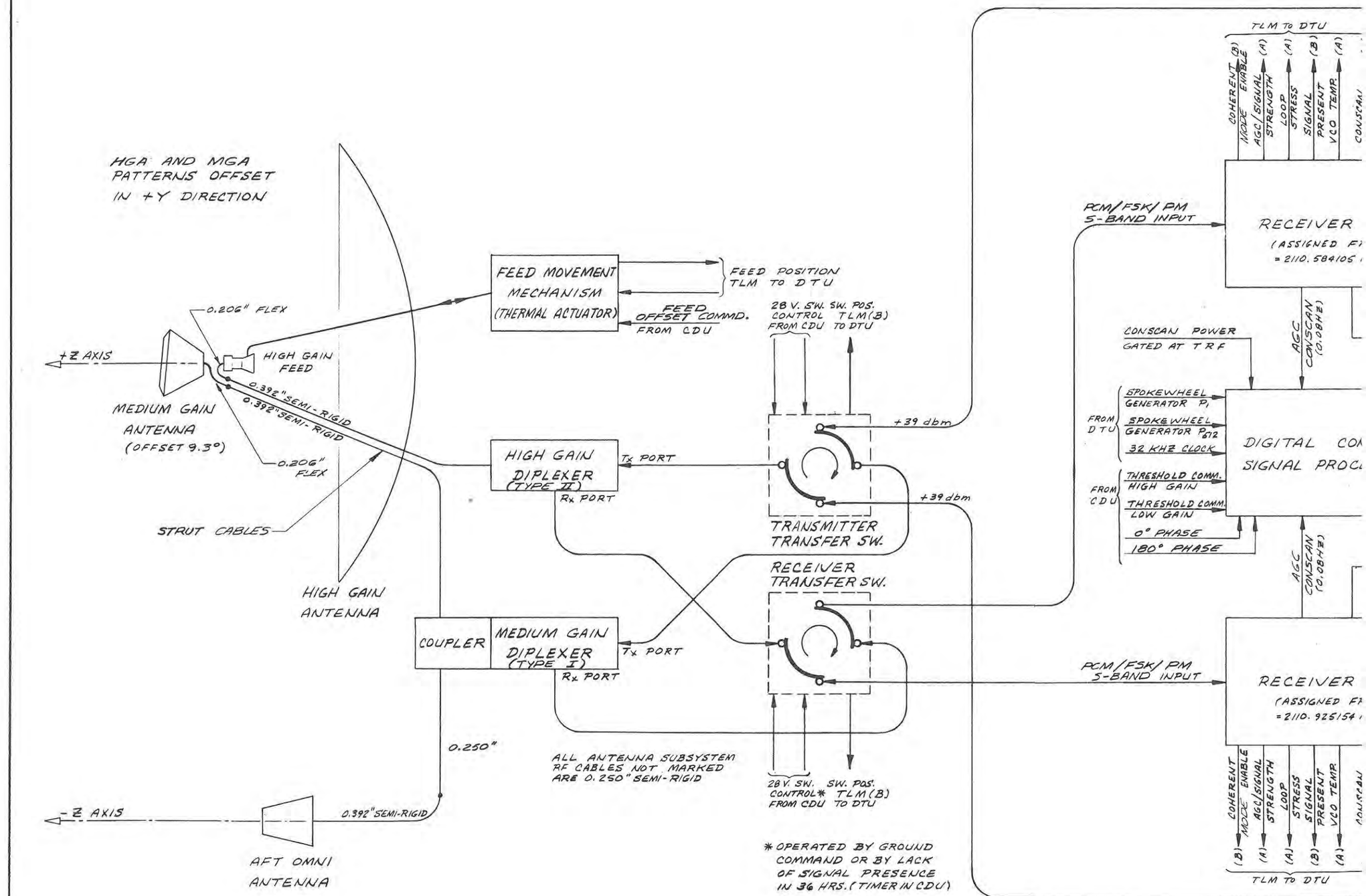
c) Receives from the DSS and demodulates commands modulated in a PCM/FSK/PM format on the uplink signal.

d) Modulates and transmits to the DSS scientific and engineering data in a PCM/PSK/PM format utilizing a single subcarrier at 33 kHz.

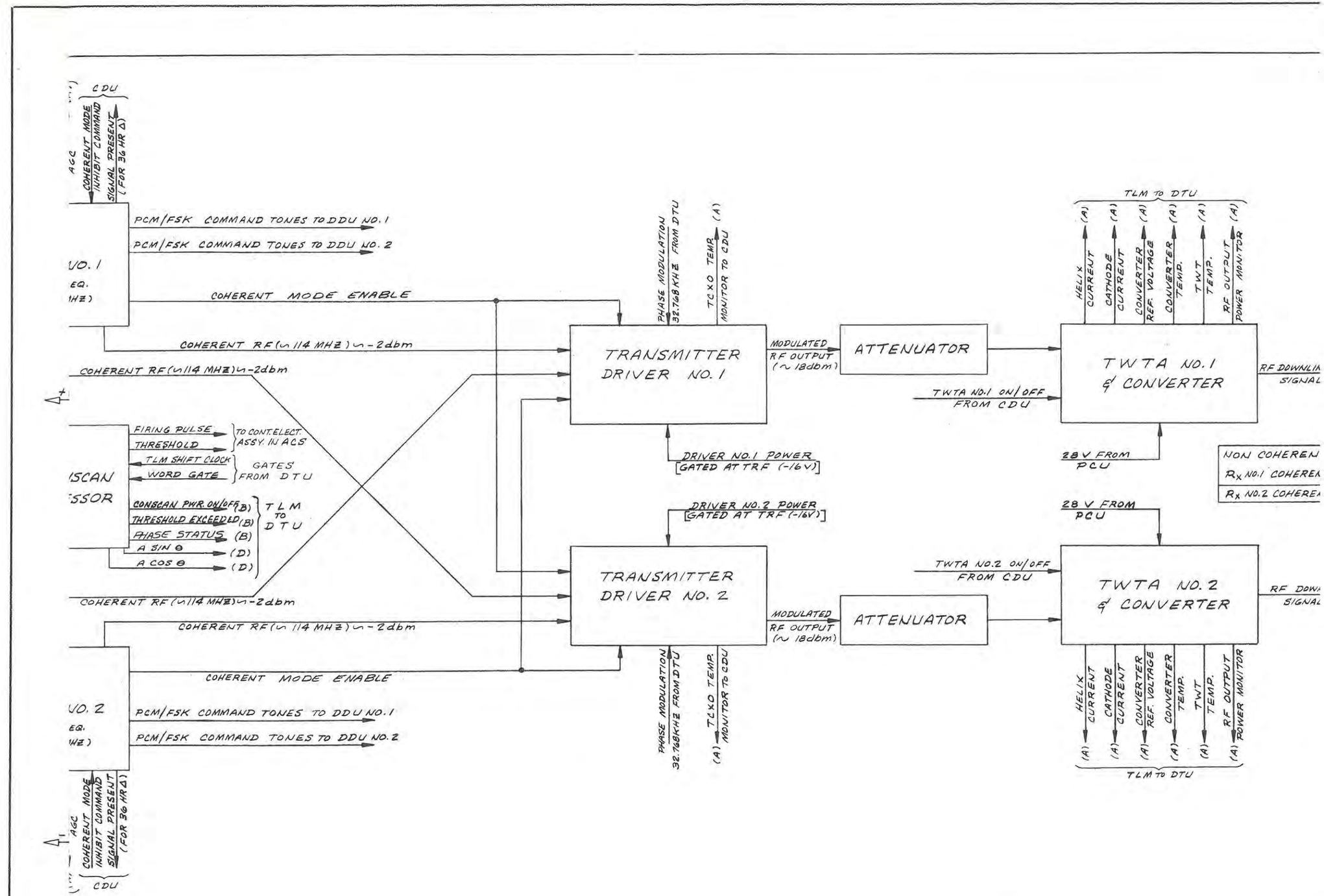
e) Generates, demodulates and processes a conical scan error signal on the RF uplink carrier that is utilized by the Attitude Control Subsystem to precess the spacecraft spin axis toward earth in a closed loop attitude control mode.

3.6.1.2 Block Diagram. A block diagram of the spacecraft Communication Subsystem as shown in Figure 3.6-1 depicts the major functional blocks along with the signal flow, all commanded functions and all telemetered quantities. Major subsystem parameters are summarized in Table 3.6-1.

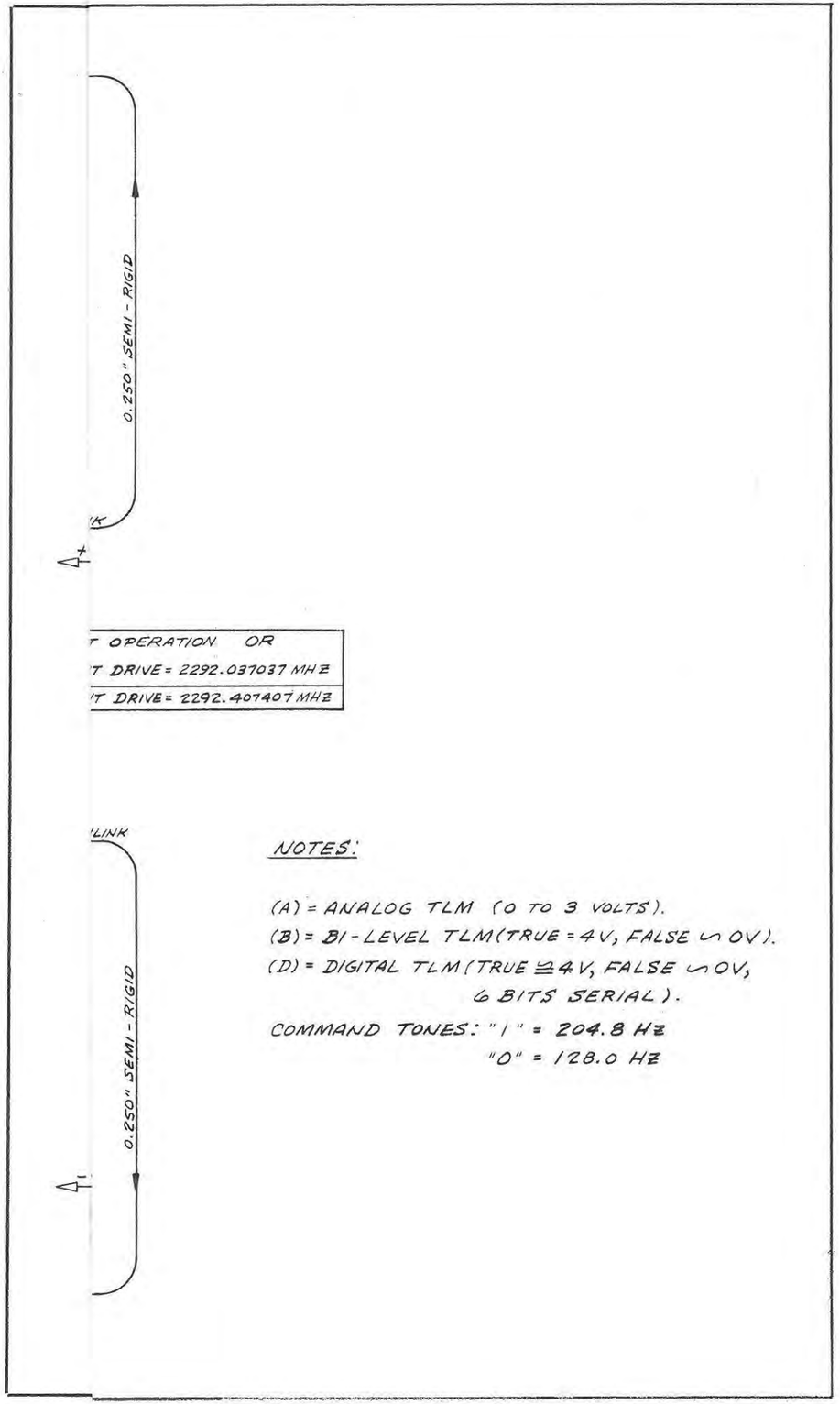
The uplink signal is intercepted either by the high gain antenna or by the medium gain horn with its pattern directed 9.3° off the +Z spacecraft axis or by the aft omni antenna with coverage centered on the -Z axis. The medium gain and omni antennas are tied together through a 3 dB coupler and give an essentially omnidirectional coverage. After separation from the high level downlink signal in the diplexer, the uplink signal is fed to one of the two receivers through the ground commanded transfer switch. The second input port of the transfer switch is connected through a second diplexer to the high gain antenna whose pattern is also directed along the +Z axis in a normal mode and 1° offset in the conscan mode. When one of the two receivers is on the medium gain/omni, the other is on the high gain antenna. A timer in the CDU reverses the antenna/receiver connections in the case that no signal presence has been indicated in 36 hours. This protects against the situation in which the receiver connected to the medium gain/omni antenna fails when the spacecraft is not pointed toward earth. The receivers are assigned different S-Band frequencies and are thus selected by the ground transmitted signal. The selected receiver, using phase lock


HGA AND MGA PATTERNS OFFSET IN +Y DIRECTION
0.206" FLEX
+Z AXIS
HIGH GAIN FEED
0.392" SEMI-RIGID
0.392" SEMI-RIGID
MEDIUM GAIN ANTENNA (OFFSET 9.3°)
0.206" FLEX
STRUT CABLES
HIGH GAIN ANTENNA
FEED MOVEMENT MECHANISM (THERMAL ACTUATOR)
FEED POSITION TLM TO DTU
FEED OFFSET COMMD. FROM CDU
28 V. S'W. SW. POS. CONTROL TLM(B) FROM CDU TO DTU
HIGH GAIN DIPLEXER (TYPE II)
Tx PORT
Rx PORT
+39 dbm
+39 dbm
TRANSMITTER TRANSFER SW.
RECEIVER TRANSFER SW.
COUPLER
MEDIUM GAIN DIPLEXER (TYPE I)
Tx PORT
Rx PORT
0.250"
ALL ANTENNA SUBSYSTEM RF CABLES NOT MARKED ARE 0.250" SEMI-RIGID
28 V. SW. SW. POS. CONTROL* TLM(B) FROM CDU TO DTU
* OPERATED BY GROUND COMMAND OR BY LACK OF SIGNAL PRESENCE IN 36 HRS. (TIMER IN CDU)
-Z AXIS
0.392" SEMI-RIGID
AFT OMNI ANTENNA
TLM TO DTU
COHERENT (B) MODE ENABLE
AGC/SIGNAL STRENGTH (A)
LOOP STRESS (A)
SIGNAL PRESENT (B)
VCO TEMP. (A)
PCM/FSK/PM S-BAND INPUT
RECEIVER
(ASSIGNED F
= 2110.584105
CONSCAN POWER GATED AT TRF
AGC CONSCAN (10.08Hz)
FROM DTU
SPOKEWHEEL GENERATOR P1
SPOKEWHEEL GENERATOR P672
32 KHZ CLOCK
FROM CDU
THRESHOLD COMM. HIGH GAIN
THRESHOLD COMM. LOW GAIN
0° PHASE
180° PHASE
DIGITAL CO
SIGNAL PROC
AGC CONSCAN (10.08Hz)
PCM/FSK/PM S-BAND INPUT
RECEIVER
(ASSIGNED F
= 2110.925154
COHERENT (B) MODE ENABLE
AGC/SIGNAL STRENGTH (A)
LOOP STRESS (A)
SIGNAL PRESENT (B)
VCO TEMP. (A)
TLM TO DTU

FIG



URE 3.6 - 1. COMMUNICATION SUBSYSTEM BLOCK DIAGRAM



NOTES:

(A) = ANALOG TLM (0 TO 3 VOLTS).
(B) = BI-LEVEL TLM (TRUE = 4V, FALSE ≤ 0V).
(D) = DIGITAL TLM (TRUE ≧ 4V, FALSE ≤ 0V,
6 BITS SERIAL).
COMMAND TONES: "1" = 204.8 HZ
"0" = 128.0 HZ

Table 3.6-1. Major Communication Subsystem Parameters

Uplink Frequencies	
Receiver 1 A CH6	2110.584105 MHz
Receiver 2 B CH7	2110.925154 MHz
Downlink Frequencies CH6	2292.037037 MHz
CH7	2292.407407 MHz
	($\frac{240}{221}$ of uplink f)
Effective Radiated Power (ERP) on Spin Axis	
High Gain Antenna	70 dBm
Medium Gain Antenna	49 ~~47~~ dBm
Low Gain	39 ~~35~~ dBm
Bit Rates	16 through 2048 bps (powers of 2)
Uplink Antenna Beamwidths (3 db)	
High Gain	~~3.5°~~ 3.2°
Medium Gain	32°
Low Gain	≈120°
Downlink Antenna Beamwidths (3 db)	
High Gain	3.3°
Medium Gain	29°
Low Gain	≈120°
Antenna Pattern Alignment (Relative to +Z Axis)	
High Gain (Normal Mode)	0°
High Gain (Conscan Mode)	1°
Medium Gain	9.3°
Low Gain	180°
Pointing Accuracy (Conscan)	
High Gain Antenna	0.3° (90%
Medium Gain Antenna	1.3° confidence)

techniques, detects the phase modulated command tones and provides them to each of two redundant digital units (DDU) for detection and decoding.

The downlink transmission of scientific and engineering telemetry data originates with the 32.768 kHz biphase modulated subcarrier which is phase modulated onto the carrier frequency of the redundant transmitter drivers No. 1 and No. 2. The RF source of each transmitter driver is selected by command to be either its self contained non-coherent temperature compensated crystal oscillator (TCXO) or a coherent RF signal from the receiver that is receiving an uplink signal. The receiver provides an RF signal to the driver that is coherently related in phase and in a 12/221 frequency ratio to the S-Band uplink signal. The driver then multiplies the modulated signal frequency by 20 providing drive to the S-Band 8 watt ~~signal~~ mode TWTA. Either TWTA No. 1 or No. 2 with self contained converters, can be selected by ground command. The two TWTAs are tied to two ports of the transmitter transfer switch and the switch output ports are connected to the two diplexers and antennas in the same manner in which the receivers are connected to the receiver transmit switch. The downlink antenna coverage is similar to the uplink.

The Communication Subsystem also implements the conical scanning for pointing of the high gain antenna towards the earth. It also measures pointing error magnitude and phase and telemeters it to earth to establish spacecraft orientation on the ground.

The conscan error signal is introduced as an amplitude modulation on the received RF carrier and is generated by the offset directional antennas as the spacecraft spins. The medium gain antenna is permanently offset by 9.3° from the +Z axis while the high gain antenna pattern is tilted by 1° by the feed movement mechanism upon ground command. The conical scan signal is demodulated by the relatively fast action of the receiver automatic gain control (AGC). The AGC output is an approximately sinusoidal signal with the frequency of 0.08 Hz (equal to the spacecraft spin rate of 4.8 rpm) and the amplitude is approximately proportional to the error angle. Normalization of the error signal is achieved through the AGC action so that the conical scan signal amplitude is independent of the received carrier strength. The concical scan signal processor (CSP) is used to extract phase (zero axis crossing time) and the amplitude near the deadzone. The phase information is supplied to the Attitude Control Subsystem to control timing of precession thruster firing and the amplitude information is used to terminate conscan when the deadzone has been reached. The CSP is digital and requires a roll reference (spin frequency) from the spin period sector generator.

3.6.1.3 Communication Subsystem Capability. The Communication Subsystem capability is summarized in Figures 3.6-2 through 3.6-10. In all cases the presented performance is nominal and the expected tolerance will be about ±2dB. In Figure 3.6-2 the nominal carrier power level received at the DSS receiver input versus communications distance is presented for both the 85 foot and the 210 foot ground station antenna configurations. (The received carrier power level shown in this figure applies to the case when the spin axis is pointed directly at the earth.) The next figure is included for convenience and relates communications distance to the time from launch.

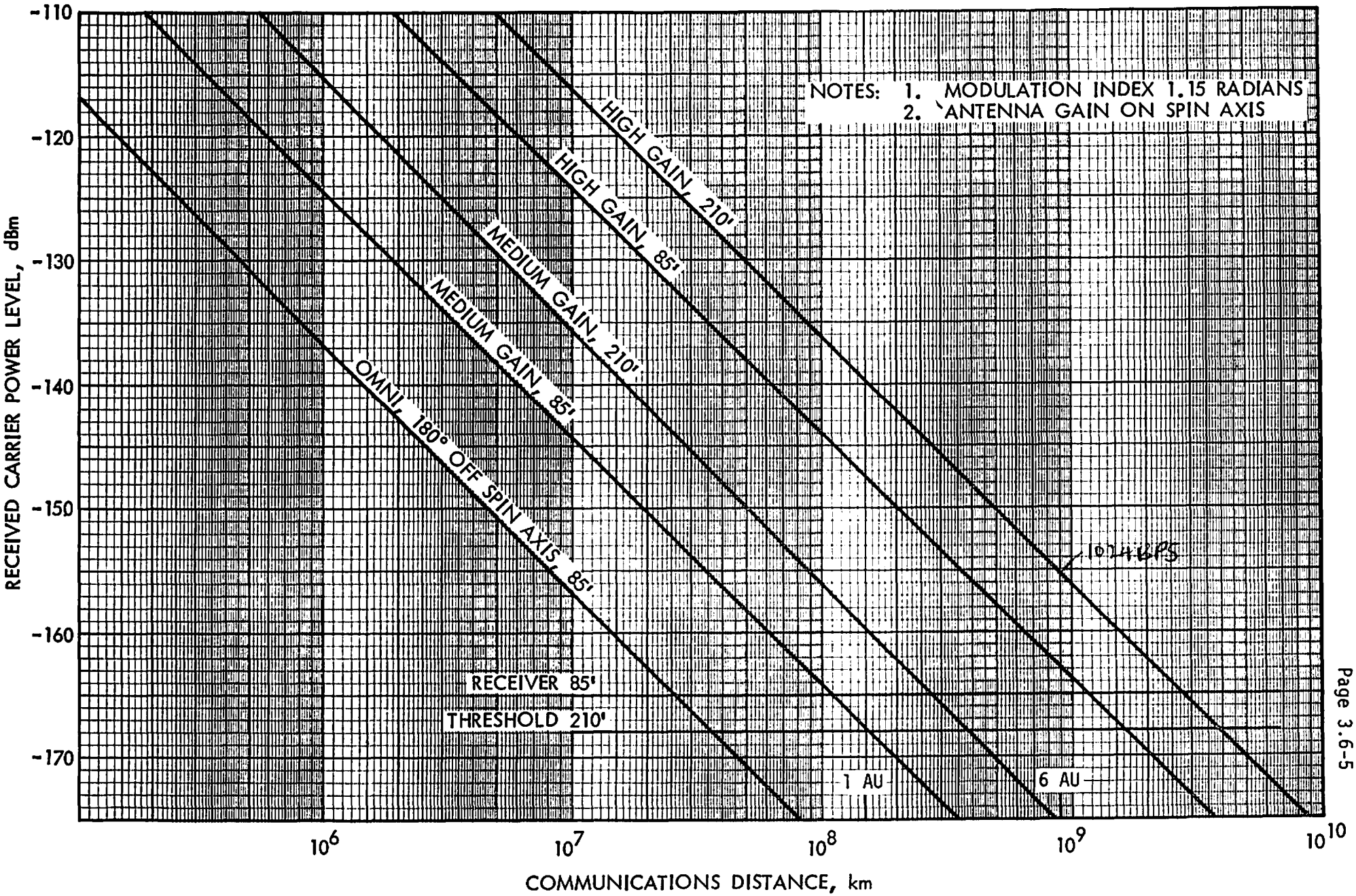


FIGURE 3.6-2. DOWNLINK RECEIVED CARRIER POWER LEVEL VERSUS COMMUNICATIONS DISTANCE

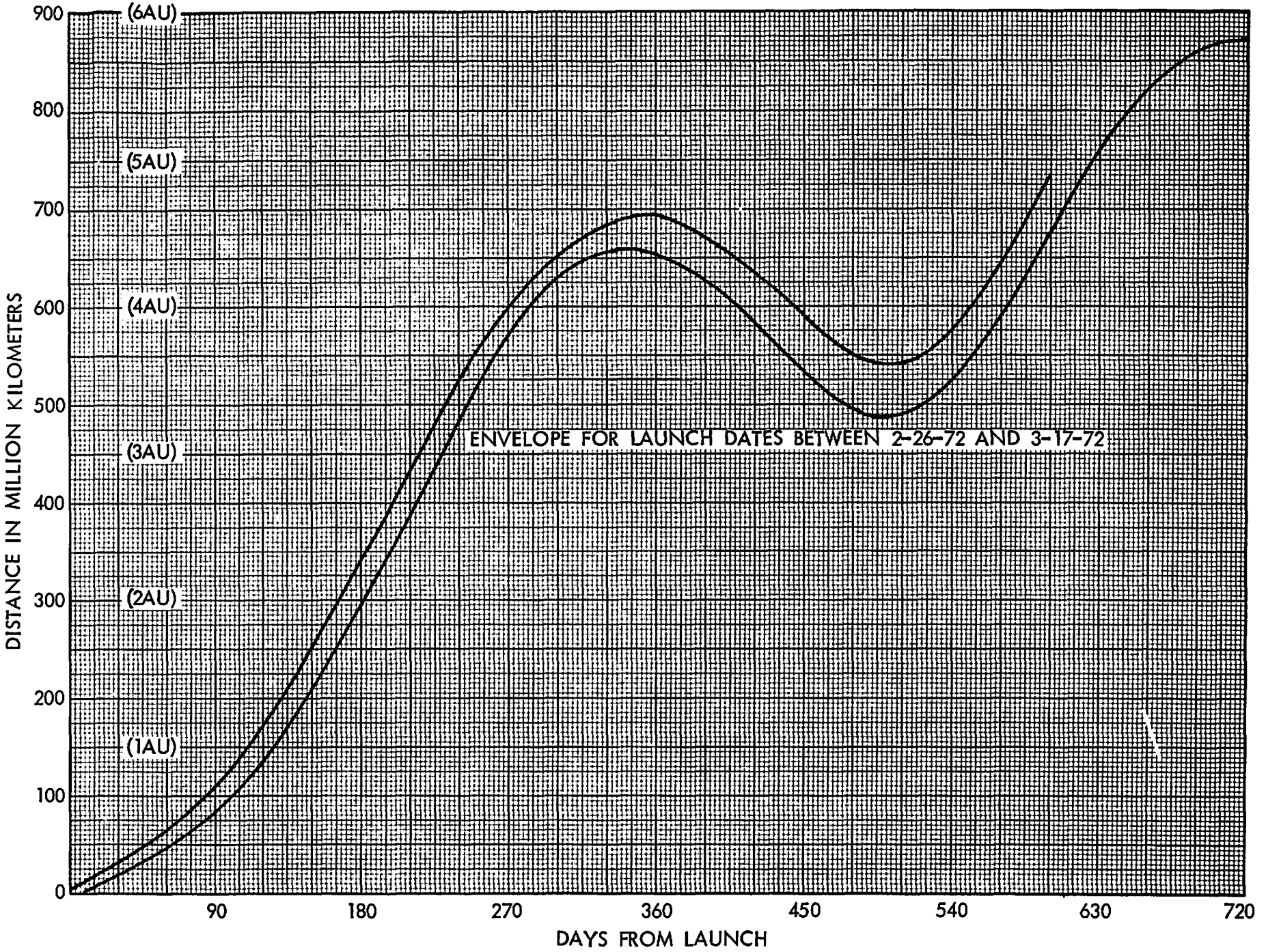


FIGURE 3.6 - 3. COMMUNICATIONS DISTANCE VERSUS DAYS FROM LAUNCH

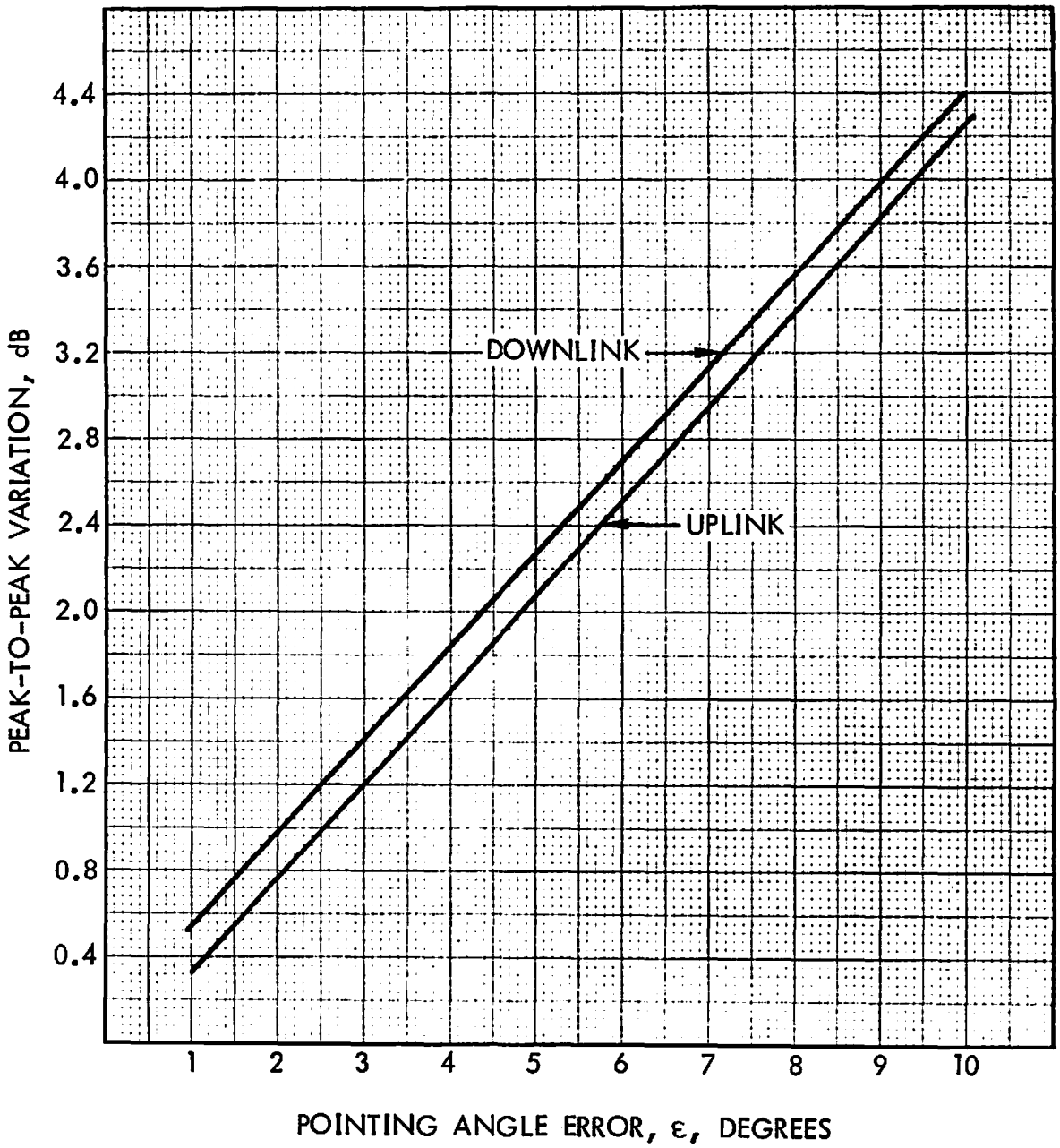


FIGURE 3.6 - 4. VARIATION IN RECEIVED CARRIER POWER LEVEL FOR MEDIUM GAIN ANTENNA

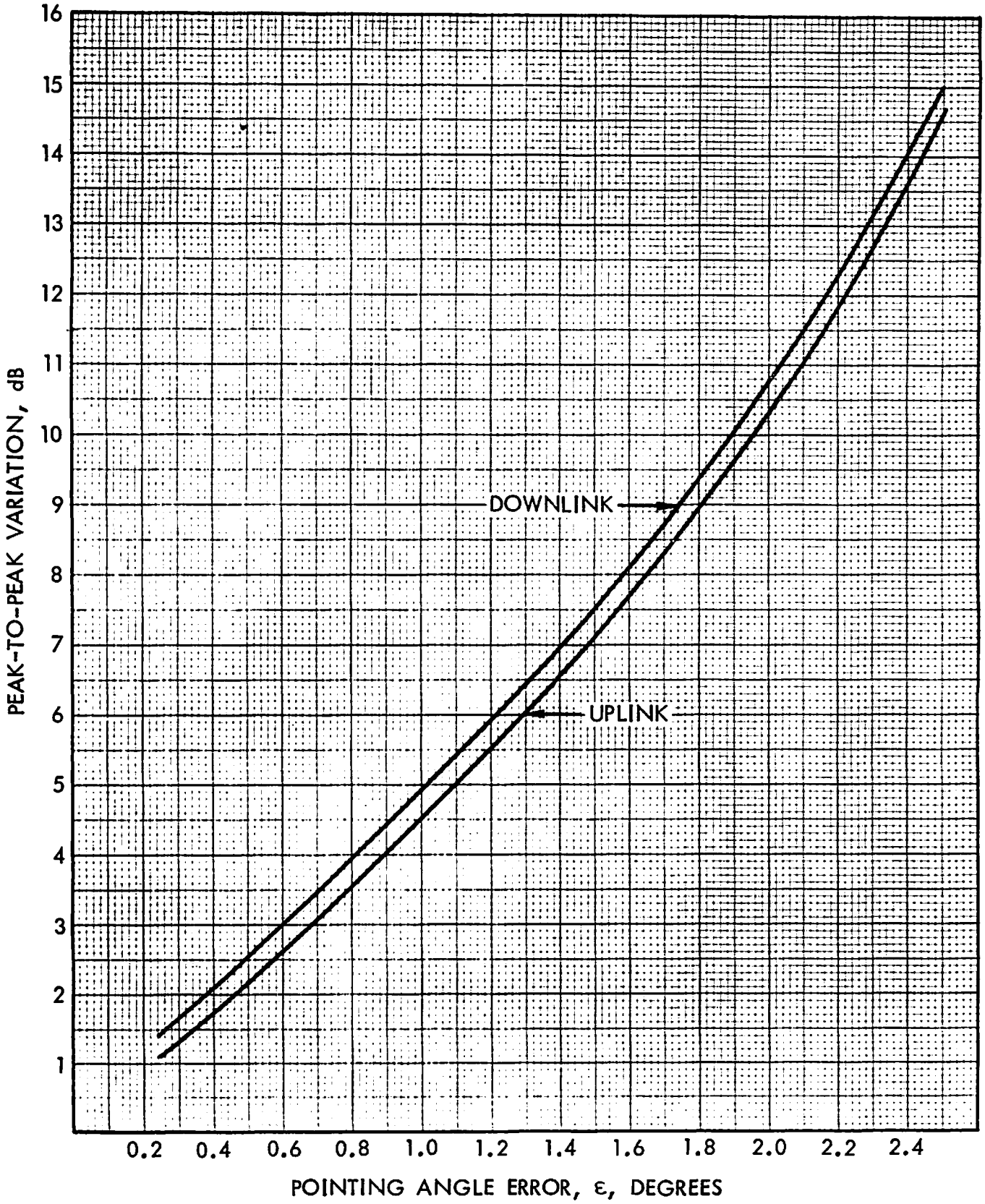


FIGURE 3.6 - 5. VARIATION IN RECEIVED CARRIER POWER LEVEL FOR HIGH GAIN ANTENNA IN CONSCAN MODE

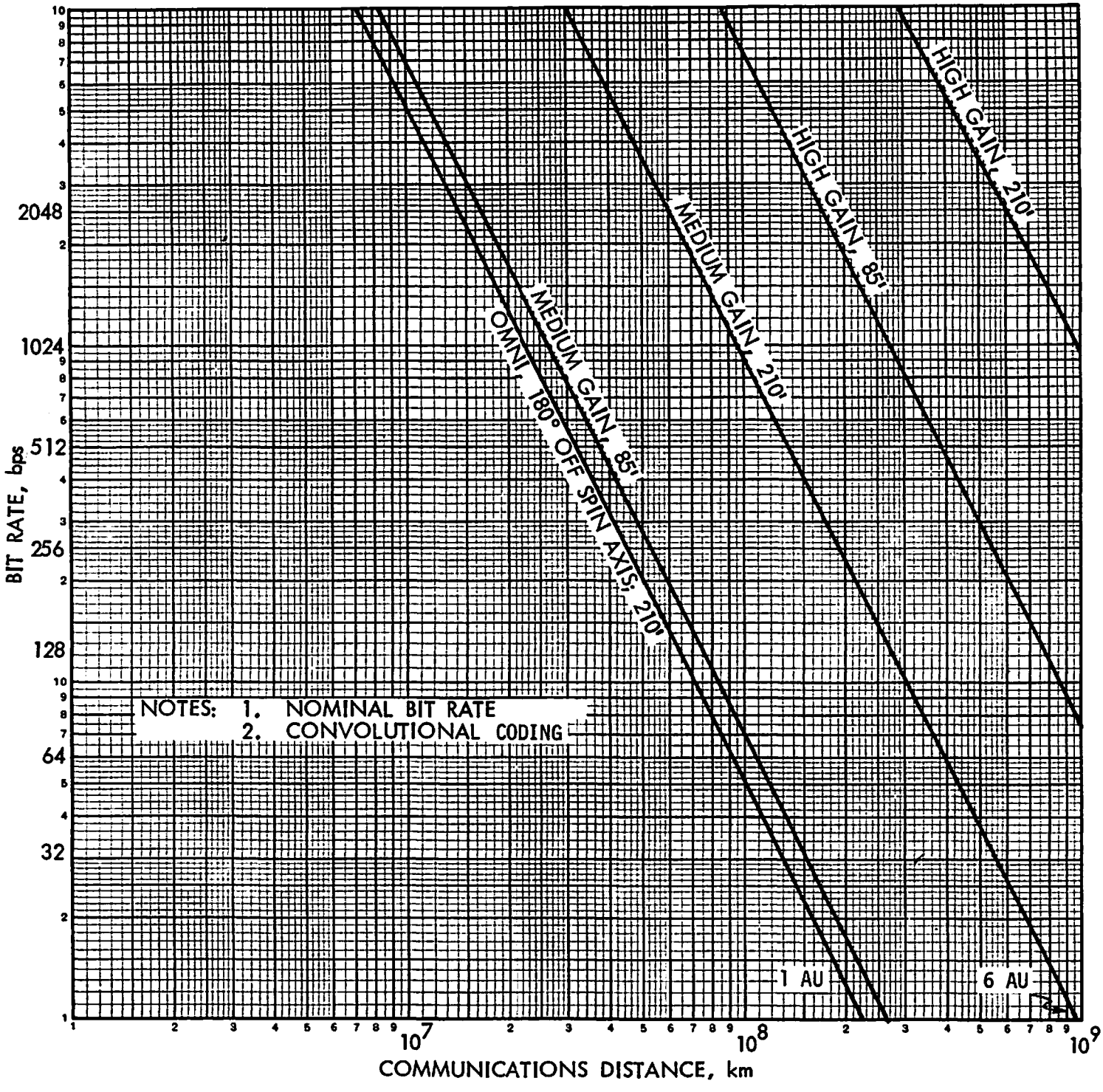


FIGURE 3.6-6. TELEMETRY BIT RATE VERSUS COMMUNICATIONS DISTANCE

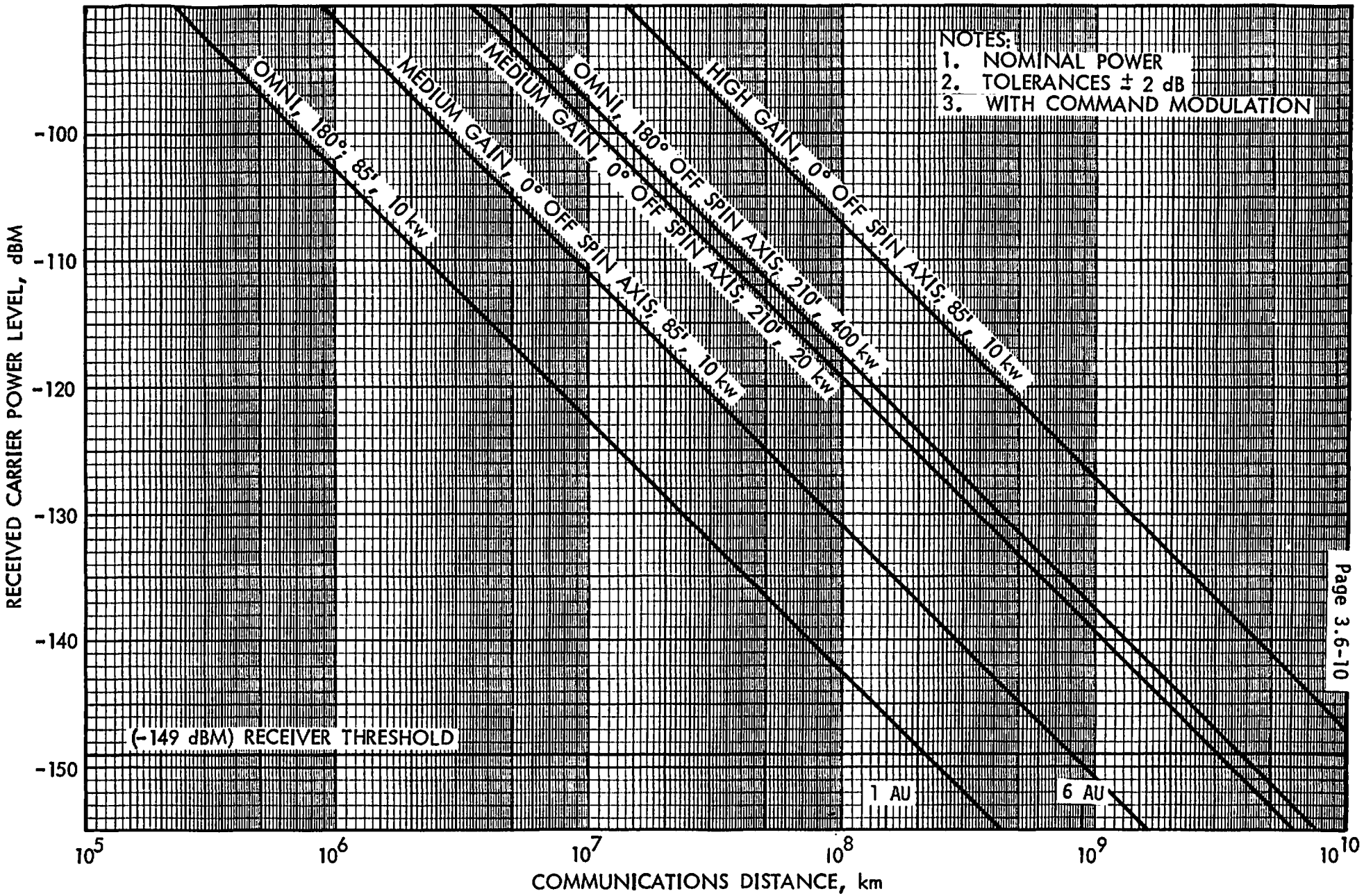


FIGURE 3.6-7. UPLINK RECEIVED CARRIER POWER LEVEL VERSUS COMMUNICATIONS DISTANCE

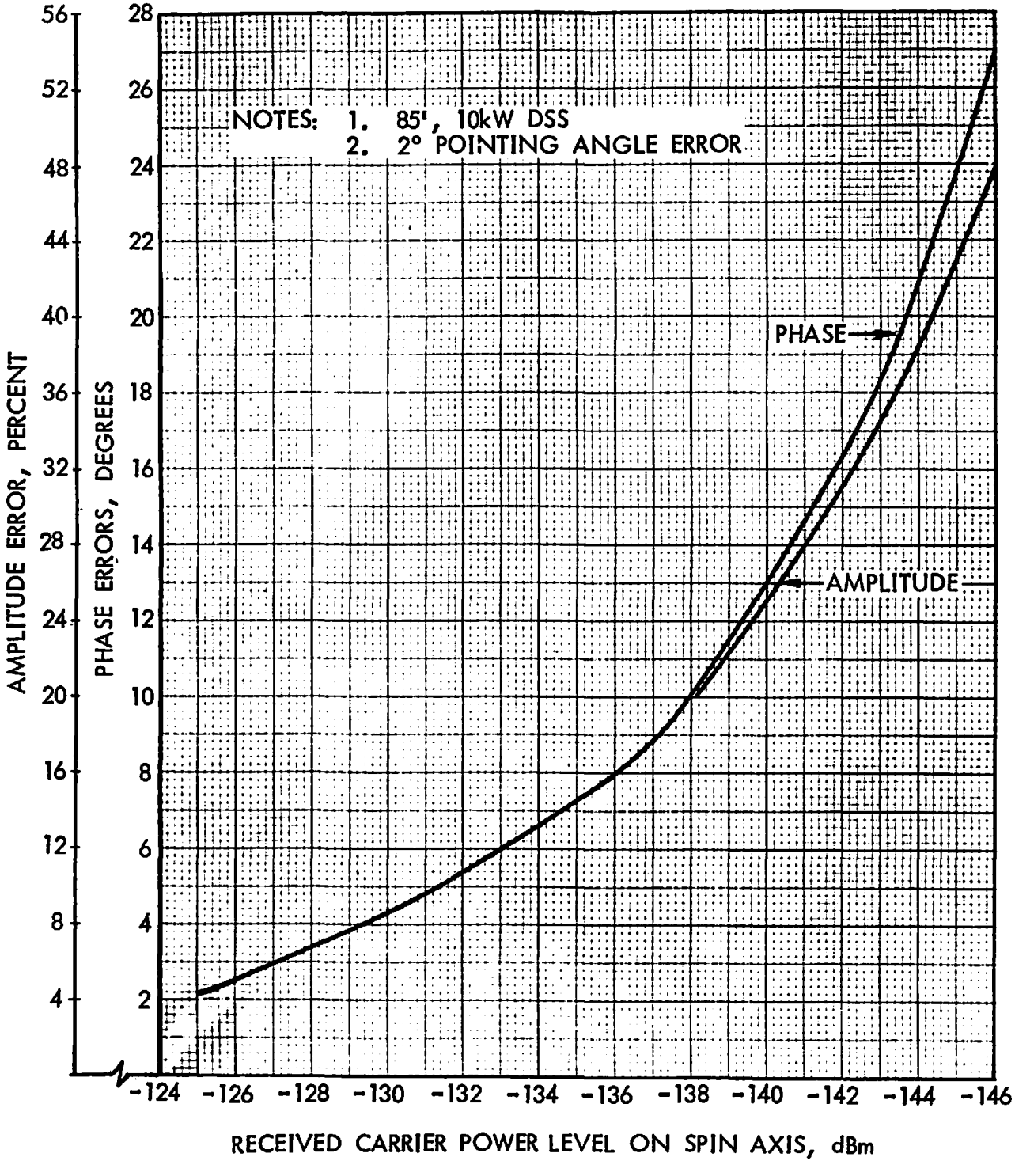


FIGURE 3.6-8. MEDIUM GAIN ANTENNA ONE SIGMA NOISE ERRORS

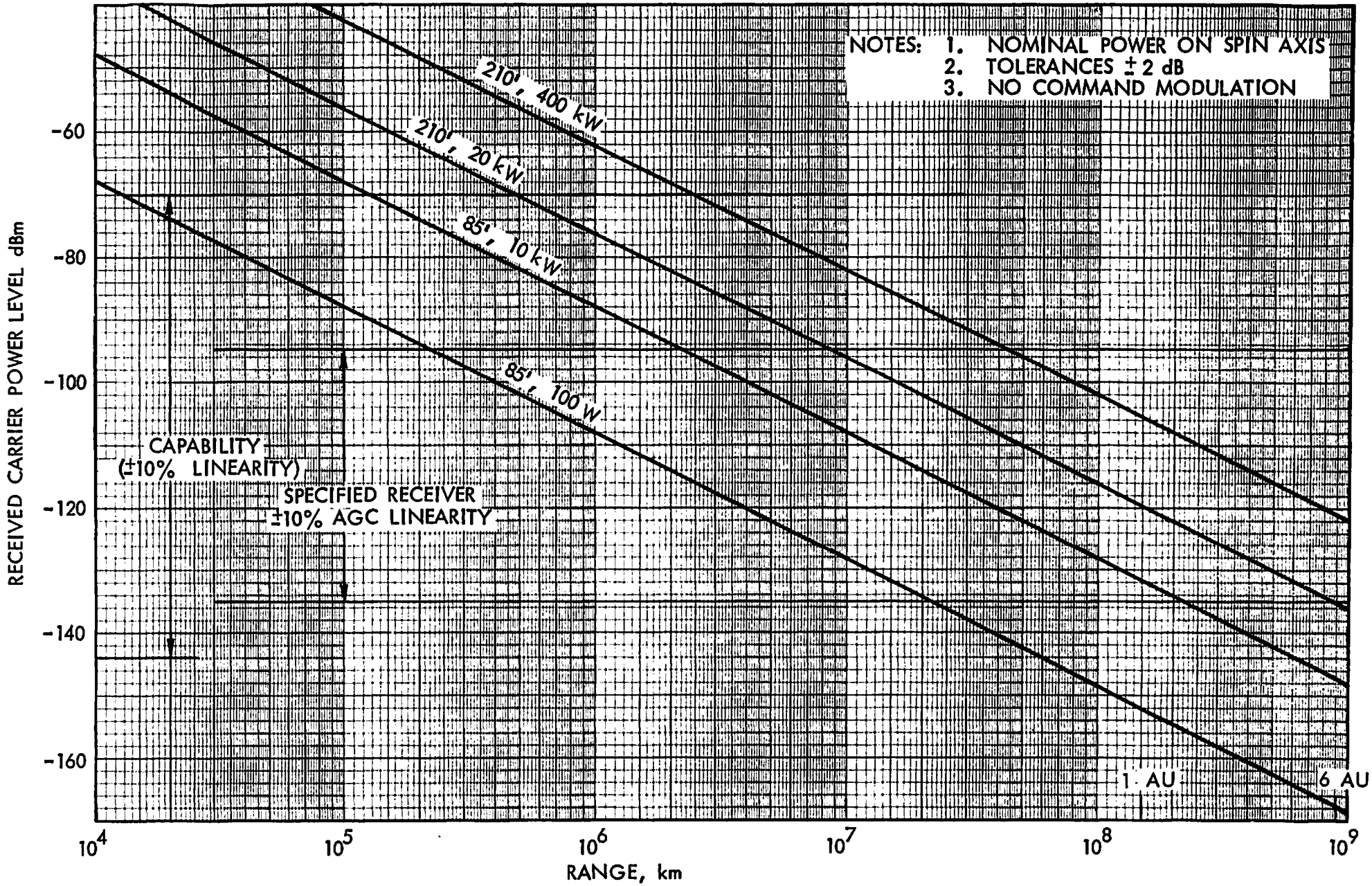


FIGURE 3.6-9. DSS ANTENNA AND TRANSMITTED POWER REQUIREMENTS FOR CONSCAN OVER MEDIUM GAIN ANTENNA

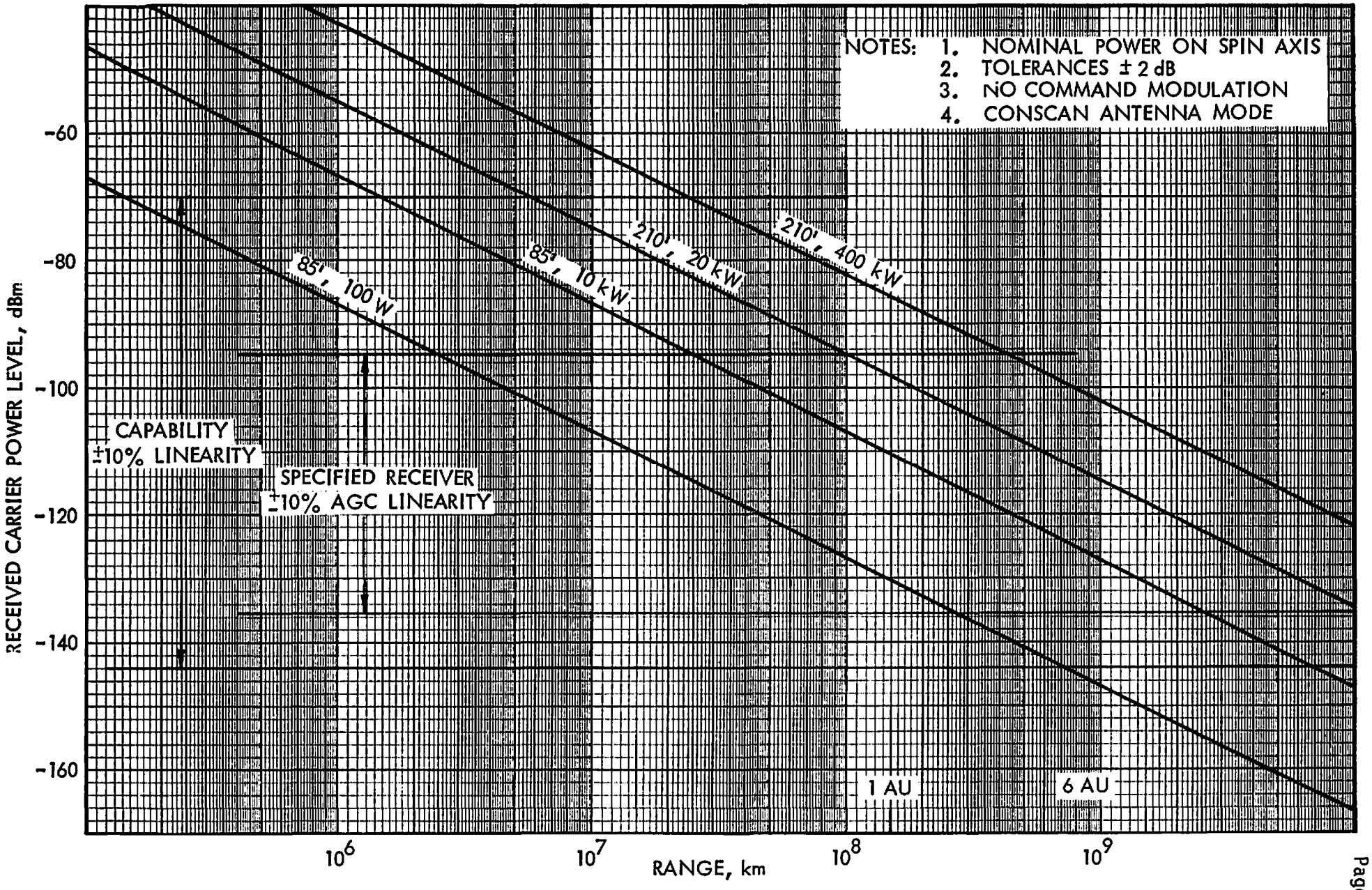


FIGURE 3.6-10. DSS ANTENNA AND TRANSMITTED POWER REQUIREMENTS FOR CONSCAN OVER HIGH GAIN ANTENNA

The normal pointing angle error range for the medium gain antenna is 1 deg to 10 deg and the corresponding variation in the downlink received carrier power level is presented in Figure 3.6-4. For the high gain in the conscan mode, the variation in the carrier power level received at the DSS is plotted in Figure 3.6-5. In addition, switching the high gain antenna to the conscan mode will cause the on axis received carrier power level to drop by about 1.2 dB.

The nominal telemetry bit rates that can be supported with various spacecraft and DSS antenna configurations are presented in Figure 3.6-6. It should be emphasized that the bit rates given in this figure apply to the zero pointing angle error condition and will have to be reduced appropriately for the error angles larger than 0.5 degrees.

The uplink received carrier power level is given in Figure 3.6-7. The spacecraft receiver threshold is defined as a six dB signal-to-noise ratio in the minimum double sided noise bandwidth of 20 Hz. The command channel thresholds at the same time as the carrier channel and, therefore, command performance margin can also be obtained from this figure. The command threshold corresponds to the command error rate of 1.1×10^{-9} or equivalent bit error rate of 4×10^{-3}.

At longer communication distances (beyond 2×10^{8} km with 85', 10 kW DSS) the conscan performance using the medium gain antenna is limited by the conscan phase and amplitude errors of Figure 3.6-8. With large random phase errors, the spin axis instead of precessing in a straight line toward the deadzone exhibits a "random walk." This phenomenon results in a somewhat increased propellant gas consumption but is not serious if the conscan can be reliably terminated in the vicinity of the selected threshold corresponding to 1.3 degrees pointing angle. However, the threshold is affected by the amplitude errors and in general will result in terminating conscan operation at pointing angle errors larger than 1.3 deg. During a nominal mission, it is recommended that conscan using the medium gain antenna not be performed at signal levels below -137 dBm with the -143 dBm level considered as an absolute threshold.

Finally, in Figures 3.6-9 and 3.6-10 DSS transmitter requirements to place the received signal level within the spacecraft receiver AGC $\pm$ 10% linearity range are presented. Under normal operating conditions, the conscan system can operate with AGC non-linearity larger than $\pm$10 percent, however, the conservative $\pm$10 percent requirement insures acceptable conscan system performance under worst case conditions (maximum communications distance for example).

3.6.2 Unit Description

3.6.2.1 High Gain Antenna

3.6.2.1.1 General. The high gain antenna assembly consists of a nine foot diameter reflector, a feed and three support struts as shown in Figure 3.6-11. This assembly provides a right hand circularly polarized pencil beam pattern with a peak gain of about 33 dB at both uplink and downlink frequencies. During the normal mode of operation the feed is

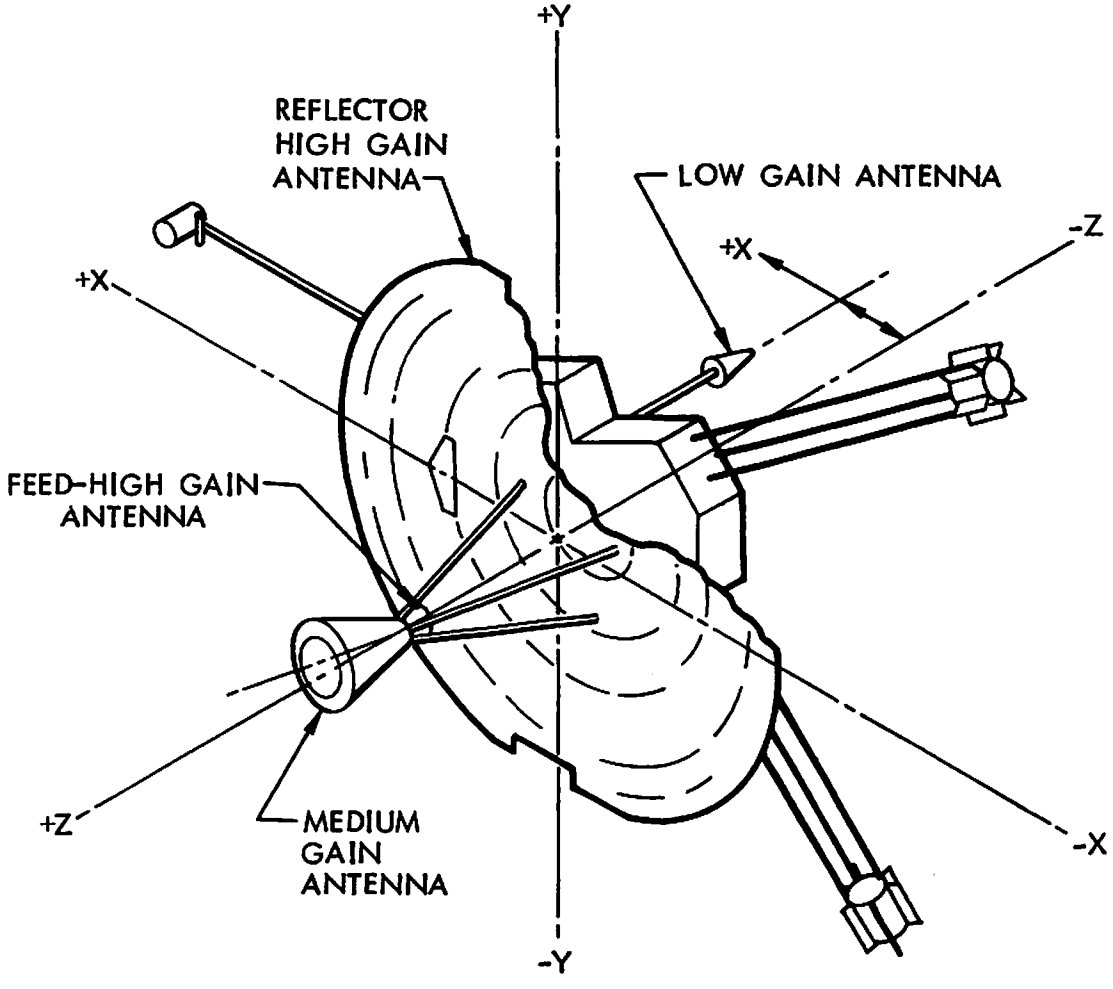


FIGURE 3.6 - 11. SPACECRAFT ANTENNA CONFIGURATION

positioned so that the beam is centered on the +Z axis. For the conscan mode of operation the feed is offset from the +Z axis in the -Y axis direction approximately 1 deg by the feed movement mechanism. This results in an antenna beam offset of about 1 deg from the +Z axis in +Y axis direction (opposite to the feed offset) and a gain of about 1 db below the peak on the +Z axis. While in the conscan mode, the rotation of the spacecraft causes an antenna gain variation or amplitude modulation of the RF carrier received from the DSS. The percent modulation or the conscan signal amplitude is approximately proportional to the angle between the spin axis and the spacecraft-earth line-of-sight or the pointing error angle.

When in the conscan mode the downlink carrier will also be amplitude modulated at a nominal rate of 4.8 rpm or 0.08 Hz. This will result in the peak-to-peak received carrier level variation shown in Figure 3.6-5.

3.6.2.2 Patterns. The uplink principal plane pattern through the ZY plane is shown in Figure 3.6-12. The downlink ZY plane pattern in the normal mode is shown in Figure 3.6-13. These patterns apply to the normal mode of operation and are shifted about 1 deg from the +Z axis in +Y axis direction in the conscan mode. Since the patterns are somewhat different for other principal planes, these differences are summarized in Table 3.6-2.

The conscan signal peak amplitude in dB (the conscan pattern) is presented in Figure 3.6-14. This conscan signal amplitude multiplied by the receiver AGC slope (nominally 0.1 volts/dB) gives the amplitude of the 0.08 Hz signal component at the input to the conscan signal processor. Combining Figure 3.6-14 with the receiver and the conscan signal processor calibration curves, the overall calibration curve for the pointing error angle magnitude versus telemetered parameters A sin θ and A cos θ is obtained.

Table 3.6-2. Major High Gain Antenna Parameters

	Uplink	Downlink
Peak Antenna Gain	(32.5±0.4)dB	(33.3±0.4)dB
Three db Beamwidth	3.4° to 3.5°	3.15° to 3.3°
Maximum 3 db Beamwidth	3.4°	3.6°
Offset Angle in Conscan Mode	--	1°
Crossover Level in Conscan Mode	--	1 dB
Location of First Null	2.2° to 2.4°	1.8° to 2.2°
Depth of the Null		
Minimum	Less than 19 dB	
Maximum	More than 40 dB	

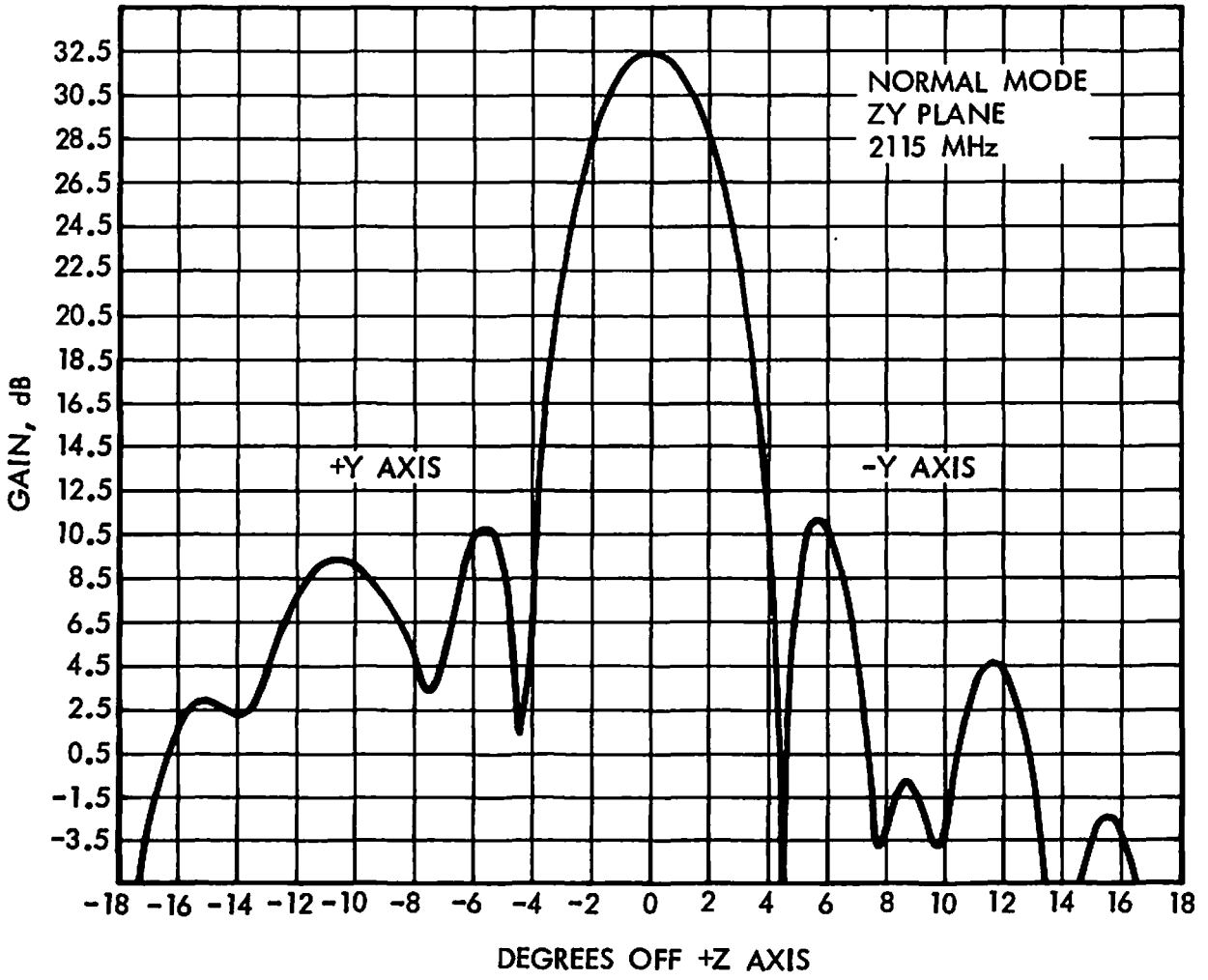


FIGURE 3.6 - 12. UPLINK HIGH GAIN ANTENNA PATTERN

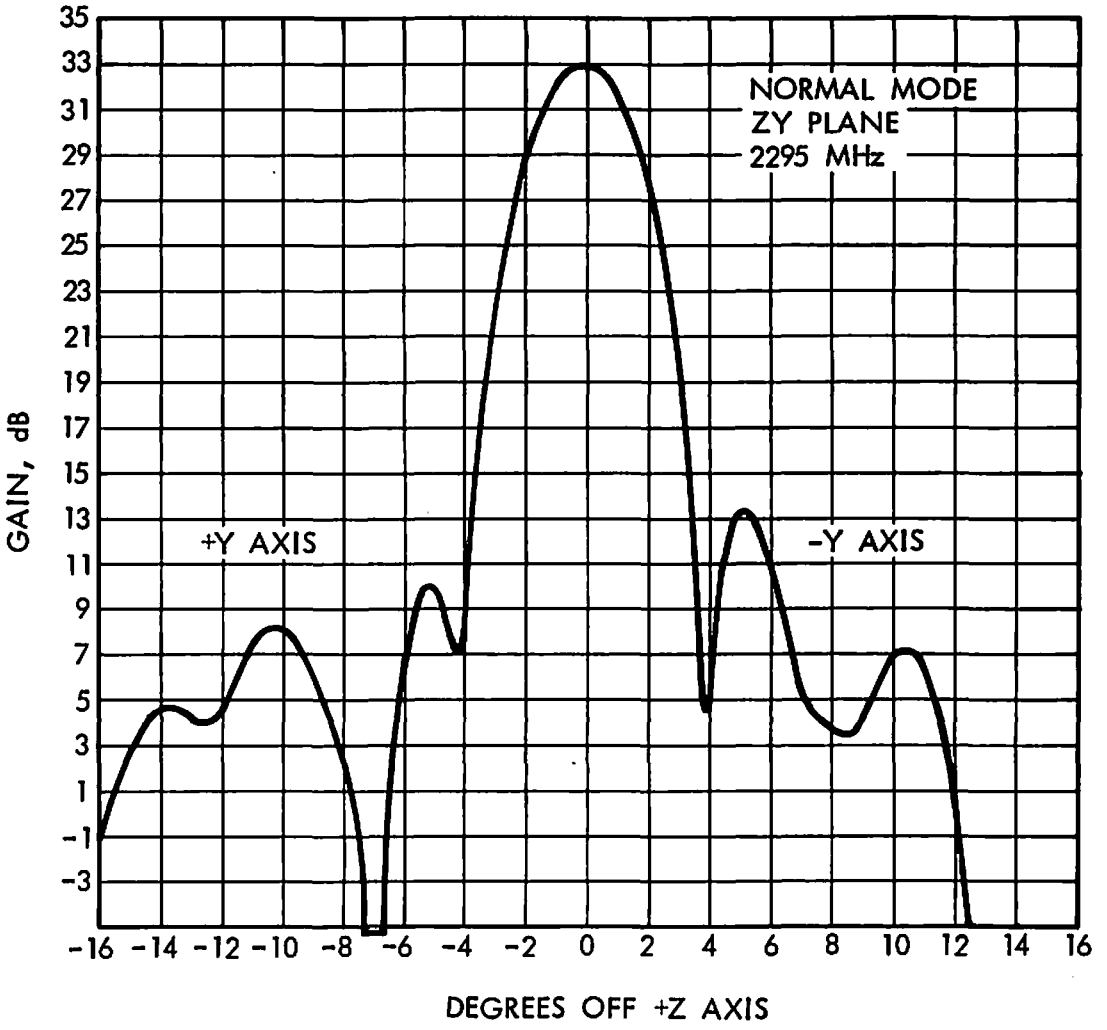


FIGURE 3.6 - 13. DOWNLINK HIGH GAIN ANTENNA PATTERN

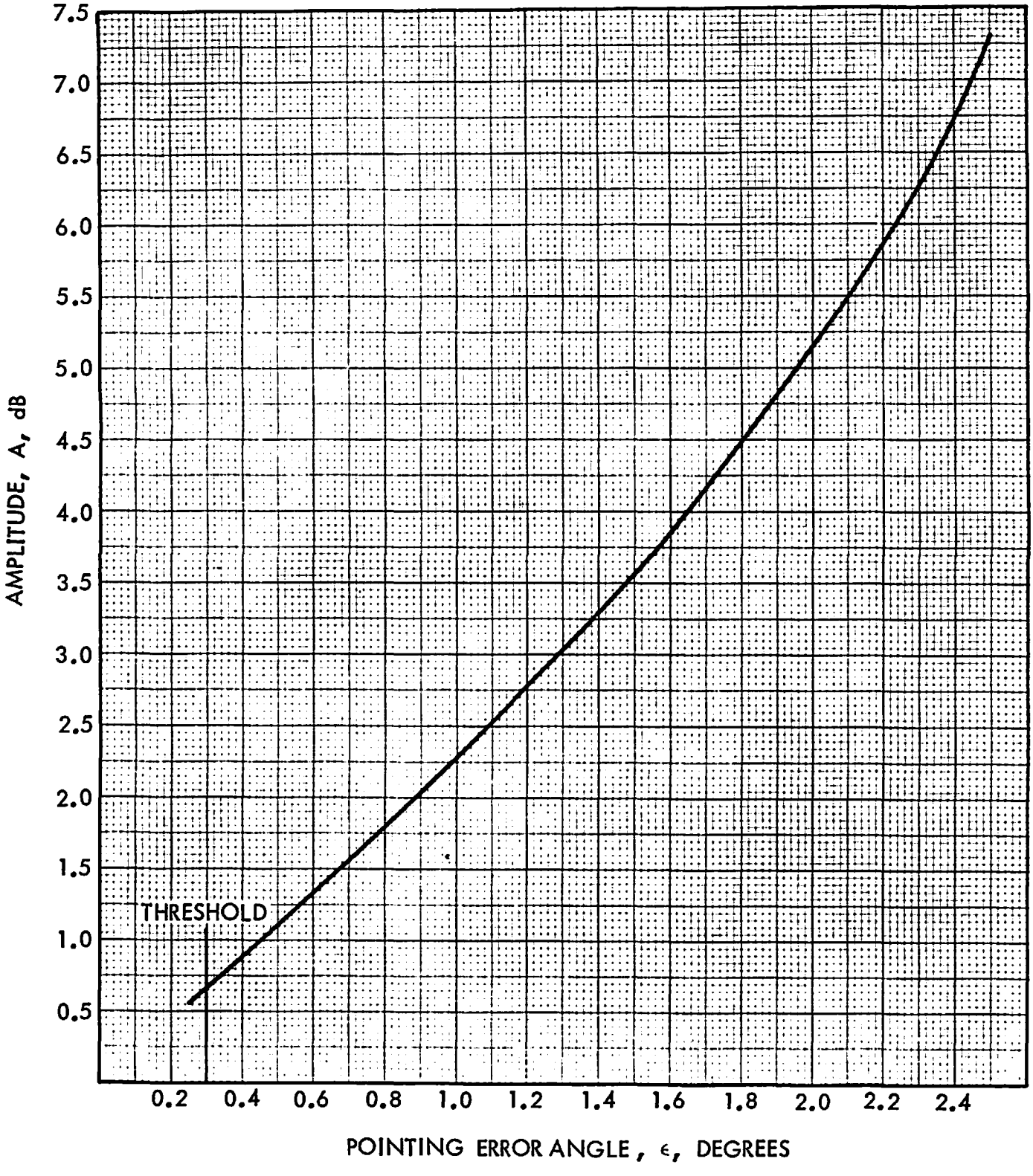


FIGURE 3.6-14. ENGINEERING MODEL HIGH GAIN ANTENNA PEAK CONSCAN SIGNAL AMPLITUDE

The conscan signal phase (negative-to-positive going axis crossing) calibration curve is given in Figure 3.6-15. This conscan signal phase is measured by rotating the offset antennas clockwise (opposite to spacecraft rotation) and recording the gain variation as a function of rotational angle. The antenna beam is offset on the +y-axis and initially the reference range is directed on the +x-axis so that the ideal case of zero phase shift corresponds to the axis crossing occurring at t = 0. Positive phase refers to phase lag or delay in time and negative phase is defined as phase lead or advance in time. Since the small phase deviation from the ideal might be partly produced by the range errors, no phase compensation is used for conscan. For this reason, phase variation cannot be completely compensated on the ground when computing θ from the telemetered parameters, A sin θ and A cos θ.

3.6.2.3 Medium Gain-Omni Antenna.

3.6.2.3.1 General. The medium gain-omni antenna consists of the medium gain element coupled through the 3 dB coupler to the omni antenna. This combination provides a low gain coverage around the spacecraft. The medium gain antenna provides coverage during the early portion of the mission and is utilized for coarse conscanning. The omni element is a truncated log conical spiral covered by a fiberglass radome. The omni antenna is mounted on a short boom at the aft end of the spacecraft as shown in Figure 3.6-11.

The medium gain antenna consists of a corrugated conical horn and a transition mount. The horn and mount are located near the reflector focal point on the structure that also houses the high gain feed. To provide coarse conscanning, the horn is permanently offset approximately 9.3° off the +z axis resulting in a crossover level of about 1dB.

3.6.2.3.2 Patterns. The complete uplink medium gain-omni antenna pattern is shown in Figure 3.6-16 and the downlink pattern in Figure 3.6-17. The patterns have been obtained on the one third scale antenna and spacecraft model and are accurate in all respects except for the exact location and the depth of nulls in the region between 54° and 78° off the spin axis. It is questionable that the depth of the nulls are accurately measured and they might be deeper than indicated in the figures. It is possible that some of the nulls will be deep enough to cause loss of downlink or uplink communications. In addition, under certain circumstances, the phase lock loop receiver performance could be degraded or the probability of losing lock increased. The degradation might be caused by the phase modulation produced by the spinning spacecraft and the 360 degrees carrier phase change between adjacent interferometer nulls.

The conscan signal amplitude and phase calibration curves are shown in Figures 3.6-18 and 3.6-19 respectively. Combining these figures with the receiver and conscan signal processor calibration curves, the overall calibration curves for the pointing error angle magnitude and phase versus telemetered parameters A sin θ and A cos θ are obtained.

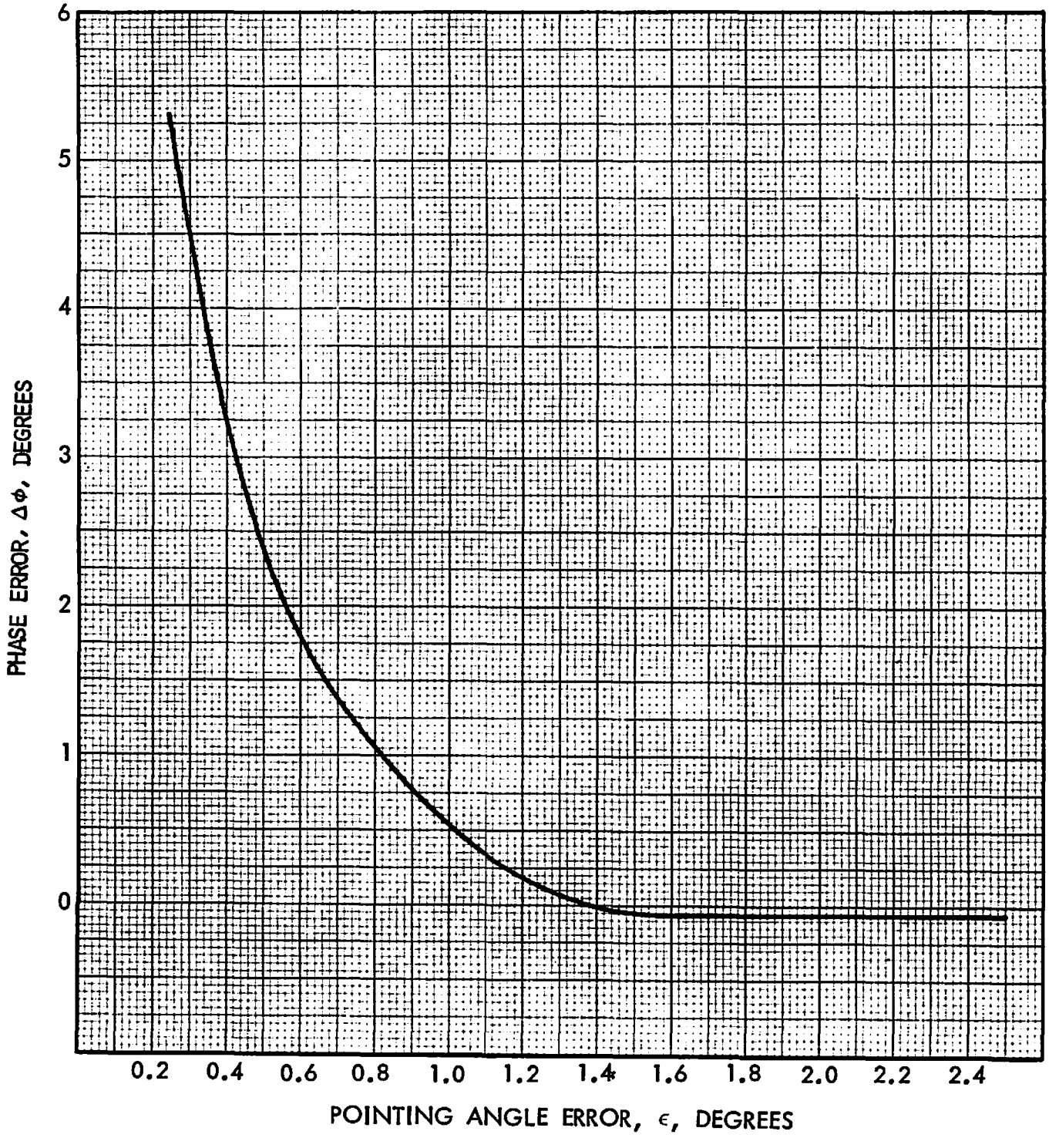


FIGURE 3.6-15. ENGINEERING MODEL HIGH GAIN ANTENNA PEAK CONSCAN SIGNAL PHASE

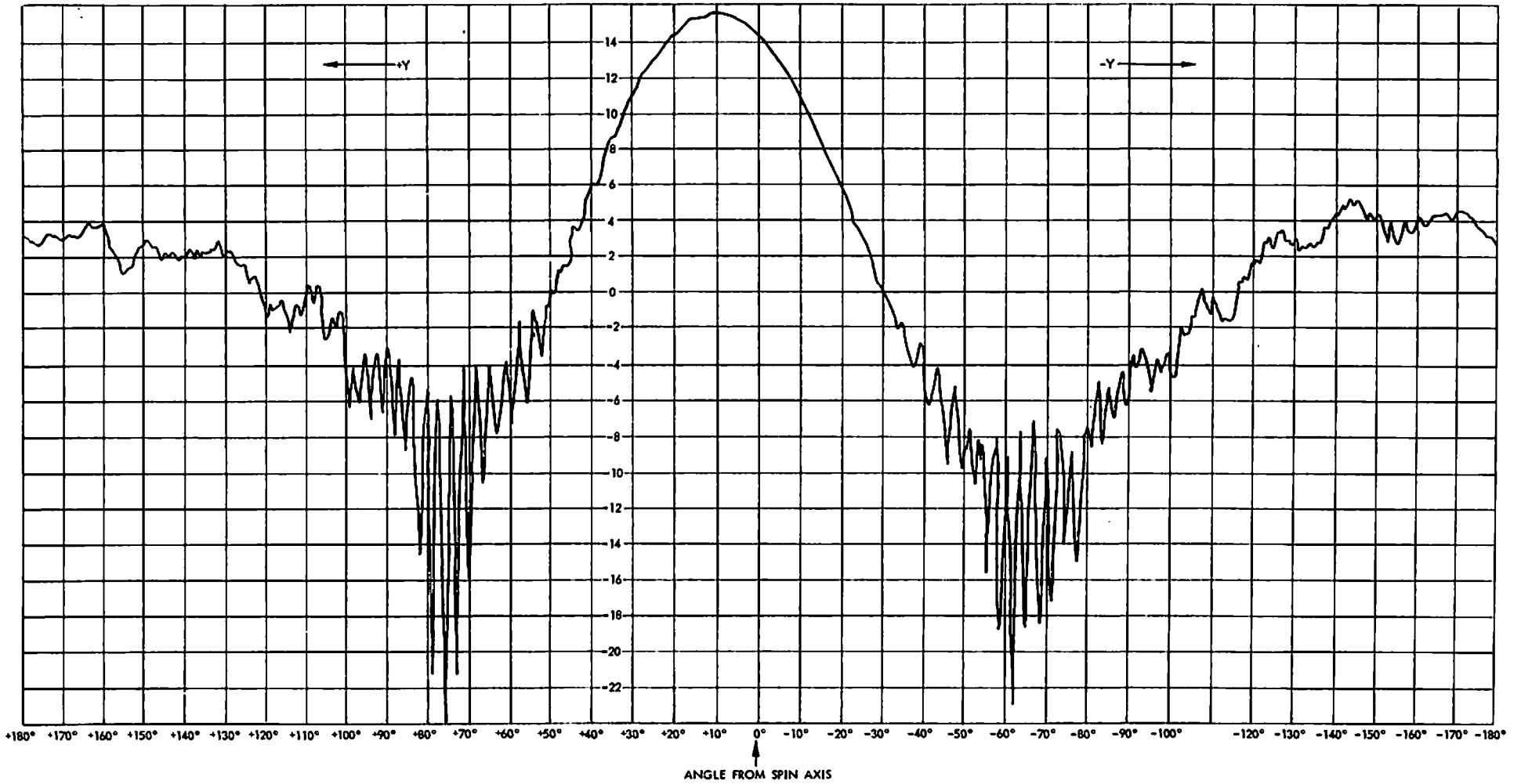


FIGURE 3.6-16. UPLINK MEDIUM GAIN/OMNI ANTENNA PATTERN

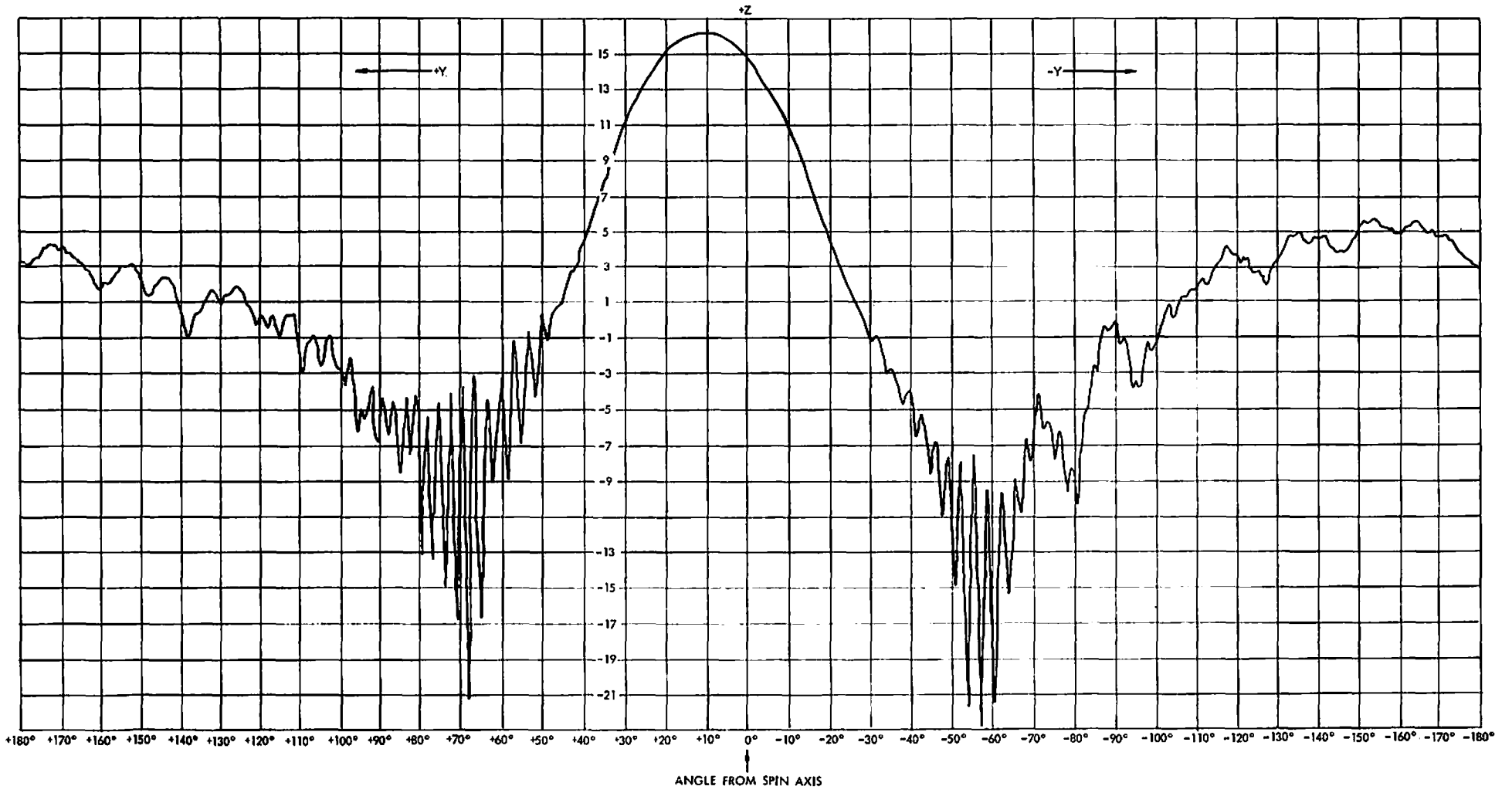


FIGURE 3.6-17. DOWNLINK MEDIUM GAIN/OMNI ANTENNA PATTERN

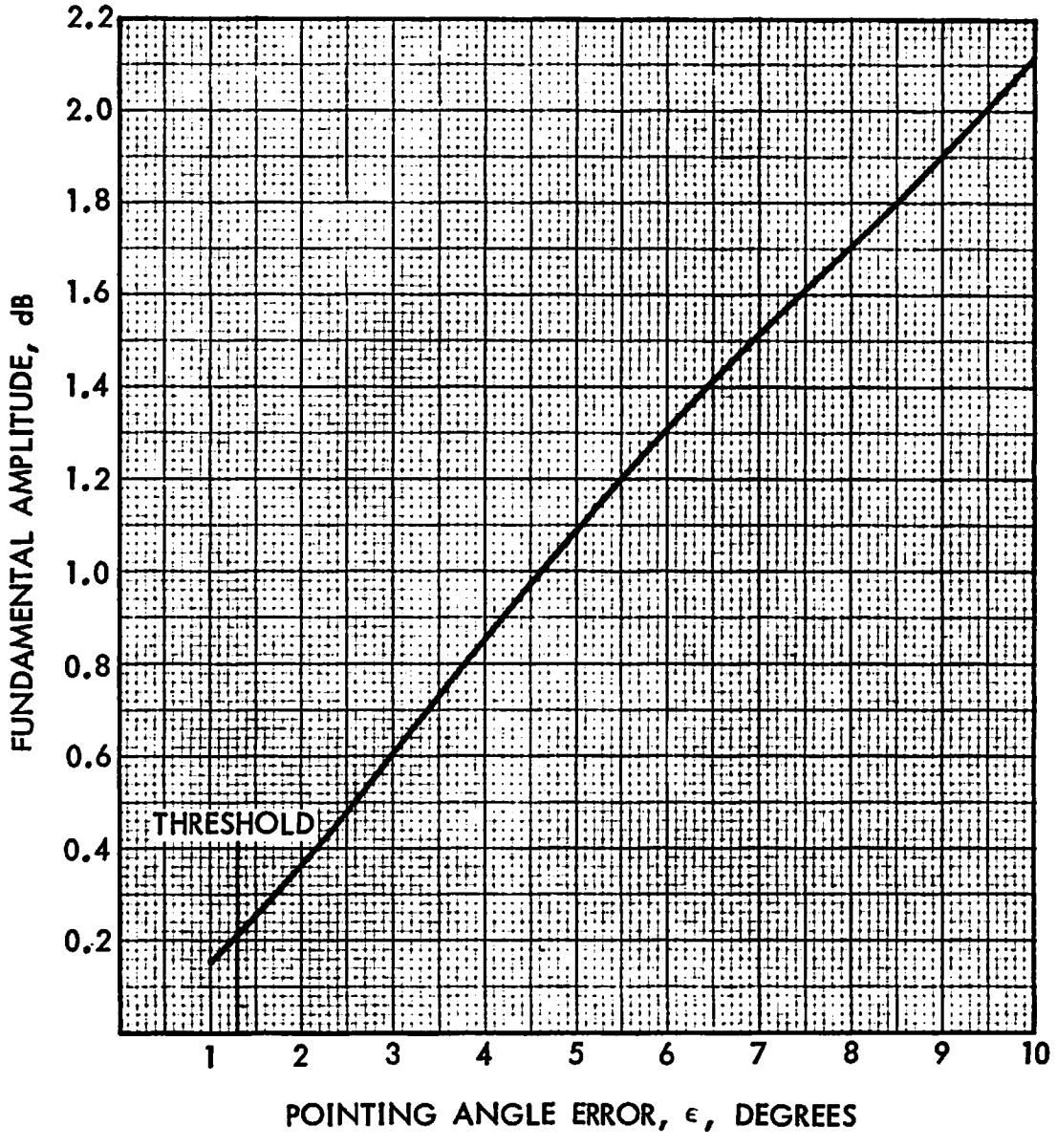


FIGURE 3.6-18. ENGINEERING MODEL MEDIUM GAIN ANTENNA PEAK CONSCAN SIGNAL AMPLITUDE

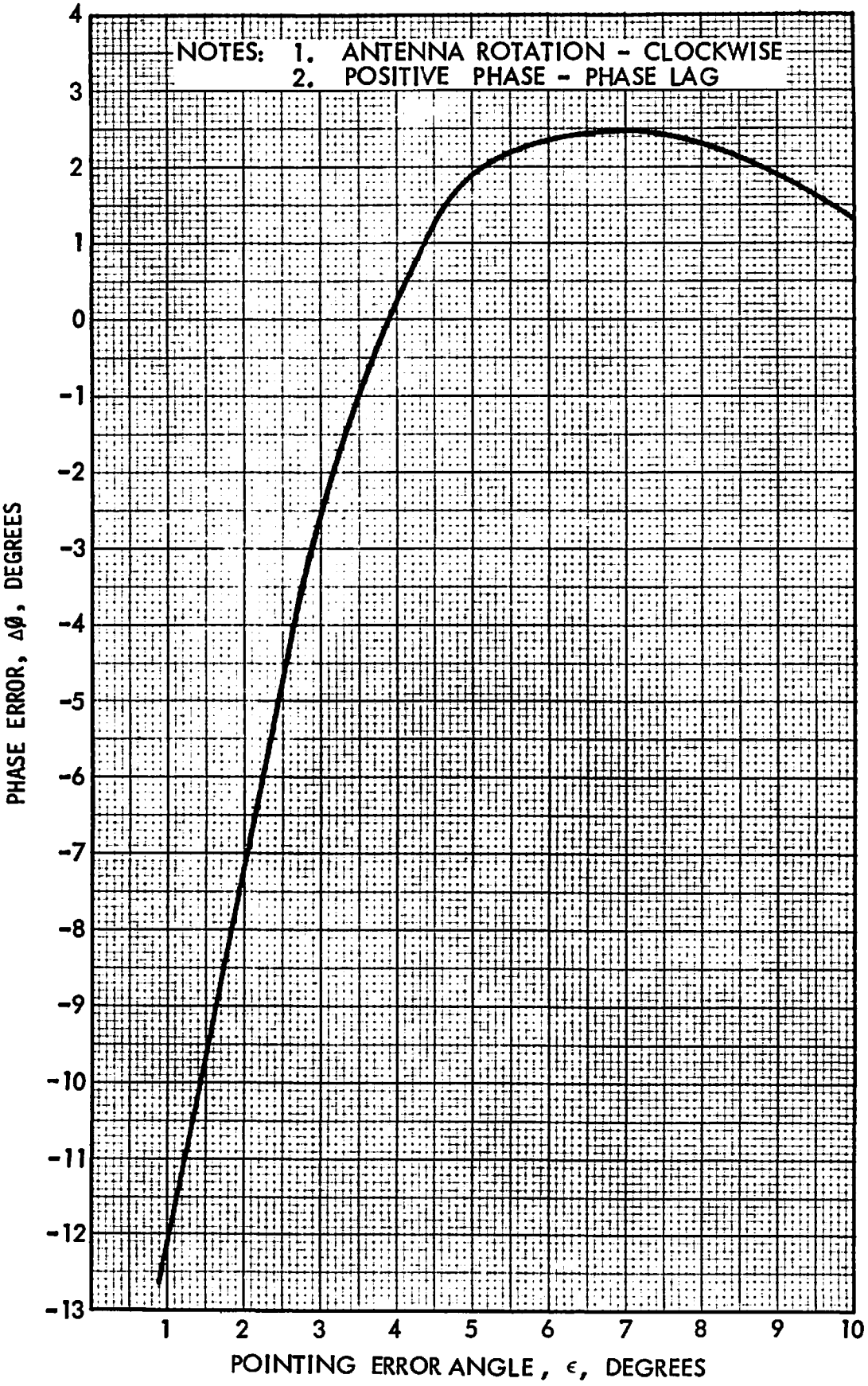


FIGURE 3.6-19. ENGINEERING MODEL MEDIUM GAIN ANTENNA PEAK CONSCAN SIGNAL PHASE

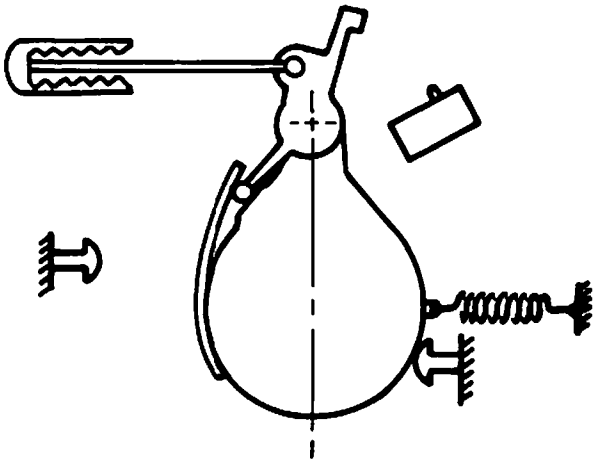
3.6.2.4 Feed Movement Mechanism

3.6.2.4.1 Description. The function of the feed movement mechanism is to offset the high gain antenna beam axis by about one degree (1db cross-over) from the spacecraft spin axis when fine conscan is required. The feed offsetting is initiated by ground command MFA9 (octal code 210). After completion of the conscan maneuver, the transmission of command MFA0 (octal code 356) causes the feed to be returned to the normal on-axis position.

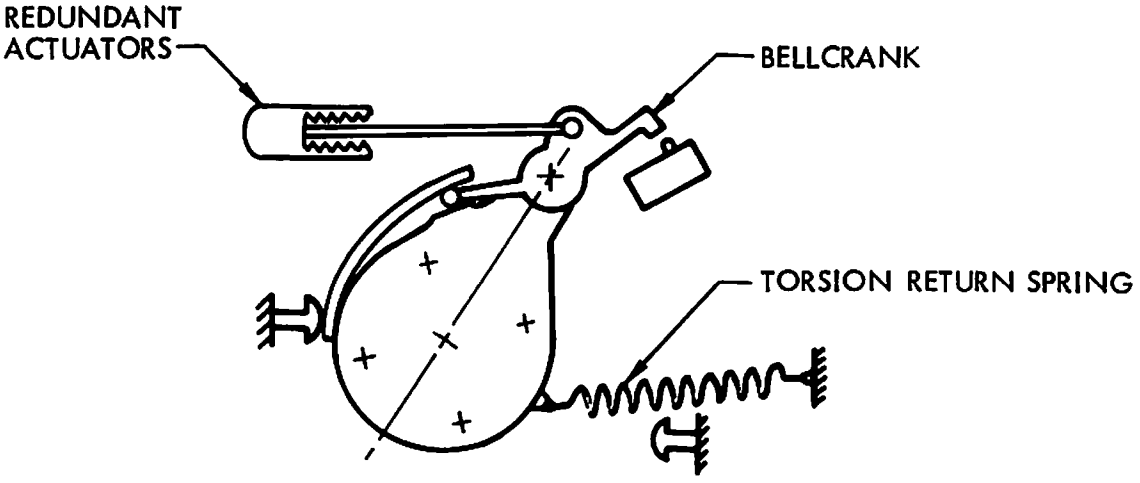
The feed movement mechanism can be described with the aid of Figure 3.6-20. The unit is spring loaded to the boresight position by a torsion spring located at the pivot point. Redundant thermal actuators are used individually to drive or pivot the feed to the conscan position. The thermal actuators are connected so that the inactive unit is not stroked by the active unit. The intent is to use one thermal actuator until a failure is noted and then switch to the other actuator. Series connected micro-switches are used for each actuator to shut off the heater power when the conscan position is reached. Actuation of either microswitch breaks the circuit. A separate microswitch is used to provide telemetry indication when the unit is in the conscan position.

For clarity, the torsion spring which returns the feed to the normal position is depicted as a tension spring. The actuator torque is in fact applied to a bellcrank which applies torque to the feed mounting plate through a leaf spring. When this feed mounting plate reaches its stop in the conscan position the bellcrank overtravels to actuate a microswitch to shut off power. As the thermal actuator cools and retracts, the bellcrank rotates back causing the switch to restore power. During the period the feed is in the conscan position, the thermal actuator is in a limit cycle condition of alternately heating and cooling and moving through a small amplitude while the feed mounting plate is restrained in the conscan position by the leaf spring loaded by the bellcrank. Removal of power permits the actuator to cool down and the torsion spring drives the feed mounting plate to the normal position.

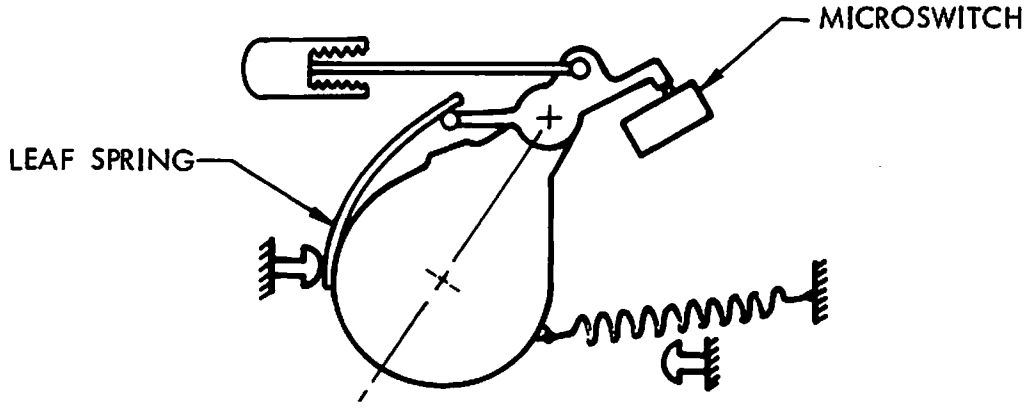
The thermal actuator is a thruster unit using Freon 21 as the active fluid and a bellows to retain the fluid and permit piston motion. The bellows is of electroless nickel with the actuating pressure applied to the exterior of the bellows. The outer housing carries a heater element to apply the heat necessary to boil the Freon, raise its pressure and cause the unit to stroke. In traveling from the normal position to the conscan position the actuator must initially build up sufficient pressure to overcome the main return spring, the resistance of the coax cable, the bellows spring force and the friction. Upon reaching the conscan position additional torque must be provided to overcome the preload of the overtravel leaf spring and then to initially operate the telemetry switch and finally the four power switches. There is torque required to deflect the telemetry switch lever after it has "bottomed out," deflect the switch lever spring and to overcome friction as well as additional bellows spring force. Once the power switch has actuated and turned off the power, the actuator cools down to a reduced pressure. Because of friction in the system there is no



NORMAL POSITION



CONSCAN POSITION



CONSCAN OVERTRAVEL

FIGURE 3.6 - 20. FEED MOVEMENT MECHANISM

motion of the bellcrank until the torque reduction is twice the value of the friction and then the bellcrank moves back under the action of the leaf spring torque. When sufficient differential motion of the power switch occurs (0.010 in max.) the power is switched on again and no angular motion occurs until the torque has increased by twice the value of friction torque. The bellcrank then moves in the overtravel direction to switch the power off and repeat the cycle.

To minimize power, the thermal actuator is designed to have a minimum amount of material to be heated to the operating temperature. For this reason the wall thickness on the housing is limited to a thickness of 0.010 inches. The actuators are insulated with multilayer mylar thermal insulation and are mounted and connected to the system with appropriate insulator materials as polysulfone and polyimide plastics.

Electrical power is controlled by the actuator heater resistance and varies with changes in the input voltage, 28.4 volts maximum and 26.78 volts minimum. The resultant possible power variation is 4.59 watts maximum and 4.00 watts minimum during the"on"portion of the limit cycle. The length of the power"on"portion of the duty cycle depends on the temperature and also varies from unit to unit , but is not greater than 50%.

3.6.2.4.2 Operation. The feed movement mechanism is pinned in the conscan position throughout the launch and ascent phase of the mission. A pin puller is actuated by a redundant ordnance device simultaneously with the initiation of the magnetometer deployment at approximately 33 minutes after liftoff. After the spacecraft emerges from the eclipse, the ambient temperature in the vicinity of the mechanism may be relatively low. When the pin is pulled, the feed can abruptly return to the normal position under the unrestrained force of the torsion spring and damage is a possibility. To avoid this potential damage the power will be applied to one or both actuators during the launch phase.

During the early portion of the mission, the ambient temperature is sufficiently high to heat the Freon and the feed movement mechanism will remain in the conscan position. This is illustrated in Figure 3.6-21 where the return time (time to travel from conscan to normal position) is plotted as a function of days from launch. The return time is here defined to be the total time from the power turn off to the time it reaches the on axis position.

During the remaining portion of the mission the feed movement mechanism normally will be held in the on-axis position. When conscanning is required, transmission of command MFA9 (or MFB9 for the back-up actuator) closes a switch and applies the power to the actuator heater. The time required to offset the feed, from command reception to conscan position, as a function of days from launch, is presented in Figure 3.6-22. The fact that the feed movement mechanism has reached the conscan position is telemetered to the ground and is indicated by the state "1" of a bilevel channel C-316. The feed position can also be established from the amount of amplitude modulation on received uplink or downlink signals. After completion of the conscan maneuver, the command MFA0 (MFB0 for the back-up

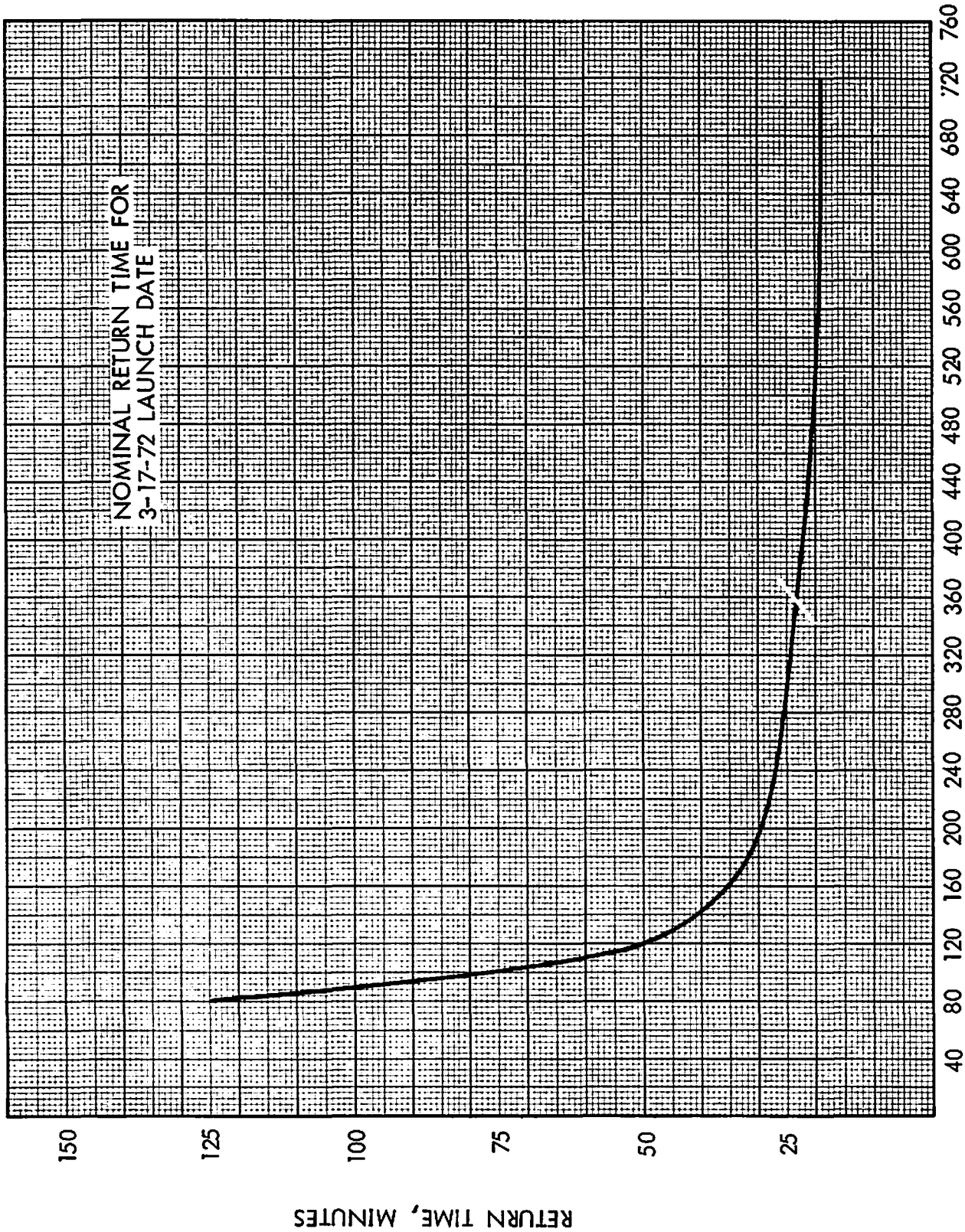


FIGURE 3.6-21. TYPICAL FEED RETURN TIME VERSUS DAYS FROM LAUNCH

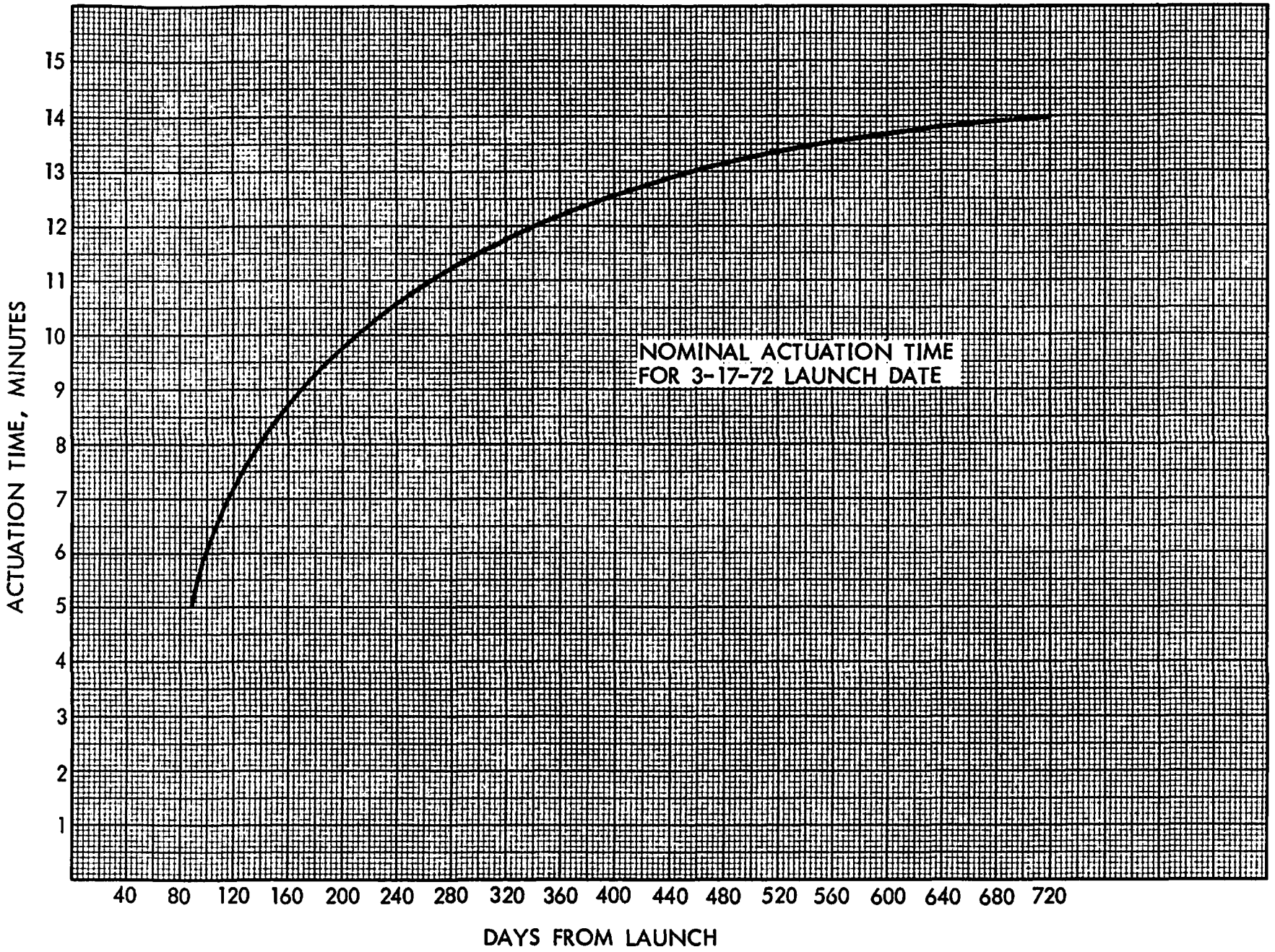


FIGURE 3.6-22. TYPICAL FEED ACTUATION TIME VERSUS DAYS FROM LAUNCH

actuator) is used to return the feed to the normal position. The on-axis position of the feed is indicated by the state "0" of the bilevel telemetry signal.

3.6.2.5 Diplexers and Transfer Switches.

3.6.2.5.1 Diplexers. The diplexers are used to isolate TWTAs from the receivers. The high gain antenna diplexer consists of a 6 resonator bandpass filter in the receiver channel and an 11 resonator bandpass filter in the transmitter channel. Additional rejection in the transmitter channel is obtained with the use of 3 band-reject resonators. A low pass filter in the transmitter channel rejects second through the fourth harmonics. At the critical receiver frequencies from 2107.5 MHz to 2112.5 MHz, a minimum of 100 db isolation is provided between the TWTAs and the receivers.

The medium gain antenna diplexer consists of a high gain antenna diplexer and a 3 db hybrid coupler. The purpose of the hybrid is to couple both the medium gain horn and the omni antenna to the receivers and TWTs.

Diplexer tests through the critical altitude region (4.5 to 60 km altitude) indicate that under the worst case antenna VSWR condition (1.5:1) corona takes place at about 9.4 watts applied to the diplexer input. While the nominal TWTA output is about 8 watts, corona might occur if the power exceeded 9.4 watts. Corona does not cause any permanent damage and can be detected on the ground by a sudden drop in the received carrier power of approximately 12 db. This loss diminishes when the critical altitude (nominally 4 minutes 23 seconds after launch) is exceeded.

3.6.2.5.2 Transfer Switches. The two identical coaxial transfer switches interconnect the high gain antenna assembly and the medium gain antenna with the redundant TWTAs and the receivers. The transmitter switch is controlled by ground commands. The receiver switch normally also is controlled by ground commands; however, if neither receiver exhibits a "signal present" indication for 36 hours, the timer in the CDU will switch to the alternate position. The CDU timer is controlled by the receiver signal presence detector. The transfer switches are mechanical latching RF switches. They are four-port devices having two modes of operation. Indicator switches are provided to telemeter switch position to the ground.

Basically, the transfer switch circuit consists of a TEM air dielectric transmission line of rectangular cross section. Sections of the center conductor are moved physically in or out of the circuit in such a way as to reroute the RF path. When the conductor is moved out of the path, it is forced against the wall of the transmission line. In this way, the section of the line between the opened contacts behaves as a waveguide below cutoff and provides isolation in excess of 70 db. The mechanical drive is supplied by two solenoids.

Operation of the switch during the critical altitude could lead to a corona problem. However, the earliest uplink acquisition by DSS-51 will take place at an altitude of about 14,000 km or about 27 minutes beyond the critical altitude.

3.6.2.6 Receiver.

3.6.2.6.1 General. The receiver employs a phase lock loop (PLL) to perform three primary functions: First, phase and frequency coherent outputs are provided to the transmitter drivers to produce a transponding ratio of 240/221 to allow measurement of 2-way doppler. The second function is to provide efficient demodulation for the command tones which are phase modulated on the uplink carrier. The third function is to provide an AGC output proportional to the earth pointing angle error to the conscan signal processor (CSP).

3.6.2.6.2 Unit Description. The block diagram configuration of the receiver is illustrated in Figure 3.6-23 with the major parameters listed in Table 3.6-3. The receiver employs three frequency conversions with four separate IF amplifiers, a reference oscillator, and a low phase noise voltage-controlled crystal oscillator (VCO). A narrow bandwidth (1.8 Hz) AGC loop is included to demodulate the amplitude modulated conscan data contained on the RF carrier. Demodulation of the FSK/PM command tones is accomplished by the phase detector in the phase lock loop. A coherent output to the transmitter driver that is exactly 12/221 times the uplink carrier frequency is also provided. Telemetry outputs from the receiver are identified in Figure 3.6-1.

The PPL is critically damped at threshold (carrier power = -149 dbm) and has a noise bandwidth of 20 Hz. As the carrier power is increased, the noise bandwidth increases until a strong signal (carrier power > -120 dbm) the noise bandwidth is essentially constant (95 Hz). The static and dynamic tracking characteristics are given in Figures 3.6-24 and 3.6-25 respectively. Noise bandwidth and tracking ability of the PPL are primarily set by the phase detector, the loop filter, and the VCO. Sufficient gain margin (>15 db) has been incorporated into the preceding IF amplifiers to negate the effect of possible gain reduction due to aging.

The AGC loop consists of the coherent amplitude detector, AGC loop filter, and AGC amplifier. Loop bandwidth is set by these three components to 1.8 Hz. Loop bandwidth does not vary appreciably over the normal receiver operating temperature and carrier power ranges. For carrier levels exceeding the nominal operating range (-70 dbm to -142 dbm) the AGC loop bandwidth will decrease slightly. Amplitude data present on the RF carrier is demodulated and band-limited (1.8 Hz) by the AGC loop. It is then scaled and supplied to the conical scan processor. A switch automatically removes this output when the receiver is not locked.

The VCO frequency is multiplied by six and fed to the transmitter driver for coherent downlink operation. When the receiver is locked this frequency is exactly 12/221 times the uplink carrier frequency.

When the coherent mode inhibit command from the CDU is present, the coherent output to the transmitter driver is disabled. A signal is also sent to the transmitter driver to enable the noncoherent oscillator. A signal is provided for telemetry indicating the mode of operation as being either coherent or noncoherent.

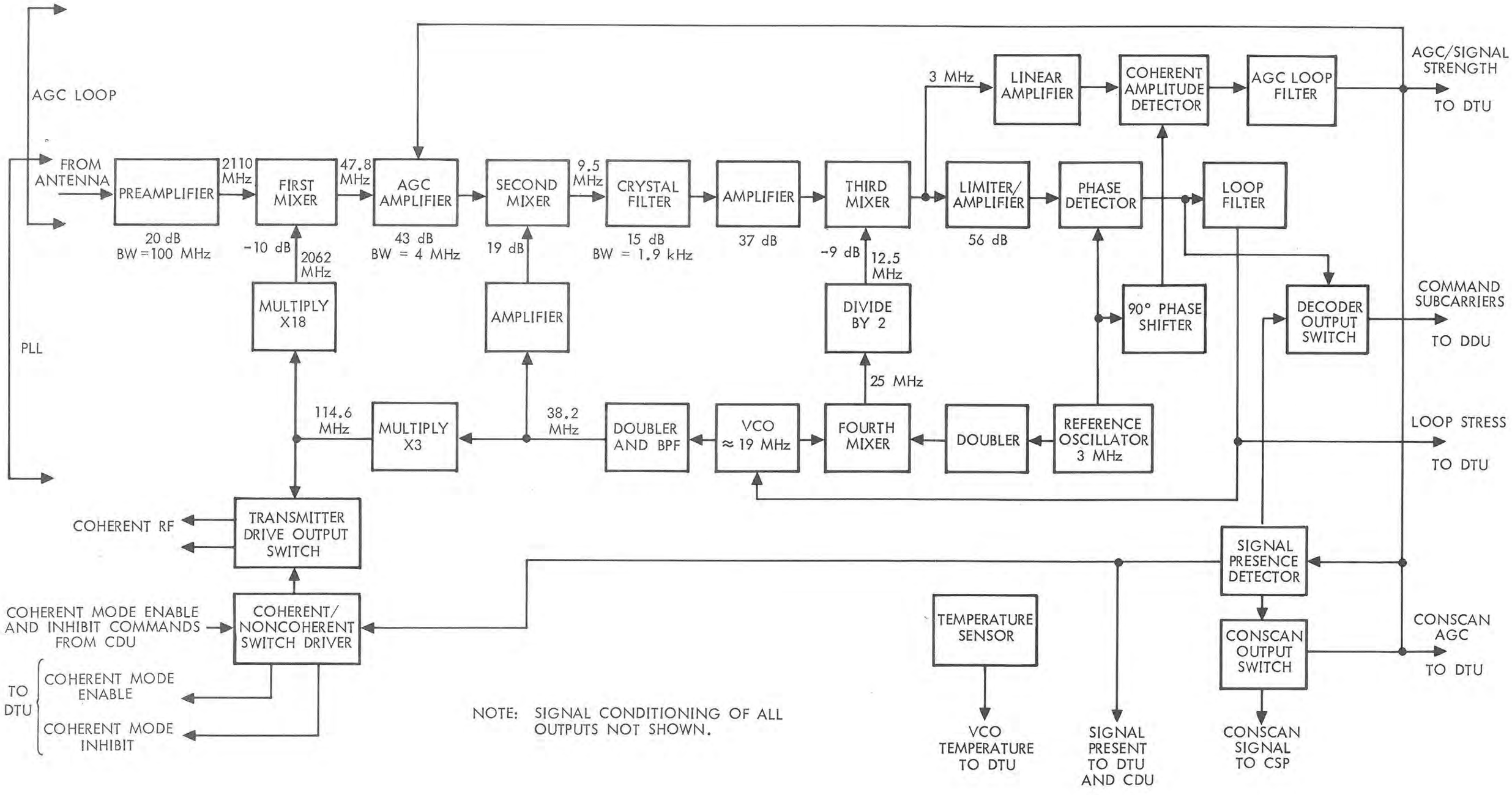


FIGURE 3.6 - 23. RECEIVER BLOCK DIAGRAM

Table 3.6-3. Major Receiver Parameter and Command List

1.	Receiver Frequencies	
	No. 1	2110.584105 MHz
	No. 2	2110.925154 MHz
2.	Receiver Threshold	-149 dbm
3.	PLL Noise Bandwidth at Threshold	20 Hz
4.	Receiver AGC/Conscan Slope	0.1 V/dB
5.	Telemetered Conscan Signal Slope	0.3 V/dB
6.	AGC 3 db Bandwidth	1.8 Hz

Commands			
Designator	Octal Code	Description	Remarks
COA 1	354	Coherent Mode Inhibit to Receiver #1	Inhibits Receiver No. 1 Output to Transmitter Drivers
COA 2	343	Coherent Mode Enable to Receiver #1	Enables Receiver No. 1 Output to Transmitter Drivers
COB 1	212	Coherent Mode Inhibit to Receiver #2	Inhibits Receiver No. 2 Output to Transmitter Drivers
COB 2	226	Coherent Mode Enable to Receiver #2	Enables Receiver No. 2 Output to Transmitter Drivers

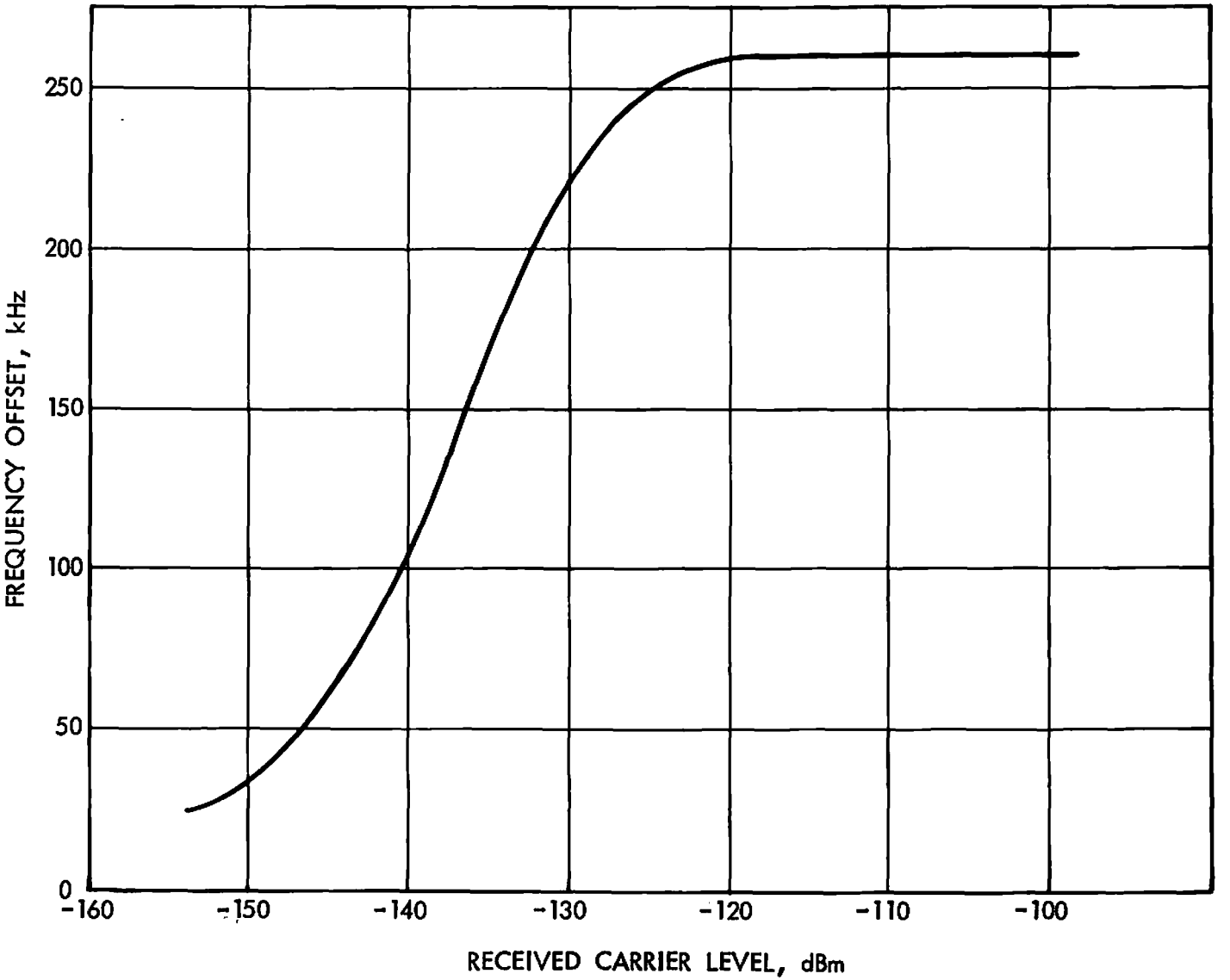


FIGURE 3.6 - 24. TRACKING RANGE VERSUS RECEIVED CARRIER LEVEL

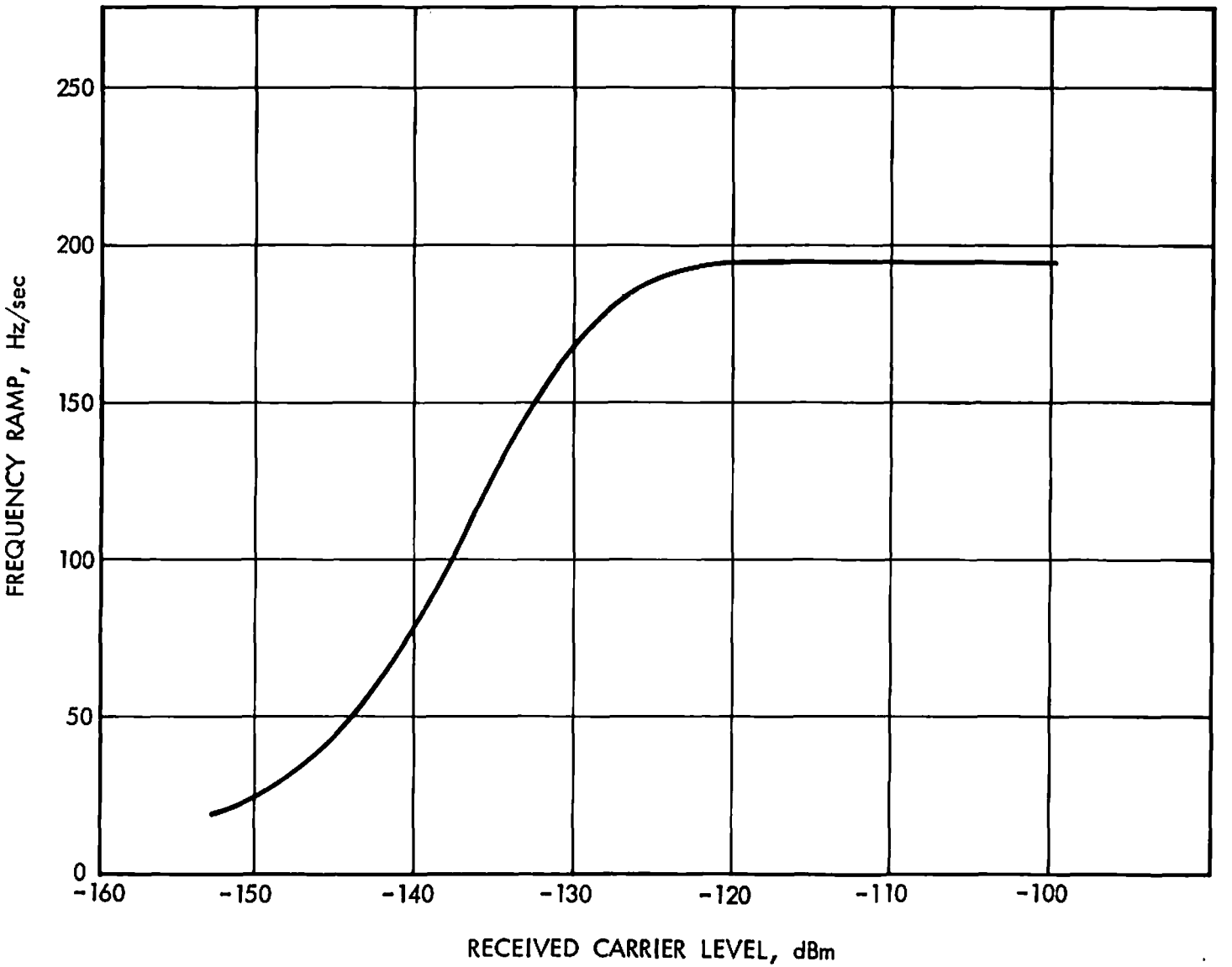


FIGURE 3.6 - 25. TRACKING RATE VERSUS RECEIVED CARRIER LEVEL

All analog telemetry signals are scaled between 0 and 3 volts. Under normal operating conditions these signals will be between these limits although normal operation of the receiver is possible with one or more signals being out of range (<0 volts or >3 volts). The value of the telemetered signal is indeterminate under these conditions.

3.6.2.6.3 Detailed Description. The preamplifier is a three-stage, low noise transistor amplifier that provides 20 dB of gain and effectively sets the overall receiver noise figure to less than 6 dB. The preamplifier is wideband and utilizes microstrip technology to obtain a minimum noise figure. The output of the preamplifier is band limited to 100 MHz by a combline bandpass filter which also provides the required image rejection.

The first mixer converts the received signal to the AGC amplifier frequency at 47.8 MHz. This first mixer consists of a balanced hybrid ring diode and provides an overall conversion gain of -10 dB. The AGC amplifier is a voltage-controlled variable gain IF amplifier at 47.8 MHz which provides 43 dB of gain at threshold. A low noise first stage with 20 dB of gain insures that the overall system noise figure is determined by the preamplifier. The AGC amplifier also provides about 70 dB of linear dynamic range and forms an important part of the conscan system.

The second mixer is a two-stage transistor mixer providing 19 dB of gain. The first transistor is a mixer that has 9 dB of gain and the second transistor provides the remaining 10 db of gain. The 9.5 MHz output signal from the second mixer is amplified by a single stage with 21 dB of gain. It is then band limited to 1.9 kHz by a quartz crystal filter. This is the critical bandwidth as far as interference is concerned. Two more stages of amplification provide the additional 37 dB of gain. The signal is then translated to 3.0 MHz in the third mixer which is a balanced diode mixer.

The limiter amplifier filters and hard limits the signal using diodes back-to-back. Three stages of transistor amplification follow to provide the required gain. The phase detector employed consists of two matched pairs of diodes balanced to the reference drive. The detector is a sampling type that takes samples at a 3 MHz rate. The DC output of the phase detector is a function of the phase difference between the signal and reference and is provided to the loop filter as the loop phase error signal. The loop filter sets the required loop noise bandwidth and damping factor for the loop. The filter is a monolithic operational amplifier which provides the required DC and AC gain. Dual junction field effect transistors are used to increase the input impedance of the operational amplifier to achieve the large time constant required to minimize the static phase error. Consistent with the required closed loop response, a pole is set at 0.0002 Hz and a zero is set at 2.2 Hz. The output of the loop filter is conditioned by a resistive network to provide the loop stress indication for telemetry. It also drives the VCO to close the PLL. The VCO consists of a single stage, low noise crystal oscillator with the frequency controlled through the use of varicaps in the feedback path. The scale factor of the VCO is set to 260 Hz volts.

The third mixer also drives a two-stage linear amplifier which provides 26 db of linear gain at 3 MHz to drive the coherent amplitude detector. The reference drive to the coherent amplitude detector is passed through a passive 90 degree phase-shifting network so that the output of this detector is proportional to the level of the input signal. This output is provided to the AGC loop filter which provides DC gain and a first order lowpass characteristic for the AGC loop. Consistent with the required closed loop response of the AGC, the lowpass corner frequency is set at 0.033 Hz and the DC gain is approximately 10. The output of the AGC filter drives three buffers. The first buffer consists of an operational amplifier which inserts a DC offset and drives the AGC IF amplifier. The second buffer supplies a gated output signal to the CSP. Before this output is gated, it is also fed to a buffer which removes the DC component and inserts a 1.5 DC offset. The output of this stage provides the conscan AGC indication for telemetry. The third buffer provides the necessary scaling to obtain the signal strength indication for telemetry.

The local oscillator drives for the first, the second, and the third mixers are derived from the VCO and the reference oscillator. The reference oscillator is a fixed frequency crystal oscillator. The output of the reference oscillator is doubled in frequency by a single transistor doubler with 20 db of gain. This signal is then added with the VCO signal in the fourth mixer which consists of a single transistor followed by a two-pole capacitively coupled filter to eliminate the unwanted mixer products. The output of the filter drives the divide-by-two circuit. The divide-by-two circuit employs a single varactor, two parallel traps in series with the varactor to control signal flow, and the input and output circuits to produce a 12.5 MHz output from the 25 MHz input. The output of the varactor is loosely coupled to a buffer amplifier in order to assure maximum stability of the divider. The output amplifier isolates the divide-by-two circuit and provides 20 db of gain delivering 0 dbm as a local oscillator signal to the third mixer.

The local oscillator frequencies for the first and second mixers are multiples of the VCO frequency. A balanced transistor doubler accepts an input from the voltage-controlled oscillator and provides an output to the second mixer buffer amplifier and to the multiply-by-three. A double-tuned filter is used in the output to reduce the level of spurious frequencies. The second mixer amplifier serves primarily as a buffer between the second mixer and the output of the filter and provides -5 dBm to the second mixer. The multiply-by-three accepts an input at approximately 38 MHz from the balanced doubler filter and delivers an output at approximately 115 MHz to the multiply-by-eighteen and an output to the transmitter drive output switch.

The multiply-by-eighteen consists of a buffer amplifier, a transistor multiply-by-three, a varactor multiply-by-six, and a combline filter. The buffer amplifier provides 8 db of gain to drive the multiply-by-three. The output of the multiply-by-three is 8 mW and is used to drive the varactor multiply-by-six. Limiting is employed in both transistor stages to maintain constant drive levels. The output of the varactor multiplier is fed

to a stripline filter. This filter is a three-pole bandpass filter to reduce the spurious outputs of the multiply-by-eighteen delivered to the first mixer. The filter is a maximally flat design with four parallel-coupled gaps.

The transmitter drive output switch consists of a double-tuned capacity-coupled filter buffer amplifier and a diode switch. High-conductance, high speed diodes are biases such that they may be back-biased or driven into conduction by the coherent/noncoherent switch driver. When the diodes are back-biased the shunt losses are negligible at 115 MHz. The signal in this condition is passed without attenuation with the buffer amplifier providing an overall gain of 0 db. When the diodes are driven into conduction the signal is attenuated by more than 65 db, cutting off coherent drive to the transmitter driver.

3.6.2.6.4 Telemetry. Table 3.6-4 lists the analog and the bilevel telemetry signals. Each of the analog signals is described in more detail below.

In Figure 3.6-26 VCO rest frequency versus telemetered temperature is presented. The normal receiver operating temperature range is 25 to 100 degrees Fahrenheit. This calibration curve is used by the DSS to establish the ground transmitter frequency sweeping range.

Loop stress of Figure 3.6-27 indicates frequency offset from the rest frequency shown in the previous figure. The maximum frequency offset that can be tolerated depends on the received carrier power level (see Figure 3.6-24) and is 260 kHz above -120 dBm and only about 40 kHz at receiver threshold.

Until the receiver has acquired lock, the AGC/signal strength telemetry shown in Figure 3.6-28 will indicate zero volts. Once the receiver is locked, the signal strength telemetry will rise to the value corresponding to the received carrier power level.

In Figure 3.6-29 the AGC/conscan slope is plotted versus AGC/signal strength telemetry. The nominal slope is 100 mV/dB and is linear to within $\pm$10 percent for received carrier power levels from -135 dBm to -95 dBm. This AGC slope calibration curve is used to predict pointing error angle magnitude from telemetered parameters A sin θ and A cos θ (see Section 3.6.2.8.5).

The slope of Figure 3.6-29 multiplied by 3 gives the calibration curve for the conscan signal telemetry as a function of received carrier power level. As can be seen, nominally, the slope for the conscan signal telemetry will be about 300 mv/dB. In addition, due to the highpass filter, there is 3 to 4 degrees phase lead between the input to the conscan signal processor and the telemetry. The conscan signal telemetry can be used for a conscan backup mode of operation in case of the conscan signal processor failure.

Table 3.6-4. Receiver Telemetry List Analog Channels

Analog Channels			
Word		Measurement	Measurement Range
RCVR 1	RCVR 2		
C-223	C-227	VCO Temperature	20 to 110°F
C-232	C-213	Signal Strength	-145 to -63 dBm
C-212	C-229	Loop Stress	-100 to +100 KHz
C-111	C-121	Conscan Signal Slope	± 4 dB MA

Bilevel Channels				
Word/Bit		Measurement	State "1"	State "0"
RCVR 1	RCVR 2			
C-308/1	C-308/2	Signal Present	Receiver Locked	Receiver not locked
C-308/3	C-308/4	Coherent Mode Enable	Coherent Mode	Non-coherent oscillator in the transmitter driver enabled

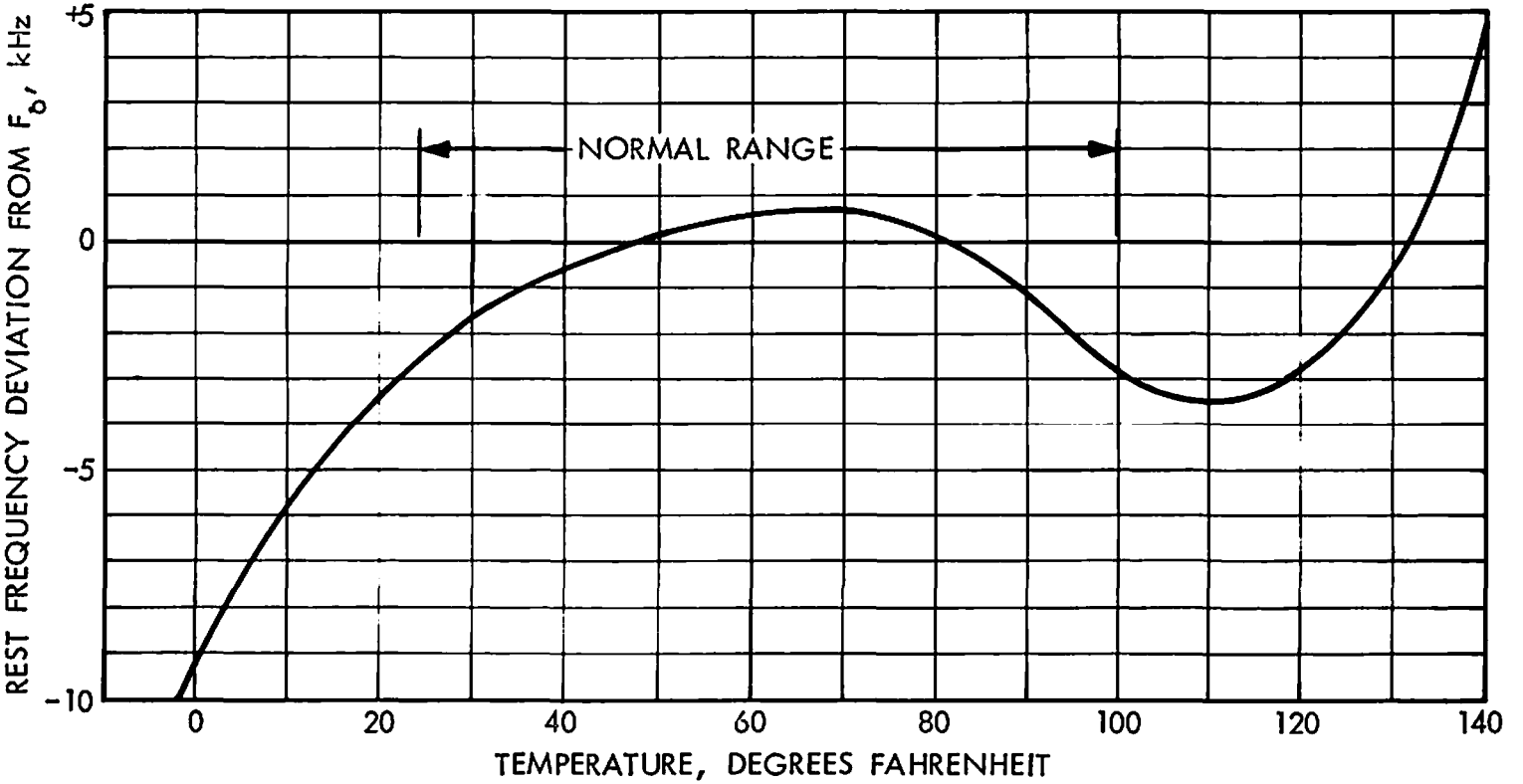


FIGURE 3.6 - 26. VCO REST FREQUENCY VERSUS TELEMETERED TEMPERATURE

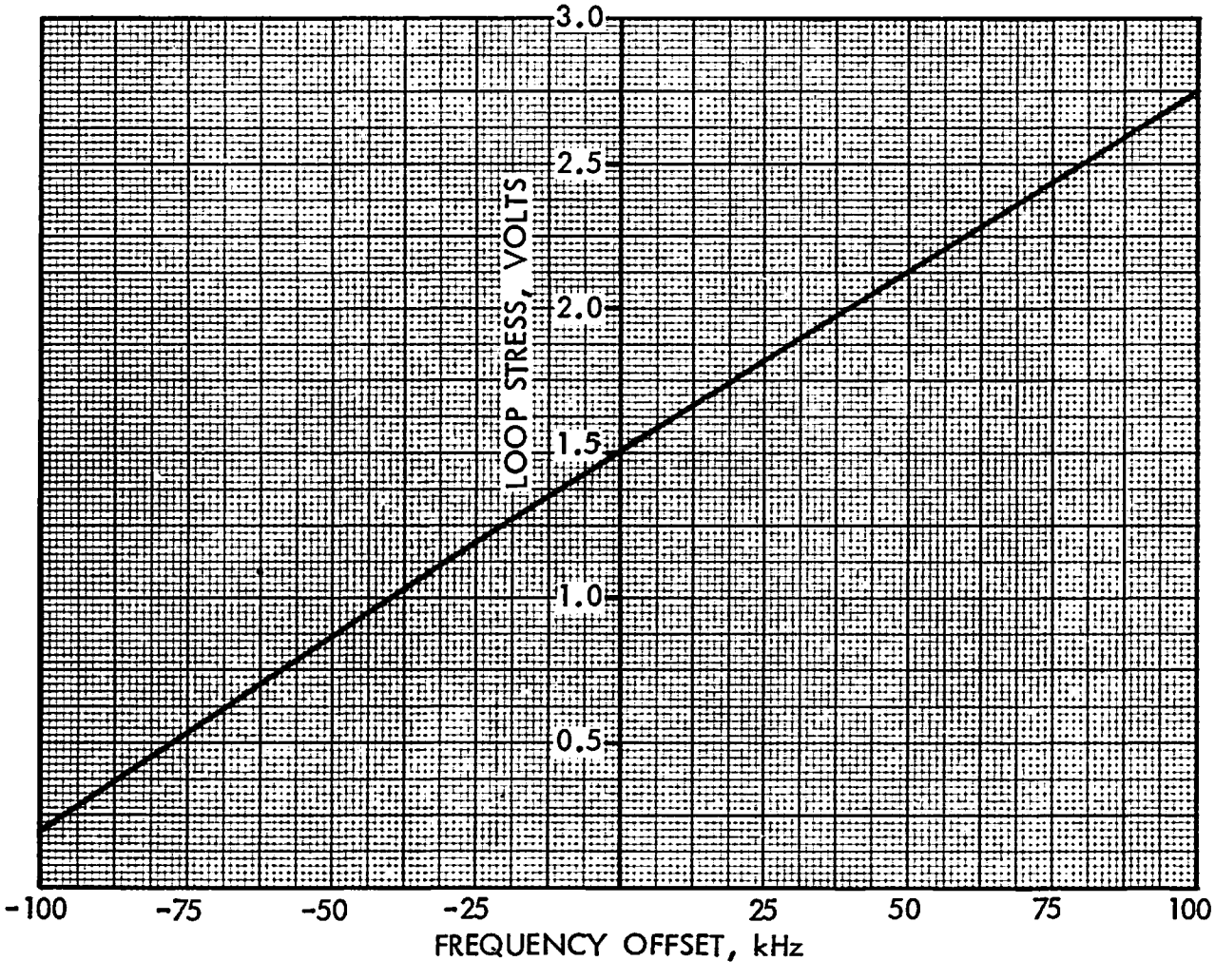


FIGURE 3.6 - 27. LOOP STRESS TELEMETRY VERSUS FREQUENCY OFFSET

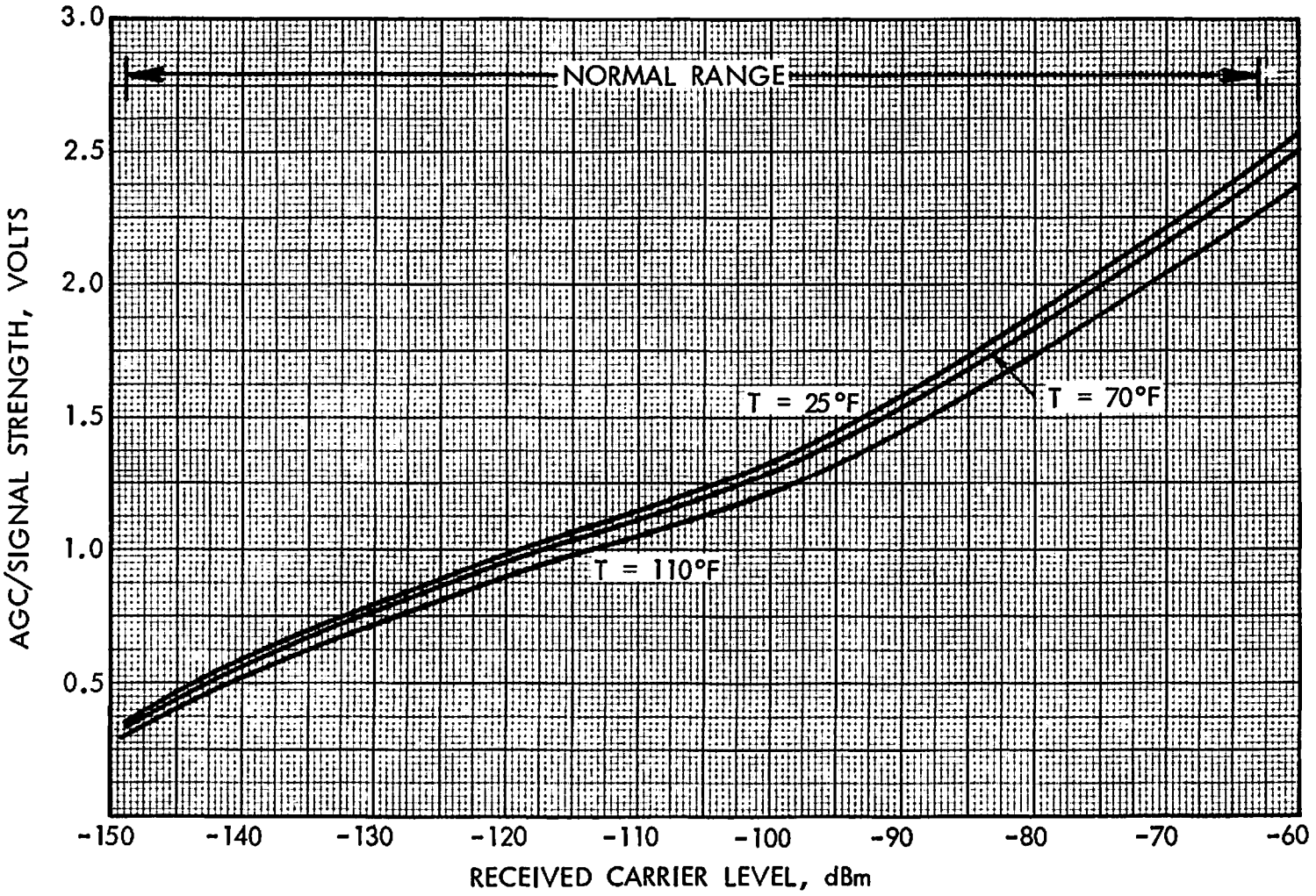


FIGURE 3.6-28. ENGINEERING MODEL AGC/SIGNAL STRENGTH TELEMETRY VERSUS RECEIVED CARRIER LEVEL

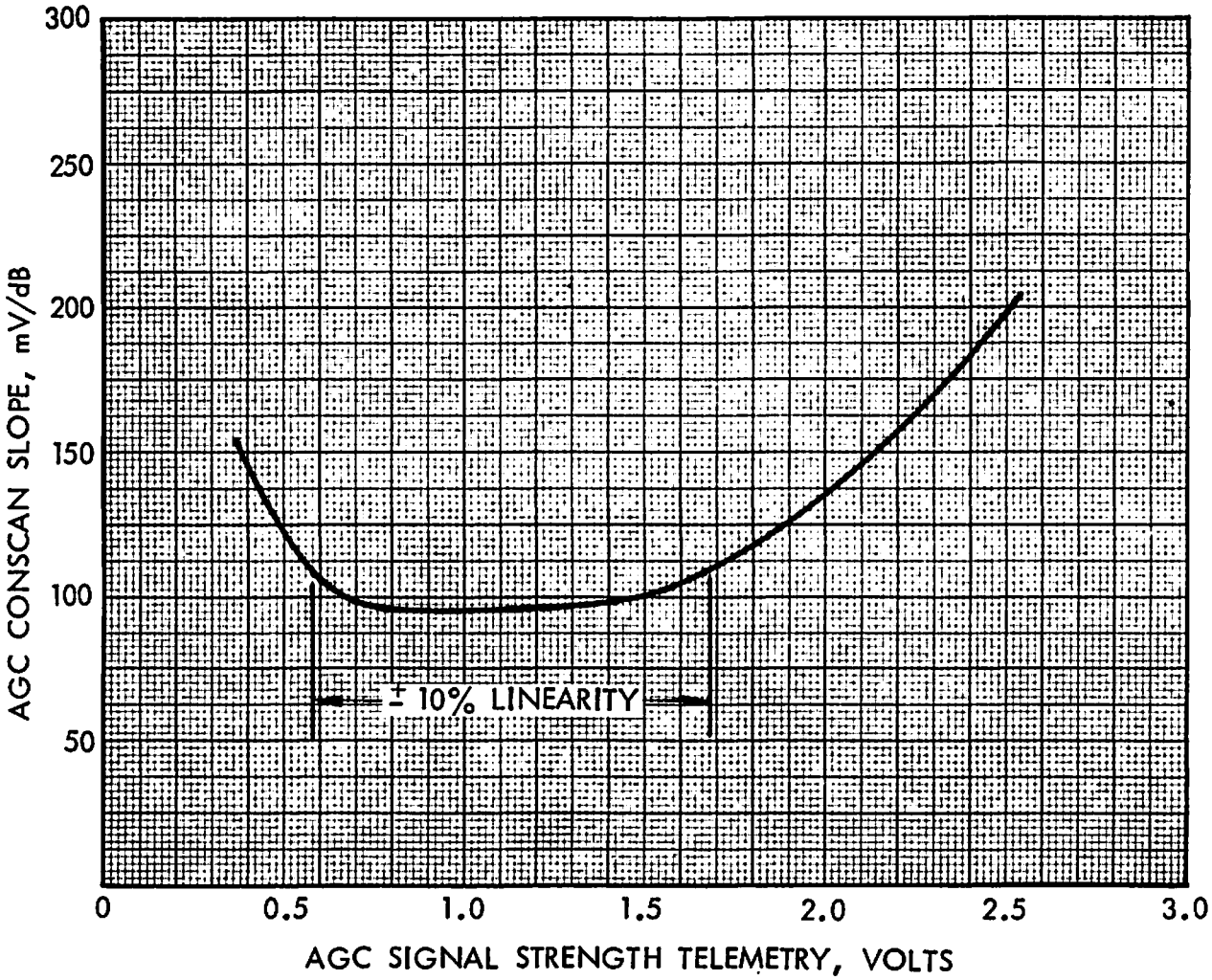


FIGURE 3.6-29. ENGINEERING MODEL AGC CONSCAN SLOPE VERSUS AGC SIGNAL STRENGTH

3.6.2.6.5 Operation. Throughout the mission both receivers remain powered with one receiver connected to the medium gain/omni antenna and the other to the high gain antenna through the receiver transfer switch. When there is no uplink, the out of lock condition will be indicated by the state "0" of the bilevel telemetry channel C-308 (see Table 3.6-4). In addition, the AGC/signal strength telemetry will indicate 0 volts and the loop stress telemetry will be about 1.5 volts. During this time the non-coherent oscillator in the transmitter driver provides the RF signal to the TWTA. Since the two receivers are assigned different frequencies, the desired receiver is selected by the transmitted uplink frequency. The rest frequency of the VCO will be predicted from the telemetered temperature (Figure 3.6-26), and an approximate frequency range, over which the DSS transmitter is swept, will be selected. Normally, sweeping once through the predicted frequency range results in the uplink acquisition and this will be indicated by the telemetry channel C-308 switching to State "1" and by a change in the downlink frequency if the coherent mode has been enabled. The receiver lockup will also be indicated by the increase in the AGC/signal strength telemetry. After verifying that the telemetered received carrier power level is in agreement with the prediction (Figure 3.6-7) the command modulation may be turned on. The receiver is now ready for command transmission.

Assuming that the receiver coherent mode enable command had been previously transmitted, as soon as the signal presence detector indicates that the receiver has acquired, downlink will be coherent with the uplink. Switching from transmitter driver oscillator to the VCO can be detected on the ground from the shift in received carrier frequency, which may cause loss of DSS receiver lock.

3.6.2.7 Transmitter Driver.

3.6.2.7.1 Description. The transmitter driver is a phase modulated transmitter which produces 50 to 80 milliwatts of power at 2292 MHz to drive the TWTA. As shown in Figure 3.6-30, the transmitter driver consists of an oscillator, modulator, frequency miltipliers and power amplifiers.

The temperature compensated crystal oscillator (TCXO) is used as a stable frequency source when coherent mode operation is not desired. When the receiver is not locked to an uplink signal, the TCXO output is automatically energized. The output of the oscillator is buffered by an amplifier which establishes a nearly constant 50 ohm load. A stable load is required to insure frequency stability over the operating temperature range. A frequency tripler follows the buffer amplifier and provides about one milliwatt of output at 114.6 MHz.

In the phase modulated amplifier the RF signal is phase modulated by the 32.768 kHz squarewave subcarrier which is bi-phase modulated by telemetry data from the DTU. The RF input to the modulator is provided, through a resistor summing network, by either one of the two receivers or the TCXO. The modulator signal from the DTU is a 32.768 kHz square wave subcarrier bi-phase modulated by the data at bit rates ranging from

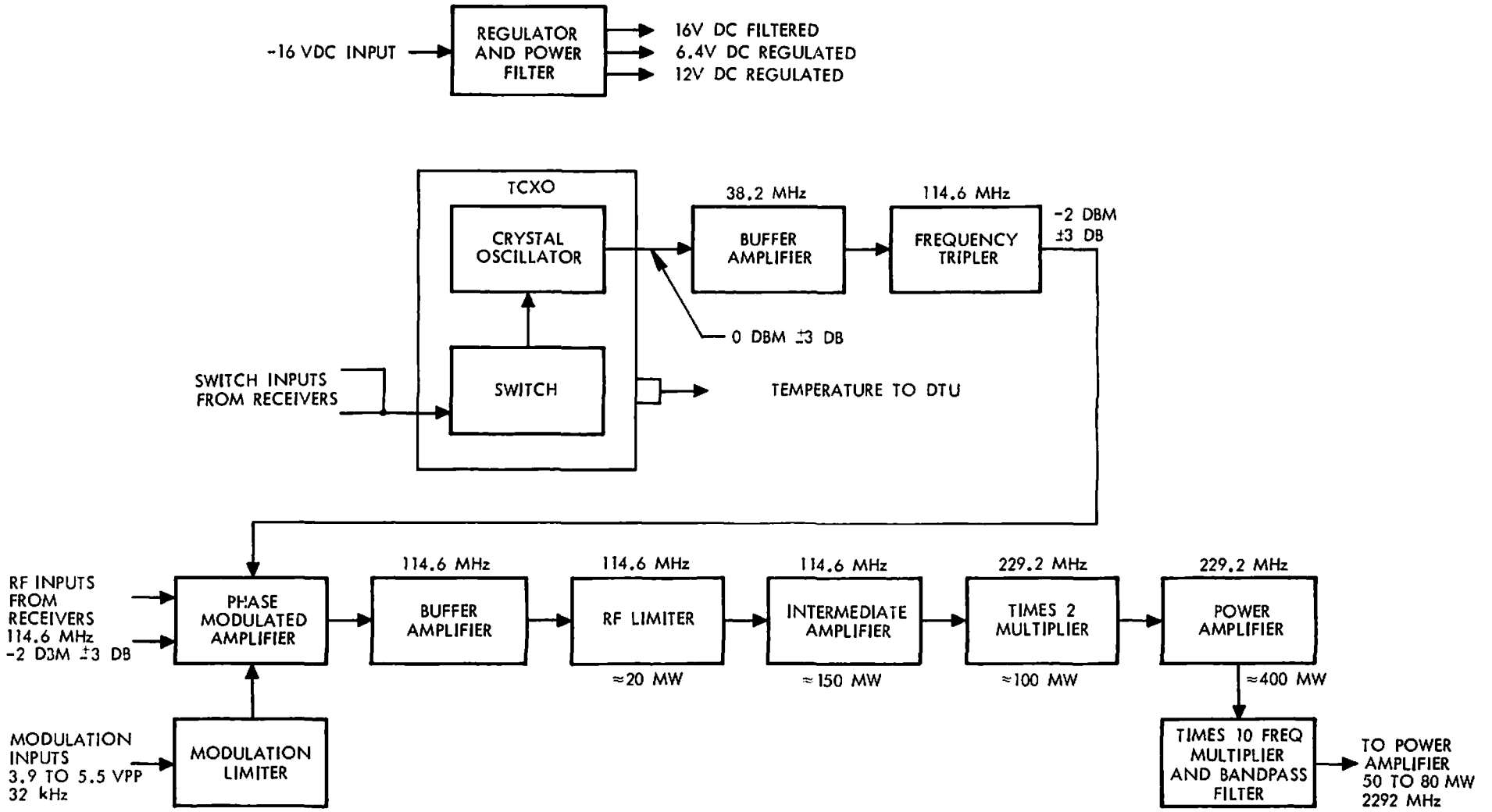


FIGURE 3.6 - 30. TRANSMITTER DRIVER BLOCK DIAGRAM

16 bps to 2048 bps. The modulation index nominally is 1.15 radians and results in 0.8 db data modulation loss and 7.8 db carrier modulation loss. The modulation limiter circuit provides a constant output signal to the modulator for input levels between 3.5 and 5.5 volts peak-to-peak and maintains a constant phase derivation.

The output of the modulator stage is supplied to a buffer amplifier stage followed by a RF limiter stage. The limiter stage output level is approximately 20 milliwatts and is adjusted by a test select resistor to provide the required drive to the following stages. The limiter stage provides a nearly constant output RF signal level during variations of input signal level from either the receivers or the auxiliary oscillator. The intermediate power amplifier provides approximately 150 mw of output power at 114.6 MHz. The output from the intermediate power amplifier is applied to the times two frequency multiplier with an output power of approximately 100 milliwatts. The power amplifier accepts the output from the times two multiplier and provides amplification to a level of approximately 400 milliwatts. The amplifier operates at 229.2 MHz.

The RF output signal from the power amplifier is coupled through the impedance matching network to the times 10 (frequency) multiplier. The bandpass filter is a three-pole filter designed to reduce the spurious harmonics on the output signal. It is designed to provide nearly 50 db of rejection to the times 9 and times 11 components of the drive signal to the multiplier. The power output from the bandpass filter will range between 50 and 80 mw and will be at approximately 2292 MHz frequency.

The voltage regulator accepts the input -16 vdc and provides regulated voltages of -6.4 vdc and -12 vdc as well as a filtered -16 vdc. Temperature stable voltage reference diodes are used as the reference source for the regulator to maintain a stable output voltage throughout the operating temperature range.

The most critical transmitter driver parameter (in the non-coherent mode) is the TCXO long term frequency stability versus temperature. For this reason the TCXO temperature is monitored and telemetered to the ground by a thermistor placed on the circuit board near the TCXO. The thermistor output is signal conditioned in the CDU and sent to the DTU for transmission to the ground. The long term frequency stability versus temperature is shown in Figure 3.6-31. Another parameter which is of some interest since it affects the division of power between carrier and data is the modulation index. The variation of the modulation index as a function of telemetered temperature is presented in Figure 3.6-32. The variation in the modulation index primarily occurs because of the changes in the phase modulated amplifier components.

3.6.2.7.2 Operation. Without an uplink signal, the receivers are not in lock. In this condition, the receivers automatically provide TCXO enable signals to both transmitter drivers. During this one-way mode of operation, the TCXO provides a stable RF frequency source for the downlink. When

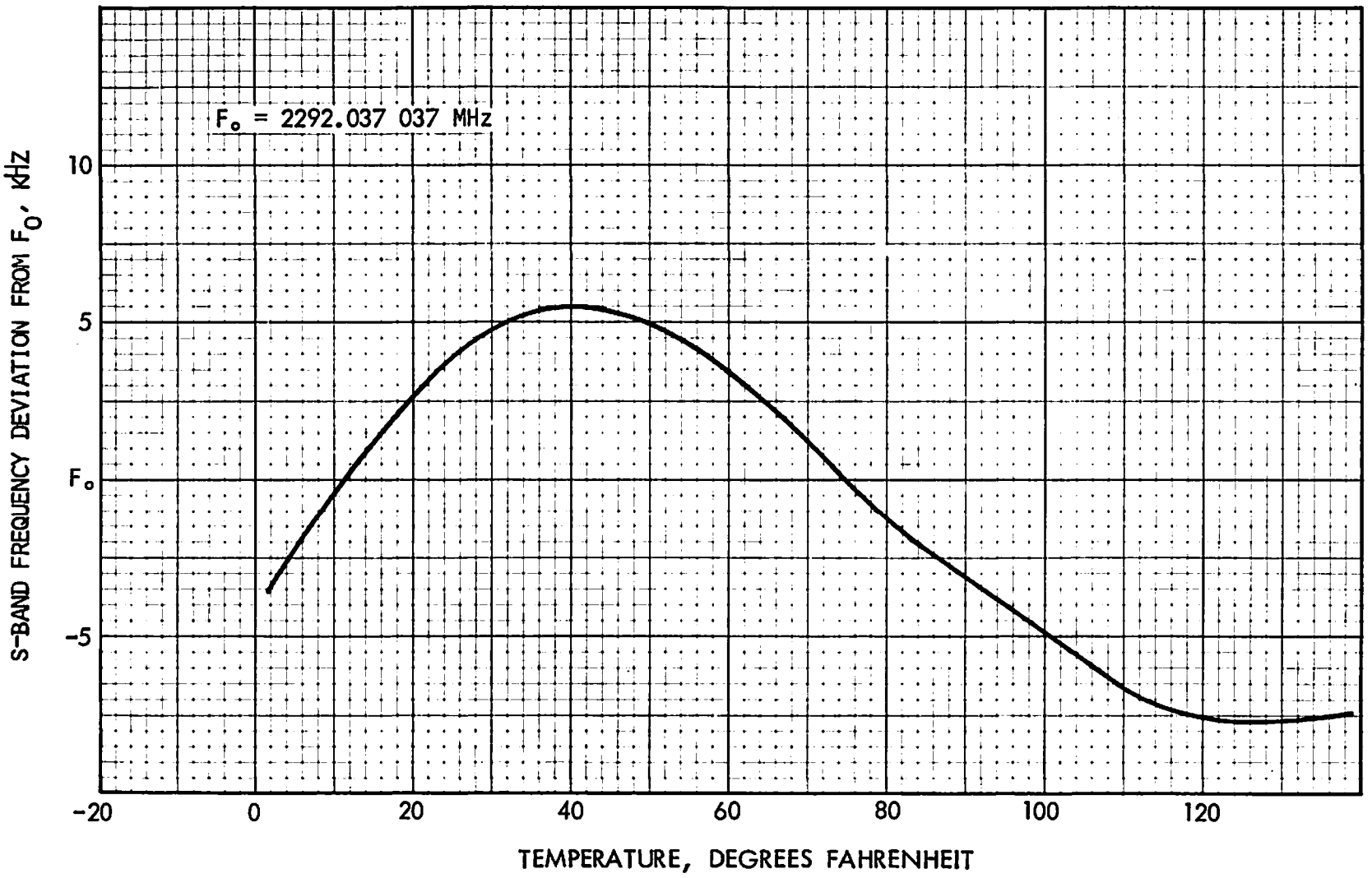


FIGURE 3.6 - 31. LONG TERM FREQUENCY STABILITY VERSUS TEMPERATURES

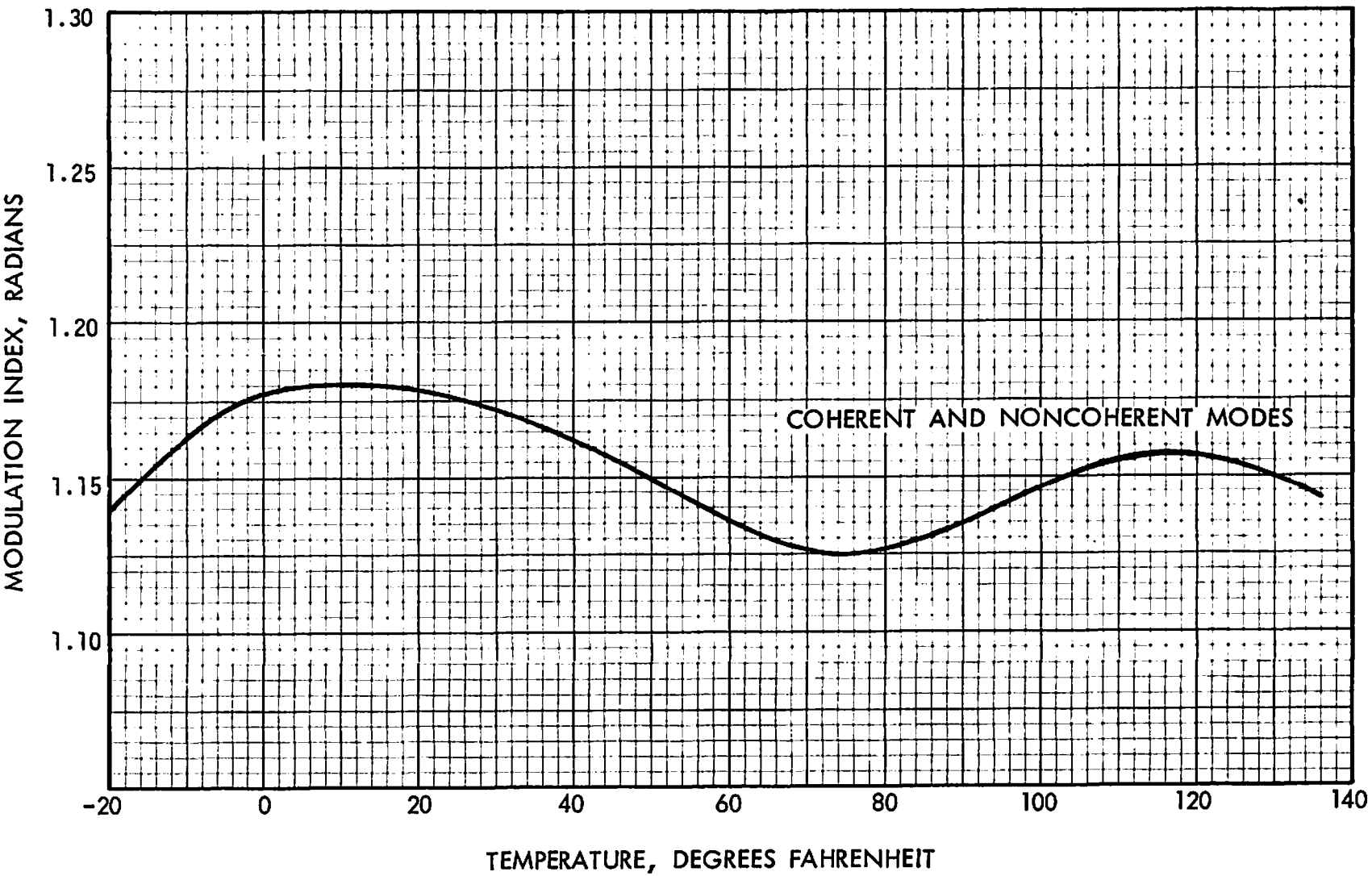


FIGURE 3.6 - 32. MODULATION INDEX VERSUS TEMPERATURE

there is uplink, and one of the receivers is in lock, the communication system can be operated either in the coherent or the non-coherent modes. The coherent/non-coherent modes are selected by the ground commands directed to the receiver which is in lock. When the coherent mode enable command is transmitted, both transmitter drivers are supplied with the RF source from the receiver. When a coherent mode inhibit command is transmitted to the receiver, the TCXOs in both transmitter drivers are enabled. Normally the power to one transmitter driver is turned on and the other is turned off. The desired transmitter driver/TWTA pair (transmitter driver No. 1--TWTA No. 1, transmitter driver No. 2--TWTA No. 2) is selected by a single ground command. The actual DC power switching (-16 V) for the transmitter driver takes place in the Electrical Power Subsystem.

During switching from the non-coherent to the coherent modes using the "enable coherent mode" commands (COA 2 and COB 2), the TCXO remains powered, but no longer controls the downlink frequency. The switch in the TCXO stops the oscillator from oscillating when the coherent mode enable is sent. Oscillator stability of 5 parts in 10^{11} for the first two min. or 3 parts in 10^{11} for the first hour still applies. The TCXO power is on at all times except when switched on or off by changing from one TWTA-driver pair to the other (commands TRA9, TRA0, TRB9, and TRB0).

3.6.2.8 Travelling Wave Tube Amplifier (TWTA).

3.6.2.8.1 General. One of the most critical spacecraft functions for the downlink communications is the RF power amplification. For this reason redundant TWTAs are used. Normally, the power to one of the amplifiers will be turned on and the power to the other amplifier will be turned off using ground commands listed in Table 3.6-5. The active TWTA will be connected to the desired antenna by the transmitter transfer switch.

The two major assemblies of each TWTA (see Figure 3.6-33) are the travelling wave tube (TWT) and the power supply. The TWT operates in the frequency range of 2290 to 2300 MHz and has a nominal power output of 8 watts. The power supply derives its power from the spacecraft 28.0 volt DC line and provides all of the operating voltages for the TWT. It consists of a solid state DC to DC converter and a regulator circuit. The total primary power consumed by the TWTA, as shown in Table 3.6-5, nominally is 27.8 watts.

Table 3.6-5. Major TWTA Parameters and Commands

1. Nominal Output Power			8 watts
2. Total Primary Power Required			27.8 watts
3. Total TWT Power Consumption			25.1 watts
4. RF Output Frequency Range			2290 to 2300 MHz
Commands			
Designator	Octal Code	Description	Remarks
TRA 9	223	Transmitter A ON	Turns TWTA and Driver A ON
TRB 9	310	Transmitter B ON	Turns TWTB and Driver B ON
TRA 0	232	Transmitter A OFF	Turns TWTA and Driver A OFF
TRB 0	331	Transmitter B OFF	Turns TWTB and Driver B OFF

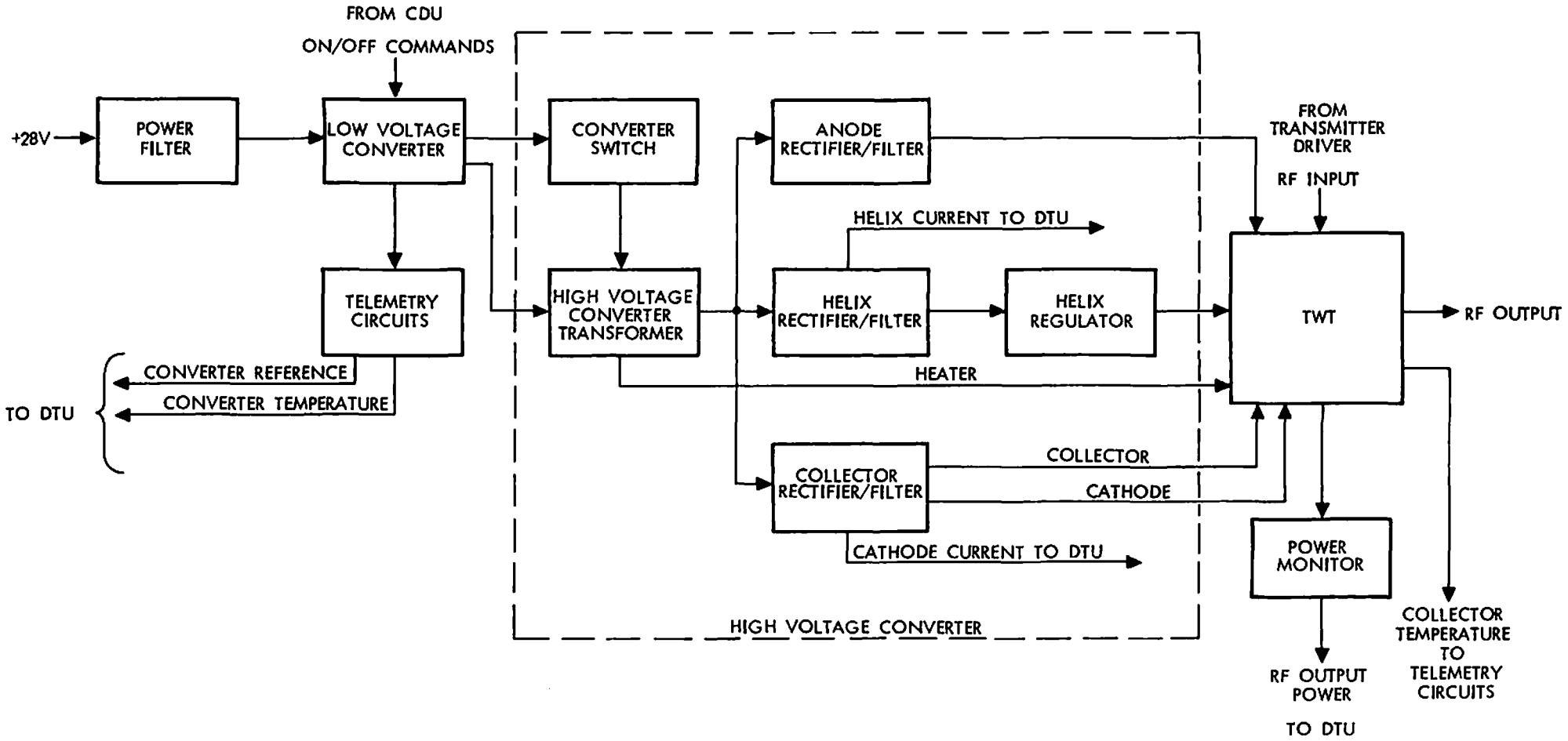


FIGURE 3.6 - 33. TWTA BLOCK DIAGRAM

3.6.2.8.2 Description. The TWT is derived from the Watkins Johnson WJ-274 space tube series and was developed specifically for the Pioneer F/G. The RF power output at the start of the mission of 38.9 dBm may degrade to 38.7 dbm at encounter. A simplified TWT circuit diagram, Figure 3.6-34, is included to clarify the power supply connections.

The RF power monitor provides to the DTU a 0 to 3 vdc voltage corresponding to 0 to 11 watts incident RF power through the RF power monitor. The RF power monitor consists of a directional coupler with approximately 30 db directivity, an impedance transformer, a voltage detector, and a filter.

The function of the power supply is to convert the direct current 28 volt input power into the high voltage regulated and filtered power required by the TWT. The TWTA power supply consists of a low voltage converter and a high voltage converter. The main function of the low voltage converter is to provide the 10 kHz drive to the high voltage converter switching circuit. It also supplies auxiliary power to the telemetry circuits, limits turn-on current transients, and provides TWTA ON/OFF command acceptance capability. The low voltage converter consists of an oscillator driver and the associated circuits.

The high voltage converter functionally consists of the converter switch, transformer, three rectifier-filters and a helix regulator. First, the 28 volts commutated DC from the converter switch is transformed to the high voltage required by the TWT. This waveform is then rectified and filtered to provide the high voltages shown in Figure 3.6-34. The helix regulator regulates the helix voltage to 0.28v peak-to-peak ripple (0.028 percent) to insure that the incidental frequency modulation and amplitude modulation are at least 52 db below the carrier. Furthermore, close regulation of the helix voltage insures gain flatness below saturation and discriminates against noise from primary interference currents. The high voltage transformer also provides 2.3 volts to the TWT heater.

The power filter smoothes the ripple on the 28 vdc TWTA/experiment bus, reduces TWTA "ON"/"OFF" switching transients to the spacecraft 28 vdc bus and attenuates the 10 kHz converter switching fundamental and harmonics on the 28 vdc bus.

The telemetry circuits provide conditioning to the collector temperature, low voltage converter temperature and primary 28 vdc voltage telemetry signals.

3.6.2.8.3 Telemetry Outputs. Table 3.6-6 lists the telemetry outputs with the anticipated measurement ranges for each 0 to 3 vdc analog channel. Calibration curves for each telemetered parameter are presented in Figures 3.6-35 through 3.6-40. In Figure 3.6-35 the RF power output telemetry is presented. For the power output range of 38 dbm (6.3 W) to 40.4 dbm (11 W) the resolution will be about 0.14 db per telemetry step for temperature between 77 and 158 degrees Fahrenheit. During the flight the temperature indicated in the figure will not be known exactly but the converter temperature of Figure 3.6-39 is a reasonable approximation. Assuming

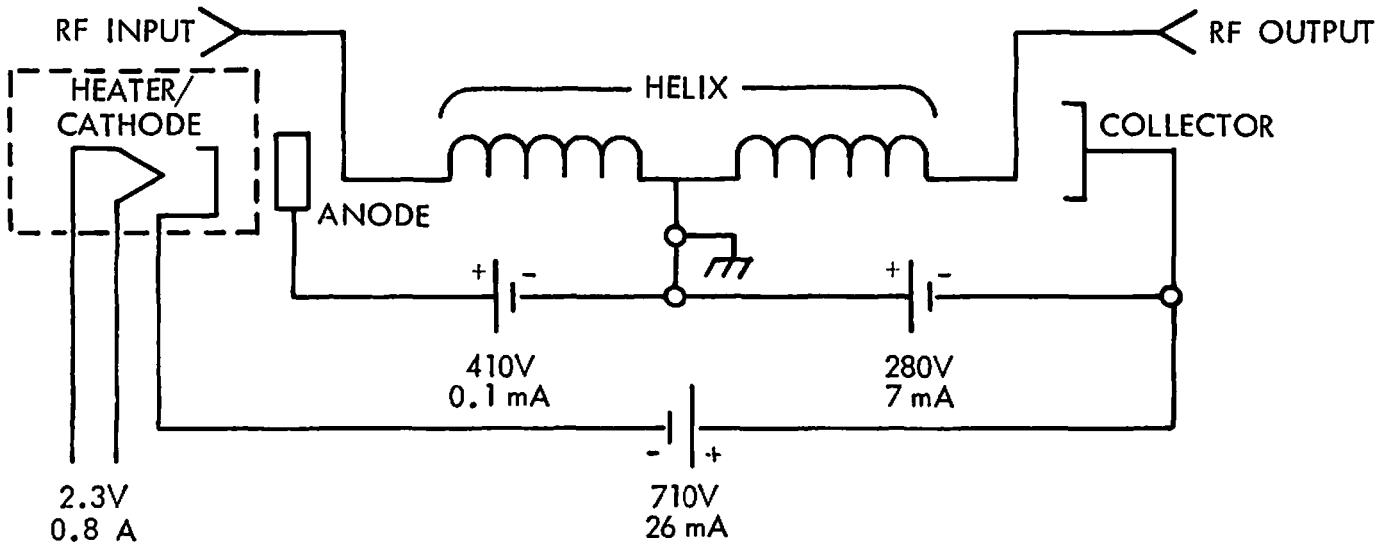


FIGURE 3.6 - 34. SIMPLIFIED TWT CIRCUIT

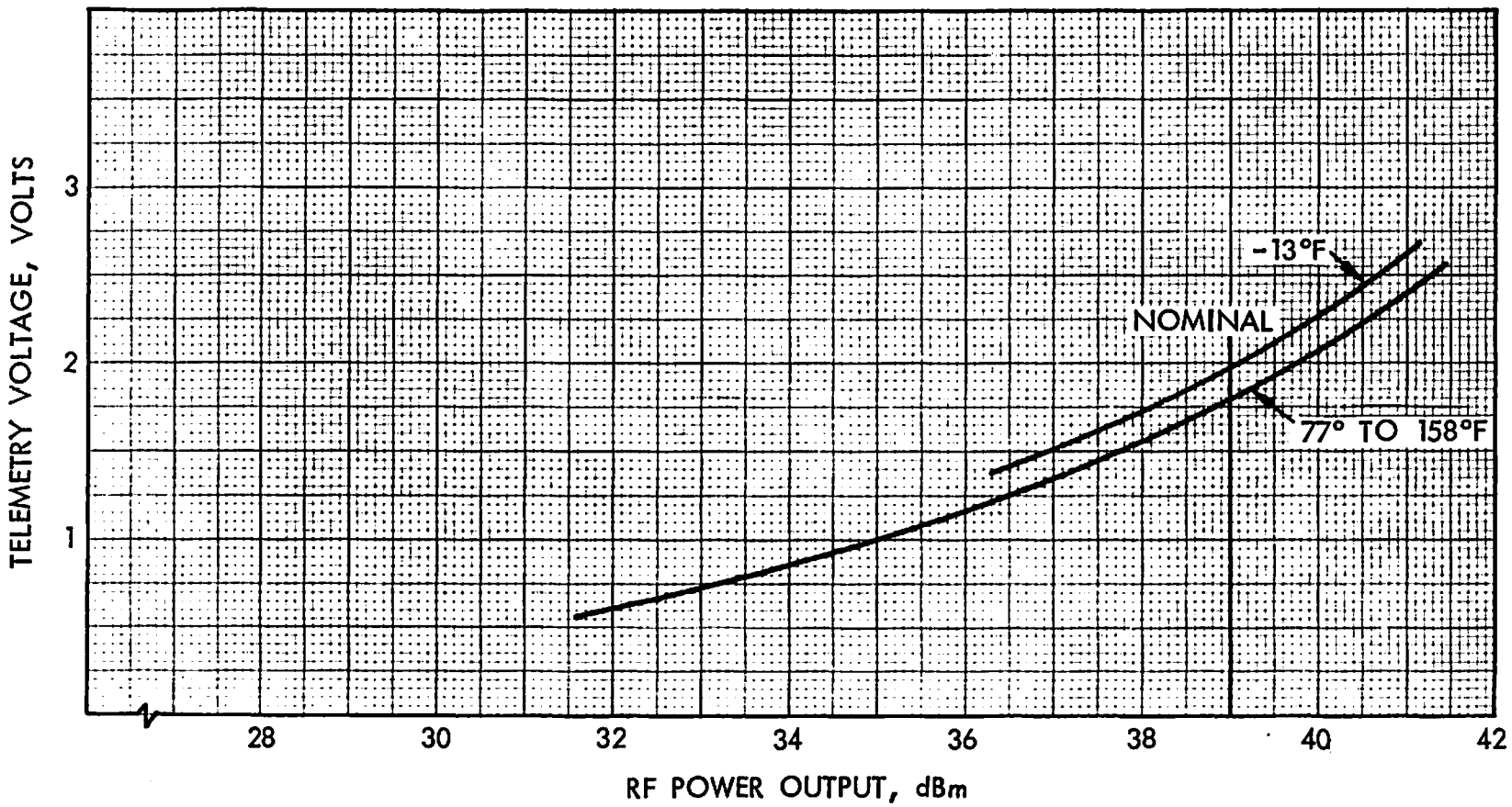


FIGURE 3.6 - 35. TELEMETRY VOLTAGE VERSUS RF POWER OUTPUT

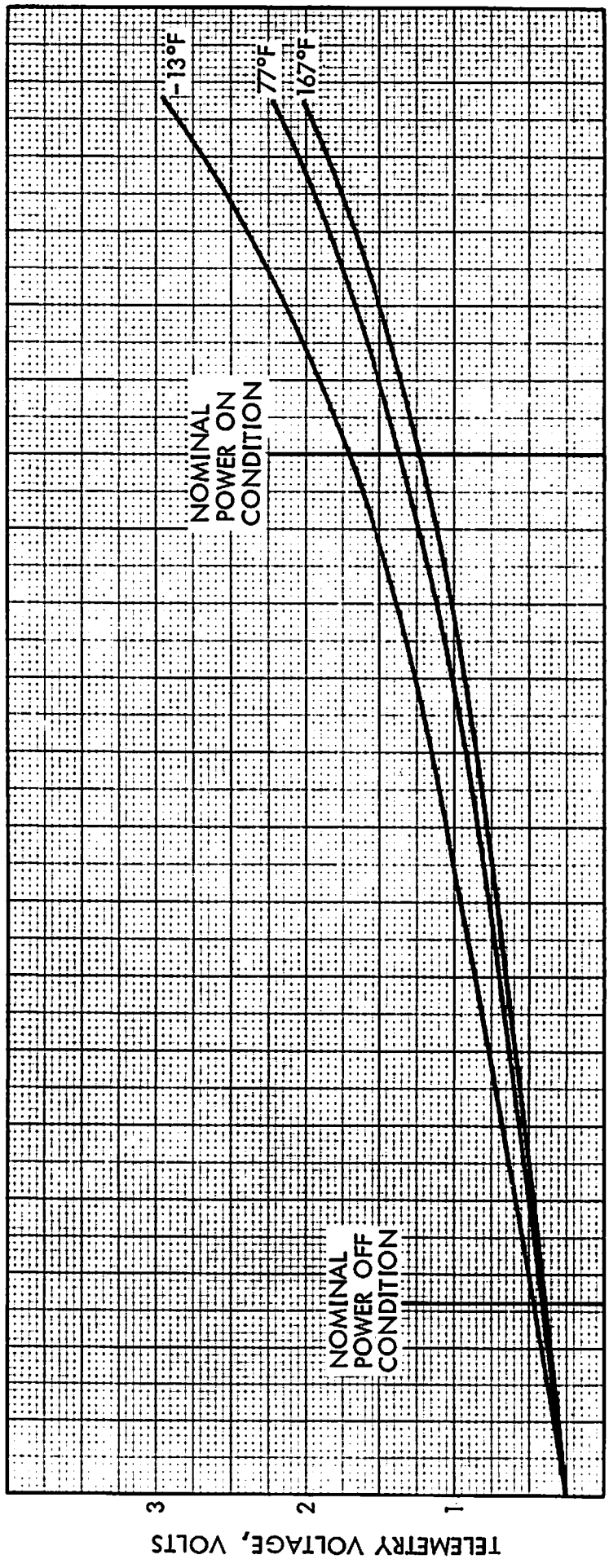


FIGURE 3.6 - 36. TELEMETRY VOLTAGE VERSUS HELIX CURRENT

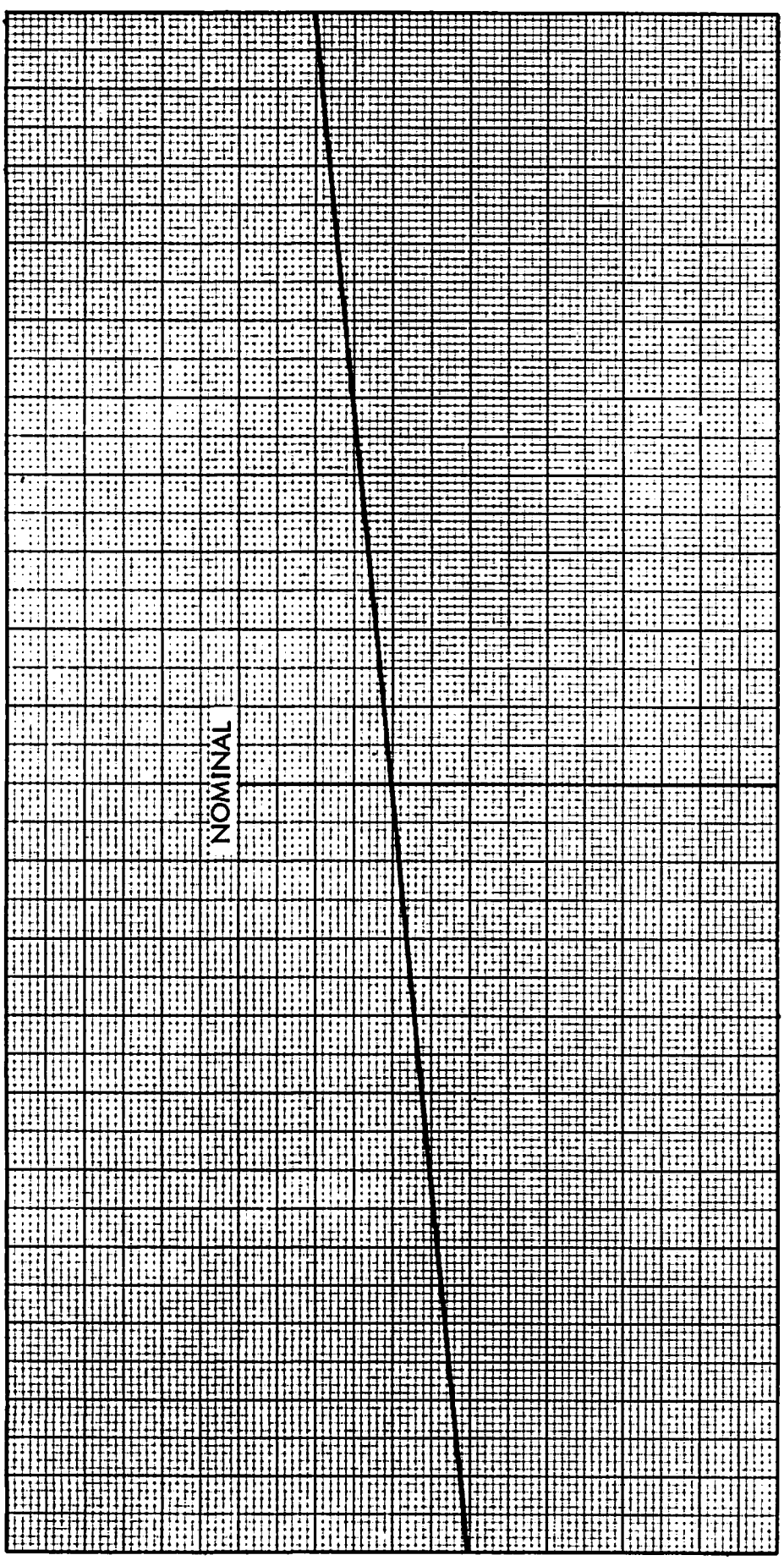


FIGURE 3.6 - 37. TELEMETRY VOLTAGE VERSUS CATHODE CURRENT

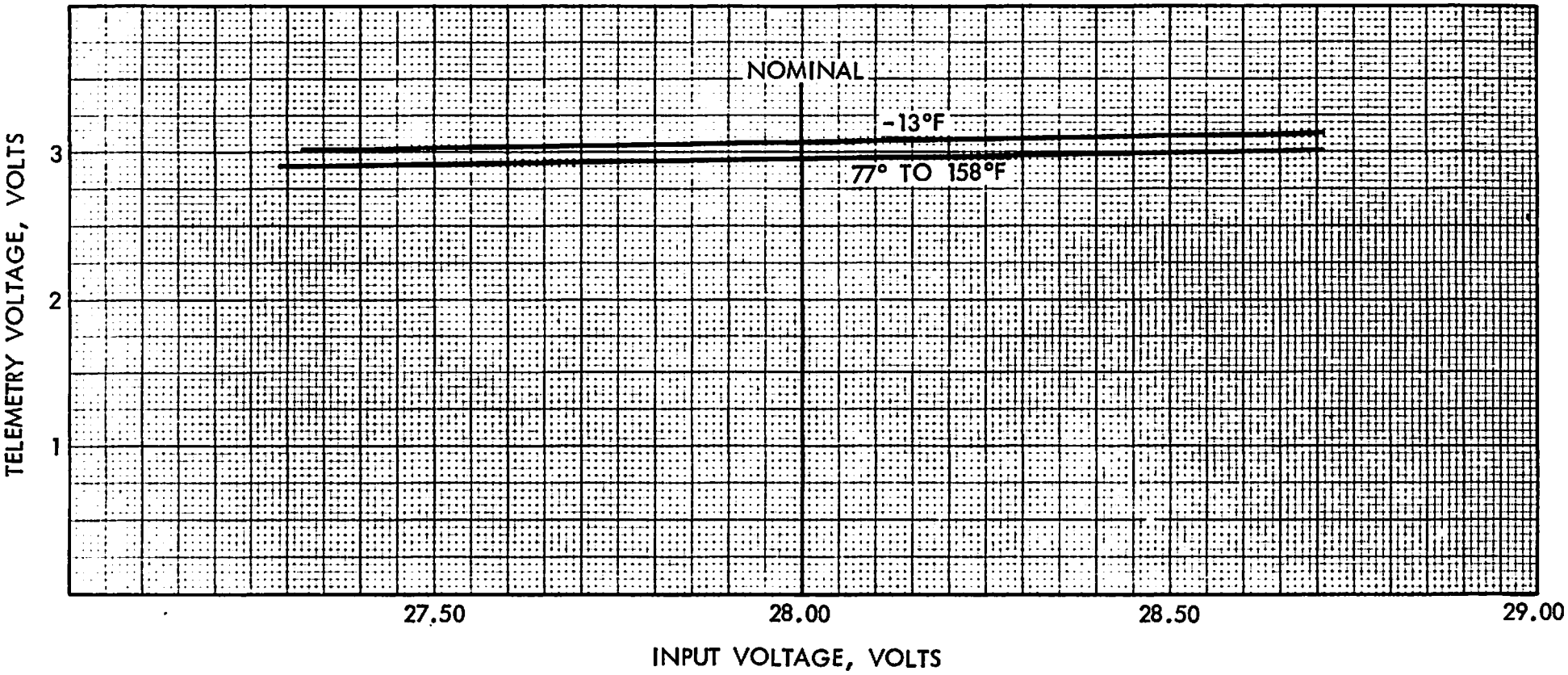


FIGURE 3.6 - 38. TELEMETRY VOLTAGE VERSUS INPUT (REFERENCE) VOLTAGE

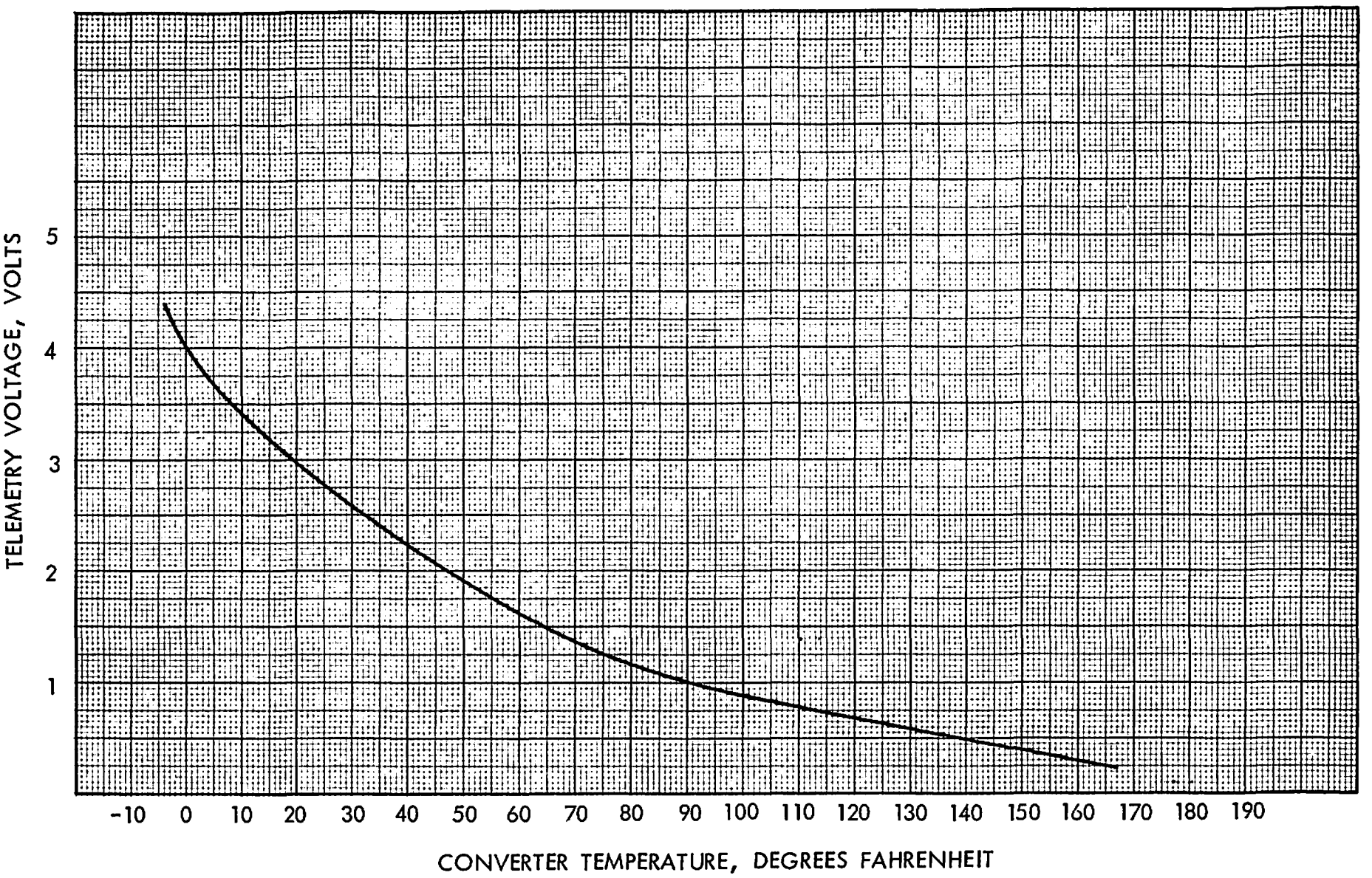


FIGURE 3.6 - 39. TELEMETRY VOLTAGE VERSUS CONVERTER TEMPERATURE

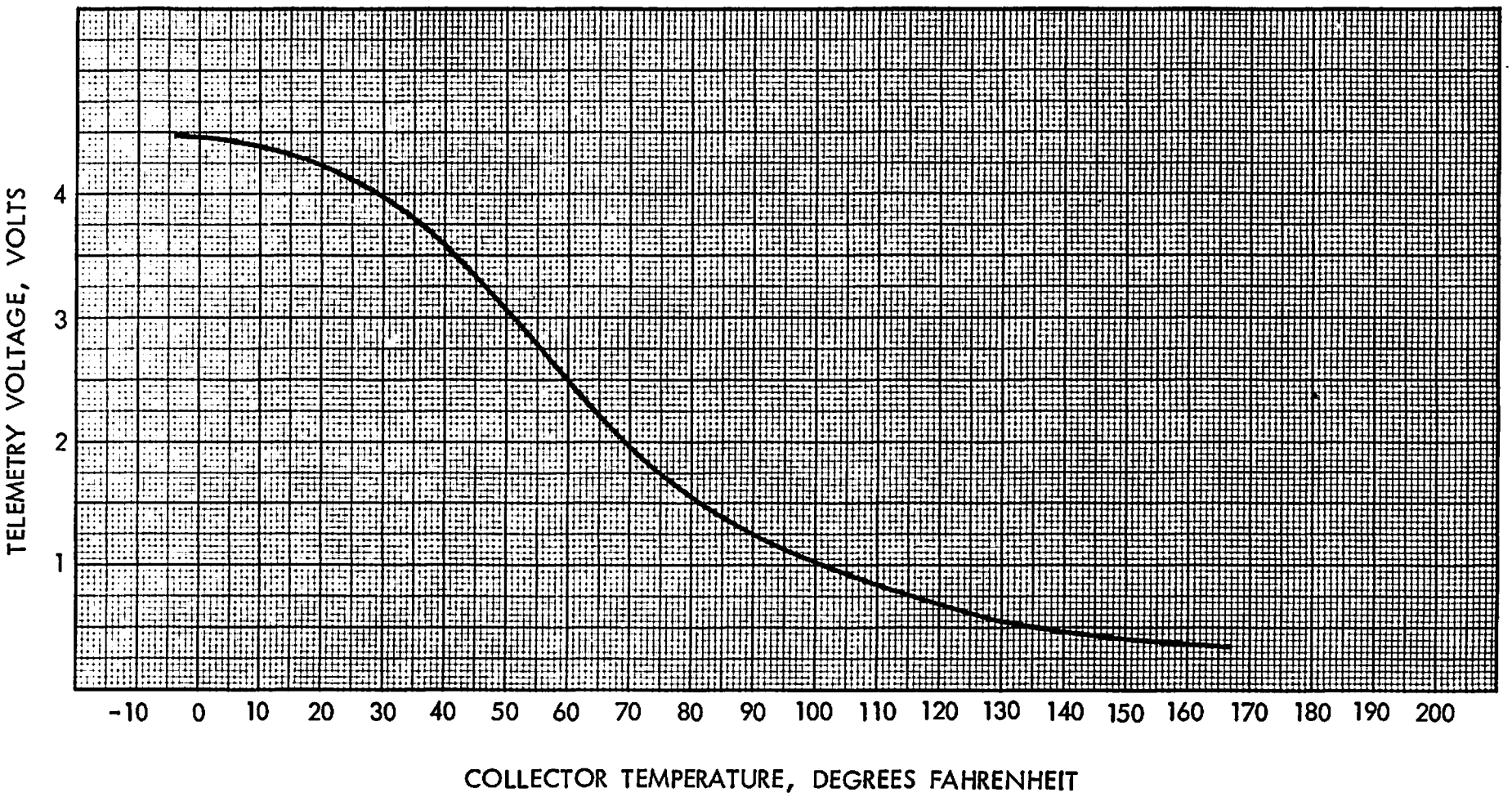


FIGURE 3.6 - 40. TELEMETRY VOLTAGE VERSUS TWT COLLECTOR TEMPERATURE

that the spacecraft antenna gain or the pointing is known, the DSS received carrier power using telemetered power output can be computed and compared with the received power level. The RF power output telemetry (with the other telemetered parameters) also gives an indication of the general health of the TWTA.

Table 3.6-6. TWTA Telemetry List

Word	Measurement		Measurement Range
C-231	Output Power Monitor	TWTA	26 - 40.4 dbm
C-214	Output Power Monitor	TWTB	26 - 40.4 dbm
C-211	Helix Current	TWTA	0 - 10 ma
C-216	Helix Current	TWTB	0 - 10 ma
C-206	Cathode Current	TWTA	24 - 30 ma
C-215	Cathode Current	TWTB	24 - 30 ma
C-224	Reference Voltage	TWTA	0 - 28 vdc
C-230	Reference Voltage	TWTB	0 - 28 vdc
C-207	Converter Temperature	TWTA	40 - 125°F
C-221	Converter Temperature	TWTB	40 - 125°F
C-205	TWT Temperature	TWTA	40 - 125°F
C-228	TWT Temperature	TWTB	40 - 125°F

Figure 3.6-36 depicts helix current versus telemetry voltage. Current telemetry resolution varies between 0.22 to 0.32 milliamperes per quantization step (over the range 1.3 to 12 milliampere) and is dependent on the converter temperature. By referring to the calibration curve of Figure 3.6-41, helix current can be used as an independent check on the RF power output. In addition, this figure shows that the TWTA RF input power can be estimated either from the helix current or the RF output.

Cathode current of Figure 3.6-37 is the sum of the TWT collector, helix and anode currents. Since the anode current is negligibly small, the collector current can be determined from the knowledge of cathode and helix currents. Collector current will vary somewhat with the 28 vdc line variation since normally the collector dissipates more power with the higher primary voltage than with the lower. The normal collector current range is estimated to be between 21 and 25 milliamperes.

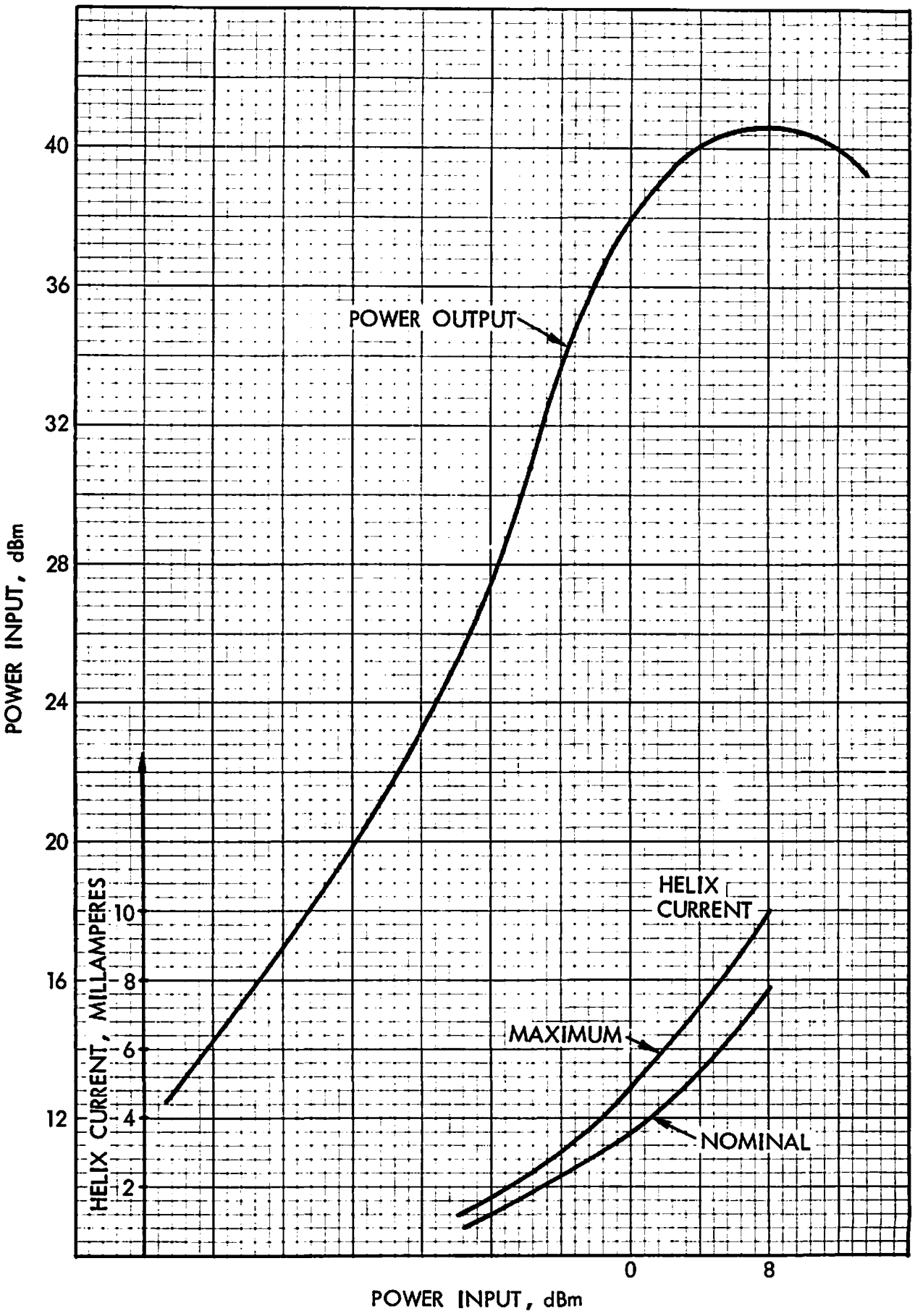


FIGURE 3.6 - 41. RF POWER OUTPUT AND HELIX CURRENT VERSUS INPUT POWER

The input or the reference voltage calibration curve is shown in Figure 3.6-38. The temperature indicated in this figure approximately corresponds to the telemetered converter temperature. Figure 3.6-38 provides an indication of the TWTA power consumption when used with the TWTA current versus voltage input data.

The converter temperature of Figure 3.6-39 is indicative of the component temperatures of the low voltage converter module (low voltage converter, power filter and telemetry circuits of Figure 3.6-33). It also provides an approximate temperature above the baseplate of the other components which are well separated from the relatively hot TWT collector. The TWT collector temperature versus telemetry voltage curve is presented in Figure 3.6-40. The acceptable collector temperature is below 185°F.

3.6.2.9 Conscan Signal Processor

3.6.2.9.1 Introduction. The receiver AGC output is a distorted sine wave conscan signal at the fundamental frequency of 0.08 Hz (corresponding to the spin rate of 4.8 RPM), with small amplitude wobble interference and wideband receiver noise. The purpose of the conscan signal processor (CSP) is to estimate the phase (or zero axis crossing time) and the amplitude which is proportional to the pointing error angle. The phase information is used to time the precession thruster firing so that the pointing angle between the spacecraft spin axis and the earth line-of-sight (pointing error) is reduced with the minimum expenditure of propellant. The amplitude information is used to terminate the thruster firing when the dead zone has been reached. The selected digital implementation of the CSP approximates the optimum analog phase and amplitude estimator shown in Figure 3.6-42. An important characteristic of this implementation is the need for an accurate frequency or spin rate reference which is available from the spin period sector generator (SPSG). The digital implementation of this estimator introduces an approximation of the amplitude A and phase θ estimates as shown in Figure 3.6-43 and provides a threshold setting accuracy of ± 6 percent and phase estimate accuracy of ± 2.5 degrees.

3.6.2.9.2 Unit Description. The detailed block diagram of the digital CSP is presented in Figure 3.6-44 and the major design parameters are summarized in Table 3.6-7. The waveforms and timing signals used in the CSP are illustrated in Figures 3.6-45 and 3.6-46 respectively. The incoming conscan signal from the receiver AGC has a phase angle, θ, with respect to the start of the unit integration period which begins 225 degrees after P_1. The SPSG pulse, P_1, is the CSP frequency reference signal. The first P_1 pulse initiates the integration period during which the conscan signal is sampled, quantized, multiplied by the in-phase and the quadrature square waves are integrated during two P_1 periods (two spacecraft revolutions) to obtain the values of $A\cos\theta$ and $A\sin\theta$.

When the integration period is completed the firing cycle is started. During the initial portion of the firing cycle, the amplitude A and $\tan\theta$ ($\cot\theta$) of the conscan sinusoid are sampled. The time required for the computation is about 7 ms, or less than 0.2 degrees of the conscan signal. The $\tan\theta$ computation takes place during the time frame D_0 and D_{14} while

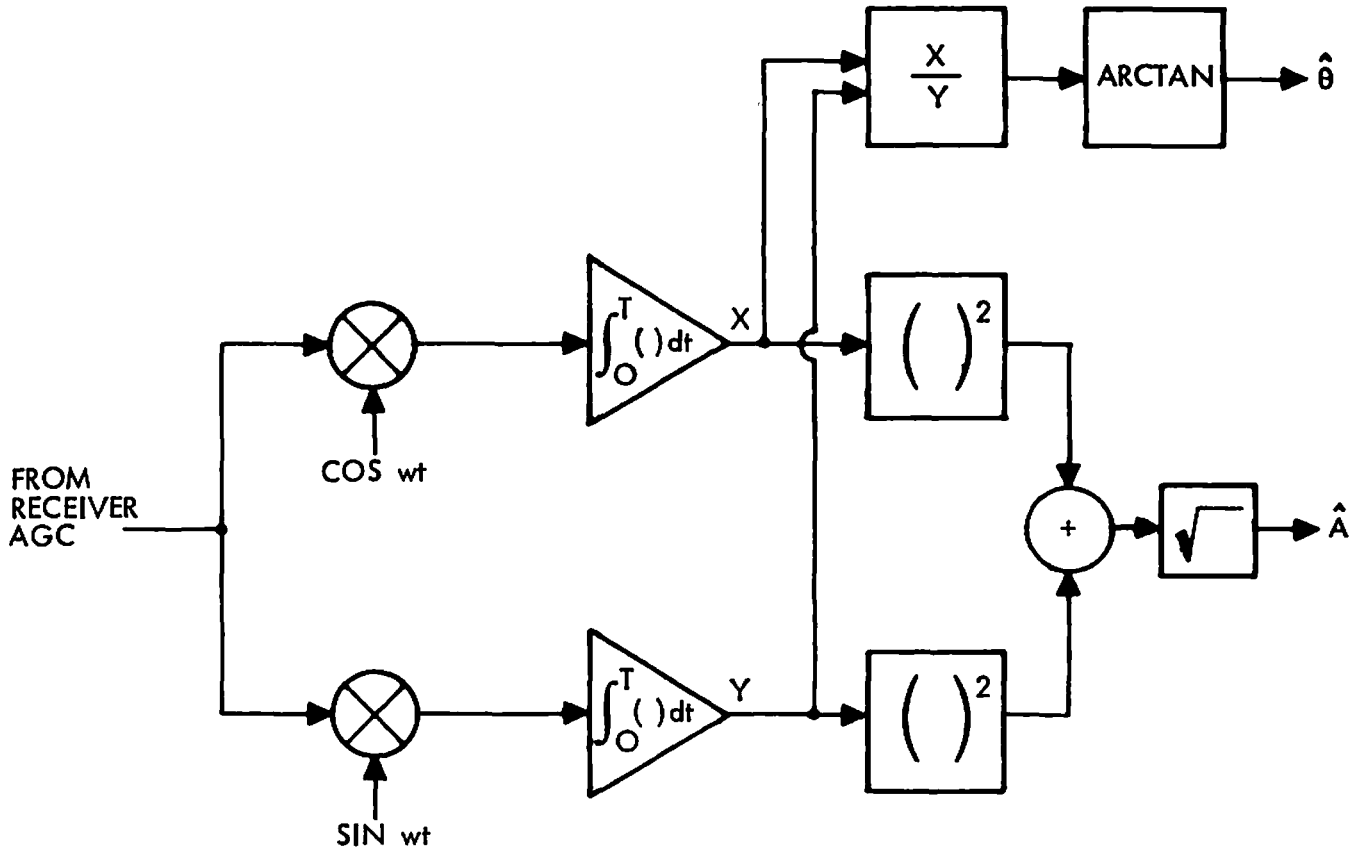


FIGURE 3.6 - 42. MAXIMUM LIKELIHOOD PHASE AND AMPLITUDE ESTIMATORS

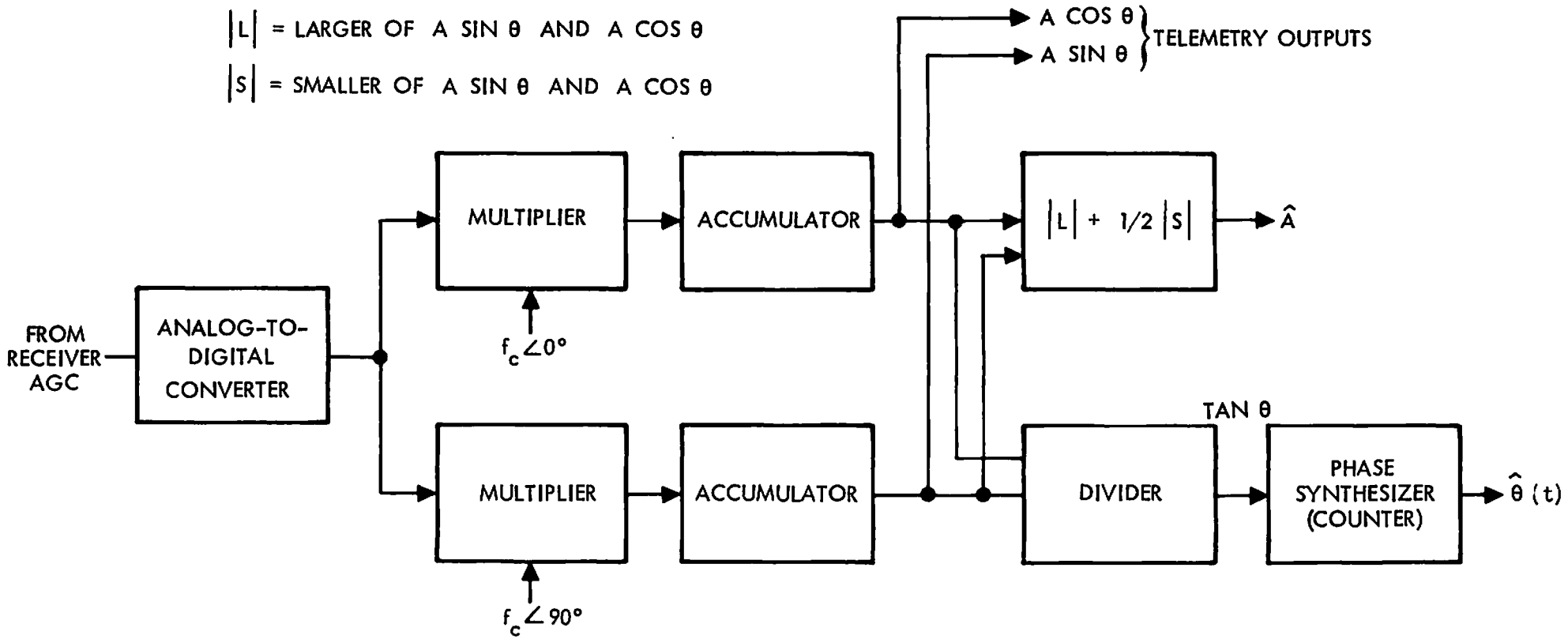


FIGURE 3.6 - 43. DIGITAL IMPLEMENTATION OF THE AMPLITUDE AND PHASE ESTIMATION PROCESS

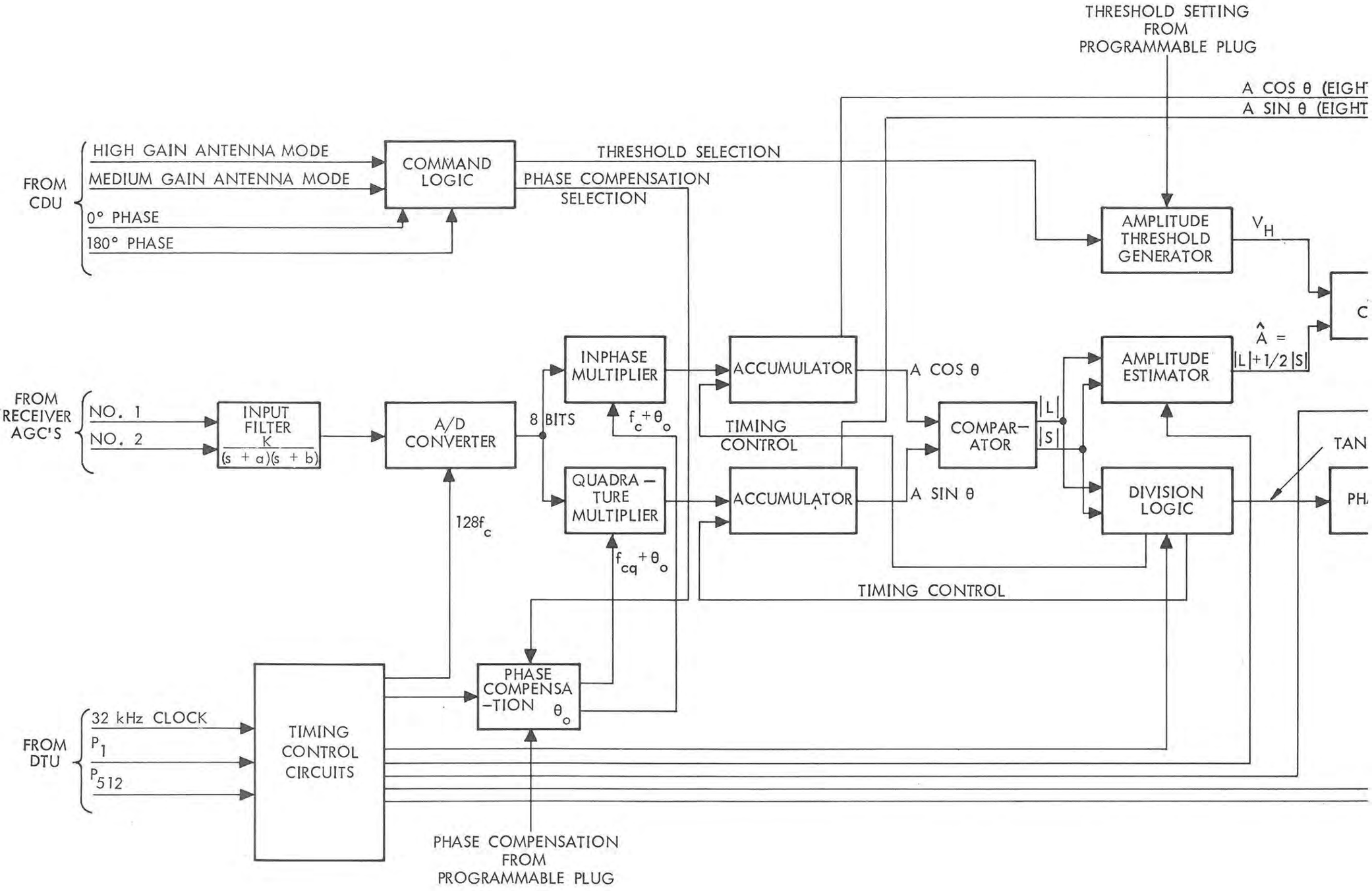
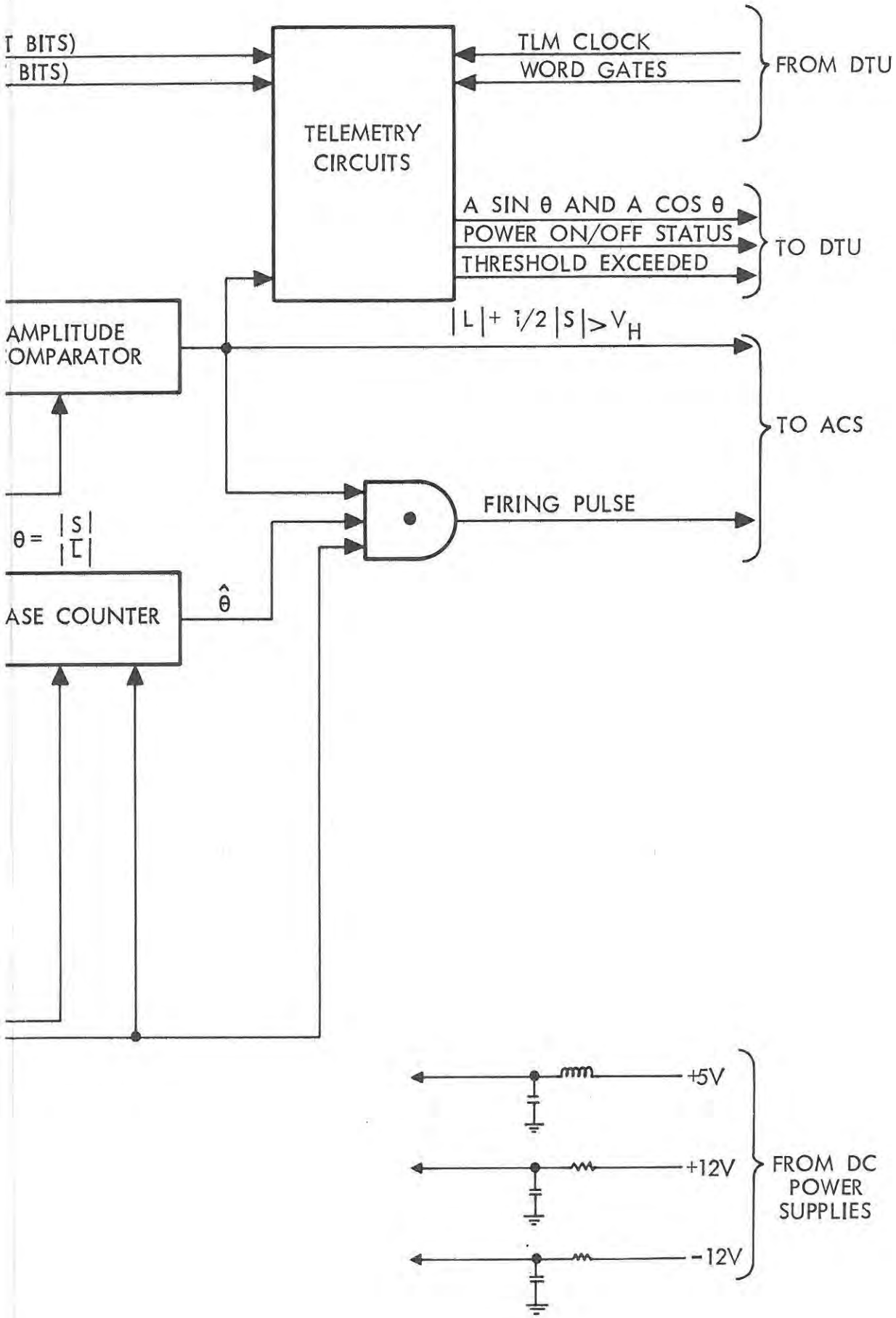


FIGURE 3.6 - 44. CONSCAN PROCESSOR BLOCK DIAGRAM


T BITS)
BITS)
TELEMETRY CIRCUITS
TLM CLOCK
WORD GATES
FROM DTU
A SIN θ AND A COS θ
POWER ON/OFF STATUS
THRESHOLD EXCEEDED
TO DTU
AMPLITUDE COMPARATOR
|L| + 1/2 |S| > V_H
TO ACS
FIRING PULSE
θ = |S| / |L|
ASE COUNTER
θ̂
+5V
+12V
-12V
FROM DC POWER SUPPLIES

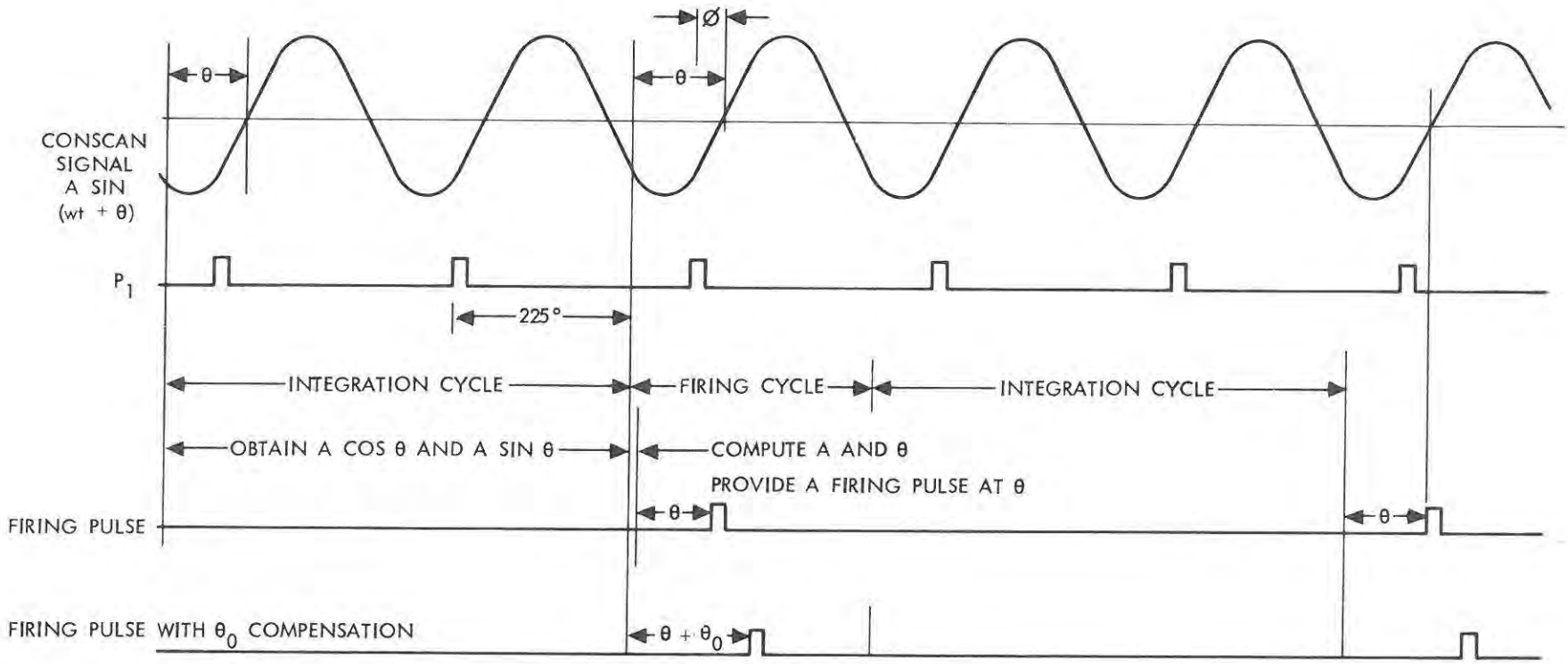


FIGURE 3.6 - 45. CONSCAN SIGNAL PROCESSOR FUNCTIONAL TIMING DIAGRAM

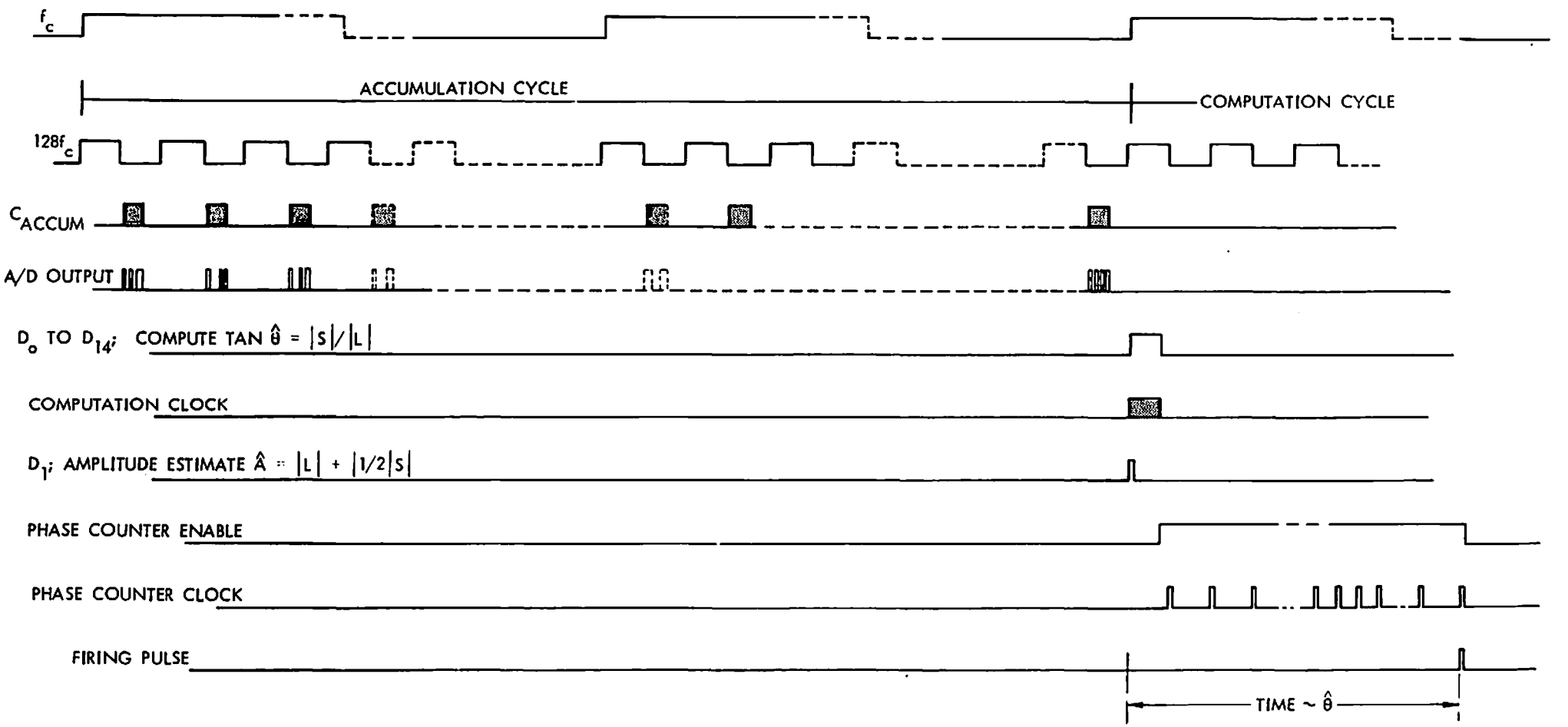


FIGURE 3.6 - 46. CONSCAN SIGNAL PROCESSOR CONTROL TIMING DIAGRAM

Table 3.6-7. Major Engineering Model CSP Parameters

1.	Integration Time	25 seconds (2 revolutions)	
2.	Threshold Setting	Corresponding to CSP Input in mv	Corresponding to Pointing Errors in Degrees
	a) Medium Gain Antenna	24.4 $\pm$ 0.2	1.3
	b) High Gain Antenna	72.1 $\pm$ 0.2	0.3
3.	Phase Compensation		
	a) Medium Gain Antenna	-1.4 degrees (lag)	
	b) High Gain Antenna	0 degrees	
4.	Limiting Level	0.53 volts peak at CSP input 2.1° for High Gain Antenna	

the amplitude is estimated and compared to the preset threshold during the time interval D_1. During the remainder of the firing cycle, the angle θ or the firing time is synthesized from $\tan\theta$ ($\cot\theta$). The phase θ measurement takes place in the phase counter which counts at a nonlinear rate such that an approximation to the tangent curve is generated. The output of the phase counter is compared with the counted value of $\tan\theta$ ($\cot\theta$) and a firing pulse is generated when the counter output is equal to the computed $\tan\theta$.

The conscan processor remains static after the firing pulse is generated and begins a new integration interval 225 degrees after the next P_1 pulse. Normally, a precession pulse is produced each 3rd revolution of the spacecraft.

The primary outputs from the CSP are:

1. Firing pulses to ACS control the time at which precession thrusters are fired. The thruster firing pulse may be selectively biased prior to flight using the CSP program plug so as to compensate for phase shift in the antenna, receiver, CSP, and 1/2 the thruster pulsewidth.

2. A firing enable signal to ACS indicates that the conscan signal amplitude is above the predicted threshold. When the amplitude drops below the predicted threshold level, the enable signal level switches to the false state and precession firing or conscan is terminated. If threshold is indicated even momentarily, conscan must be reenabled by command CNS5 in order to continue.

3. Telemetry of the Asinθ and Acosθ parameters are in the form of two 8 bit telemetry words. From these parameters and the calibration curve the pointing error angle magnitude and phase can be computed on the ground for the purpose of establishing spacecraft orientation. In addition, the parameters Asinθ and Acosθ can be used for partial checkout of the CSP before the closed loop operation is commenced.

3.6.2.9.3 Detailed Description. The input filter is an active bandpass filter centered around 0.08 Hz, with the upper 3 db corner frequency of about 1.7 Hz. The midband voltage gain is about 6. This filter decouples the high level DC component of the signal coming from the receiver AGC, and also limits the input noise bandwidth.

The analog to digital (A/D) converter uses the successive approximation technique to encode each sampled analog input voltage to an eight bit binary word. Thus, each voltage level at the sampling time is quantized into 256 levels with the quantum level size of about 4 millivolts. The sampling clock is at 128 f_c (see Figure 3.6-46 for definition of 128 f_c) rate and therefore 128 samples are taken every conscan cycle. The sampling rate reduces the quantization and sampling errors to negligible values. The highest quantum level determines the level at which limiting takes place and corresponds to about 0.53 volts peak at the CSP input.

The A/D converter output is multiplied by the inphase and quadrature components of f_c and shifted into two 16 bit accumulators by the internal shift clock C_{ACCM}. The square wave f_c is generated by the roll reference pulse P_1 and is at the conscan frequency. Since the output of the A/D converter is a binary word in the form of the sign -2's - complement, the multiplier is a "2's - complementor" which changes the incoming binary word into its complement, or allows it to feed through unchanged depending upon whether the multiplying signal is a "-1" for a "+1" (logic 0 or logic 1).

The accumulator is a digital integrator that integrates the multiplier outputs and provides the quantities Acosθ and Asinθ as outputs. The accumulation is delayed by 225 degrees relative to P_1 and lasts for two conscan cycles (nominally 25 seconds) so that 256 binary numbers are accumulated. The accumulation is accomplished by adding each number to the previously accumulated sum. A 16 bit register is used to store the accumulated value. The 16 bit words from the two accumulators, representing Asinθ and Acosθ are truncated to 8 bit words (the most significant bit is the sign). Upon receiving a word gate from the DTU the words are telemetered to the ground.

The two accumulator outputs, Asinθ and Acosθ are input into the comparator where the absolute value of the larger $|L|$ and the smaller $|S|$ are computed. The amplitude is estimated by the approximation $A = |L| + 1/2\ |S|$ and results in a peak computation error of about ± 6 percent.

The estimated amplitude is compared serially with a binary sequence that is related to the high or low threshold amplitude programmed at the programming plug. One of the two threshold levels is selected by a ground command. If the estimated amplitude exceeds the commanded threshold, the firing pulse is enabled and if the estimated amplitude drops below the selected threshold, conscan is terminated.

The division logic shares the same registers with the accumulator. These registers accumulate the digitized input signal to obtain the $A\cos\theta$ and $A\sin\theta$ during the accumulation cycle. The division logic computes the tangent or co-tangent of the phase angle of the signal by dividing the two accumulated values during the computation cycle.

The phase counter is a non-linear counter that provides a time equivalent to the computed phase by inverting the arc-tangent or arc-co-tangent of the output of the divider. It starts at the end of the division and generates a firing pulse to the ACS if the amplitude of the conscan exceeds the preset threshold. The firing pulse may be compensated (by means of the programming plug) such that a bias is added to the measured phase. The firing pulse always occurs during the cycle following the integration.

The timing control circuit receives a 32.768 KHz square wave from the DTU as the main clock. The roll reference pulses P_1 from the SPSG occur once every revolution and provide the spin rate (frequency) reference to the multipliers. The P_{512} signal provides 512 pulses for each P_1 and is used for timing purposes. The timing control function generates the necessary control logic to program the processing of the conscan signal. It also supplies timing pulses to digitize the input signal, integrate the digitized signal and to control the computation sequence and the comparators.

The phase compensation logic circuits generate the square wave multiplying signals. The multiplying signals have the same frequency as the P_1 pulses from the spin period sector generator, but may be selectively shifted in phase to compensate the output firing pulse for system phase biases. Two different phase compensation values programmed at the programming plug are available. One phase compensation value is required for the high gain antenna the other for the medium gain antenna. The phase compensation range for both antennas is ± 45 degrees and prior to flight the programmed phase compensation value is set in the program plug. In addition, the phase compensation logic circuits have the capability of introducing 180 degrees phase shift (independent of the programmed compensation). The 180 degrees and the zero degree phase shifts are selectable by ground commands and are used with the thruster pairs 2 & 3 and 1 & 4 respectively.

The telemetry circuits provide the processing status, in digital form, to the DTU that includes: bi-level power on and off status, $A\cos\theta$ and $A\sin\theta$, threshold mode status and thruster phase status.

Table 3.6-8. Telemetry and Command List

Telemetry Word	Bit Number	Function	Comments
C-313	1	Phase Status	State "0" for 0°
C-313	2	Antenna Mode	State "0" for medium gain antenna
C-313	3	A sin θ Bit 1	Least significant bit, A_1
C-313	4	A sin θ Bit 2	A_2
C-313	5	A sin θ Bit 3	A_3, TLM-A_i terms are equal to 0 or 1, the value of the
C-313	6	A sin θ Bit 4	A_4 corresponding bit or its
C-314	1	A sin θ Bit 5	A_5 inverse depending on sign
C-314	2	A sin θ Bit 6	A_6 bit
C-314	3	A sin θ Bit 7	Most significant bit, A_7
C-314	4	A sin θ Sign	State "0" for +
C-314	5	A cos θ Bit 1	Least significant bit, A_1
C-314	6	A cos θ Bit 2	$A_2{}^1$
C-315	1	A cos θ Bit 3	$A_3{}^1$
C-315	2	A cos θ Bit 4	$A_4{}^1$
C-315	3	A cos θ Bit 5	$A_5{}^1$
C-315	4	A cos θ Bit 6	$A_6{}^1$
C-315	5	A cos θ Bit 7	Most significant bit, A_7
C-315	6	A cos θ Sign	State "0" for +
C-316	1	CSP Status	State "0" for power off
C-316	2	Threshold	State "0" for signal below threshold

Commands			
Designator	Octal Code	Description	Comments
CNS0	133	CSP Power Off	Power gating takes place at TRF
CNS9	026	CSP Power On	
CNS1	234	High Gain Antenna Mode	High gain antenna threshold and phase compensation is selected.
CNS2	321	Med. Gain Antenna Mode	Med. gain antenna threshold and phase compensation is selected.
CNS3	007	0° Conscan Phase	Firing pulse is not shifted in phase.
CNS4	144	180° Conscan Phase	Firing pulse is shifted 180 degrees.

The command logic accepts conscan mode commands (high gain antenna mode or medium gain antenna mode) from the CDU that select threshold amplitude and phase compensation for operation with the high or medium gain antennas. Thruster phase commands from the CDU select an output firing pulse phase that is 0° or 180° from the compensation phase selected (by programming plug) with the threshold commands.

3.6.2.9.4 Operation. Normally when the conscan system is not operating, the power to the CSP is turned off by ground command CNS0 to the TRF of the Electrical Power subsystem (see Table 3.6-8). The fact that the CSP power is off is indicated by the "0" state of the first bit of the bilevel telemetry channel C-316. After the power is turned on by ground command CNS9, the first bit of channel C-316 switches to state "1" indicating that the +5 volts power has been applied. After threshold is exceeded the second bit (2) goes to state "1". Although warmup time is only about three minutes, it is advisable to partially verify that the CSP is operating properly before other commands are transmitted.

The partial inflight checkout of the CSP unit can be made by monitoring the telemetered parameters Acosθ and Asinθ on the ground. Positive indications that the conscan units including the CSP, the antenna, and receiver are operating nominally are:

1. The average amplitude A should be within the predicted range.

2. The variation of amplitude A from point-to-point should be within the predicted range.

3. The variation of phase θ from point-to-point should be within the predicted range.

4. If the CSP backup telemetry (conscan AGC from the receiver) is being processed, the conscan signal amplitude and phase computed from the two separate telemetry channels should be in reasonable agreement.

5. If the conscan signal amplitude and phase are being computed on the downlink at the DSS, their values should be in agreement with A and θ.

After partial verification that the CSP operation is within the nominal range, commands can be transmitted to select the medium gain or the high gain antenna conscan mode. Command CNS2 selects the medium gain antenna threshold and the ground programmed phase compensation and command CNS1 selects the high gain antenna threshold and phase compensation. To provide conscan system operational capability with either thruster pair, command CNS3 selects 0 degrees conscan output phase to be used with the thruster pairs 1 and 4 and command CNS4 will select 180 degrees conscan output phase to be used with thruster pair 2 and 3.

3.6.2.9.5 Calibration of Pointing Error Angle. The CSP calibration rule relates telemetered amplitude A and phase θ to the amplitude and phase of the conscan signal at the input to the CSP. Combining this with the antenna calibration curves (Figures 3.6-14, 15, 18 and 19) and the receiver AGC slope (Figure 3.6-29), A and θ can be related to the pointing error angle magnitude and phase. In addition, significant bias errors that can be compensated on the ground are also described.

The parameters Asinθ and Acosθ are telemetered, least significant bit first, as digital eight bit words are defined in Table 3.6-8. The last bit (MSB) of each word is the sign and state "1" (minus sign) requires that each bit of such a word be complemented. (2's complement, therefore, add one to the least significant bit after taking the complement.)

The peak conscan signal amplitude at the CSP input, V, in volts is given by

$$V = 1.0603 \sqrt{V_s^2 + V_c^2}$$

V_s = input corresponding to telemetered |Asinθ|, volts

$$= K \sum_{i=1}^{7} 2^{(i-7)} A_i \qquad A_i = \text{"0" or "1"}$$

V_c = input corresponding to telemetered |Acosθ|

$$= K \sum_{i=1}^{7} 2^{(i-7)} A_i \qquad A_i = \text{"0" or "1"}$$

K = 0.4 volts for Engineering Model (Specific value for each flight unit is supplied as part of the factory calibration data)

= peak conscan signal amplitude at CSP input corresponding to the most significant bit of Asinθ and Acosθ.

$2^{(i-7)}$ The A_i's are the various bit values as given in Table 3.6-8, and represents the binary value of the bit divided by 128. In words, V_S and V_C are found by summing the weights of the telemetered bits corresponding to state "1" and multiplying by the calibration constant K. Since the calibration constant K is measured for the threshold setting and includes 6.03 percent computation error, the RMS value of V_S and V_C have to be multiplied by the factor 1.0603 on the ground.

In the first quadrant the conscan signal phase at the input to the CSP, θ, is given by

$$\theta = \arctan\left(\frac{V_s}{V_c}\right) \pm \theta_o, \text{ degrees}$$

where θ_0 is the phase compensation in degrees and the conscan phase is 0 degrees (command CNS3). In other quadrants θ is determined from the signs of Asinθ and Acosθ using conventional trigonometry rules. Angle θ is measured relative to the start of integration time which is delayed by 225 degrees relative to the roll reference pulse P_1. The phase $\emptyset$ of the conscan signal relative to the P_1 pulse is given by

$$\emptyset = 225 + \theta$$

where positive $\emptyset$ indicates time delay relative to P_1.

The parameters Asinθ and Acosθ are stored in the shift registers as 16 bit digital words. Since the telemetry words are only eight bits long, a truncation error results. The maximum truncation errors for both Asinθ and Acosθ are equal to the weight of the least significant bit and are always negative

$$\Delta V = - 2^{-7} \left(\frac{2K}{2V_i}\right) = - \frac{\left(\frac{2K}{2V_i}\right)}{128}$$

by adding

$$1/2\ /\Delta V/ = \frac{\left(\frac{2K}{2V_i}\right)}{256}$$

to V_S and V_C the maximum truncation error is converted from $- \frac{\left(\frac{2K}{2V_i}\right)}{128}$ to $\pm \frac{\left(\frac{2K}{2V_i}\right)}{256}$.

Other bias phase errors that can be compensated are summarized in Table 3.6-9.

Example of Conscan TLM Reduction. In Table 3.6-10, an example of the Asinθ - Acosθ telemetry is given.

Table 3.6-10. Example TLM Reduction for Conscan

Word	Bit	Definition	TLM	$A_i \times (2^{i-7})$
C-313	3	i = 1, LSB	0	0
C-313	4	i = 2	1	1 x (4/128)
C-313	5	i = 3	1	1 x (8/128)
C-313	6	i = 4	0	0
C-314	1	i = 5	1	1 x (32/128)
C-314	2	i = 6	0	0
C-314	3	i = 7, MSB	1	1 x (128/128)
C-314	4	Asinθ; sign bit	0	+ sign
C-314	5	i = 1, LSB	0	0
C-314	6	i = 2	1	1 x (4/128)
C-315	1	i = 3	0	1 x (8/128)
C-315	2	i = 4	1	0
C-315	3	i = 5	1	0
C-315	4	i = 6	0	1 x (64/128)
C-315	5	i = 7, MSB	1	0
C-315	6	Acosθ; sign bit	1	- sign

To compute V_S and V_C, the terms in the last column of Table 3.6-10 are summed. The A_i's are the TLM bit values (2's complement for a minus sign) and the term 2^{i-7} is equivalent to $2^i/128$.

$$\sum_{i=1}^{7} A_i(2^{i-7}) = 172/128 = \frac{V_S}{K}$$

$$\sum_{i=1}^{7} A_i\,(2^{i-7}) = -76/128 = \frac{V_C}{K}$$

Therefore,

$$V = 1.063\left[\left(172\right)^2 + \left(76\right)^2\right]^{1/2}\left(\frac{0.4}{128}\right) = 0.623 \text{ volts}$$

The AGC conscan slope is about 0.096 v/db, therefore;

$$A = \frac{0.623 \text{ v}}{0.096 \text{ v/db}} = 6.5 \text{ db}$$

On the high gain antenna, the pointing error is 2.35° as found in Figure 3.6-14. Also,

$$\theta = \tan^{-1}\left(\frac{172}{-76}\right) = 114.9°$$

The phase relative to the roll pulse is therefore 339.9°.

Table 3.6-9. Bias Phase Errors

Type of Error		Error in Degrees	Comments
1. Receiver AGC and CSP Filter Response		1.9	These errors will be compensated for in the CSP and will not have to be corrected on the ground. The net compensation, θ_0, (error No. 1 + error No. 2) is 0° (-0.1°) for the high gain antenna -1.4° (-1.45°) for the medium gain antenna.
2. Control Electronics Assembly (CEA)	125 Millisecond Pulse	-0.45	
	31.25 Millisecond Pulse	-1.8	
3. Spin Axis Shift Due to Fuel Consumption	Medium Gain Antenna	Arctan $\frac{\Delta}{9.3}$	Δ = spin axis shift in degrees from the nominal position due to fuel consumption. Spin axis shift is in the XZ plane and therefore has a negligible contribution to amplitude error.
	High Gain Antenna	Arctan Δ	
4. Antenna Phase Variation	Medium Gain Antenna	Figure 3.6-19	It should be cautioned that part of the phase variation in Figures 3.6-15 and 3.6-19 might be due to antenna range errors and not to the antenna pattern characteristics.
	High Gain Antenna	Figure 3.6-15	

SECTION 3.6 NOTATION

A	Amplitude
AC	Alternating Current
ACS	Attitude Control System
A/D	Analog to Digital
AGC	Automatic Gain Control
AU	Astronomical Unit $\sim 150 \times 10^6$ km
bps	Bits per second
CDU	Command Distribution Unit (Described in Section 3.9)
CSP	Conscan Signal Processor
dB	decibels
dbi	db relative to isotropic antenna
dbm	db in milliwatts
DC	Direct Current
DDU	Digital Decoder Unit
$\Delta\phi$	phase error
DSS	Deep Space Stations
DTU	Digital Telemetry Unit (Described in Section 3.5)
D_1	Time interval of amplitude computation
ϵ	pointing error
ERP	Effective Radiated Power
f	Frequency
f_c	Computation cycle period; 1/2 accumulation cycle period
F_o	Rest Frequency, 2110.584105 MHz
FSK	Frequency Shift Key
Hz	Hertz, cycles per second
IF	Intermediate Frequency
KHz	KiloHertz, cycles per second
Km	Kilometers
kW	kilowatt
$\|L\|$	Larger of A sin θ, A cos θ
LSB	Least Significant Bit
mA, ma	milliamp
MHz	MegaHertz

SECTION 3.6 NOTATION (CONT'D)

ms	millisecond
MSB	Most Significant Bit
mV	millivolts
PCM	Pulse Code Modulation
PLL	Phase Lock Loop
PM	Pulse Modulation
ppr	pulses per revolution of spacecraft
PSK	Phase Shift Key
P_1	P_r, roll pulse from SPSG (1 ppr)
P_{512}	512 ppr from SPSG
$\emptyset$	$225° + \theta$
RCVR	Receiver
RF	Radio Frequency
rpm	revolutions per minute
$\|S\|$	Smaller of $A \sin \theta$, $A \cos \theta$
SPSG	Spin Period Sector Generator
θ	phase angle from start of integration period ($P_1 + 225°$)
$\hat{\theta}$	angle from arc tan phase synthesizer $\|S\|/\|L\|$
θ_o	Phase compensation in program plug
TCXO	Temperature Compensated Crystal Oscillator
TLM	Telemetry
TRF	Transformer-Rectifier-Filter
TWTA	Travelling Wave Tube Amplifier
VCO	Voltage Controlled Oscillator
VSWR	Voltage Standing Wave Ratio
W	Watt

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3.7 ELECTRICAL POWER SUBSYSTEM

This section describes the Electrical Power Subsystem functions, capabilities and performance. It is broken into three major sections which are; Subsystem Description, Detailed Unit Description, and Management of the Subsystem during Normal and Failure Operational Modes.

3.7.1 Power Subsystem Description

3.7.1.1 Functional Definition. The electrical power subsystem consists of two inverter assemblies (IAs), a power control unit (PCU), a silver cadmium battery, a central transformer rectifier filter (CTRF) and a shunt radiator. A simplified block diagram of the subystem is shown in Figure 3.7-1. The power subsystem units are functionally described below.

- Inverter Assemblies (IA). The IAs process the low voltage DC output of the RTGs and produce at their output an AC trapezoidal waveform of 61.0 vrms (line to line). The IAs contain fault isolation circuits which protect the spacecraft against failures of RTGs, RTG cables or internal inverter faults. The IAs condition RTG temperature telemetry data. The IA outputs are cross-strapped in the PCU to form the AC bus.

- Power Control Units (PCU). The PCU transforms and rectifies the AC bus into a regulated DC bus. AC bus regulation is derived as a direct result of DC bus regulation in the PCU. The PCU provides battery charge and discharge control and subsystem telemetry conditioning. The RTGs are operated at nearly maximum power throughout the mission by the PCU. Command control of power subsystem functions is provided in this unit.

- Battery. The battery consists of eight 5 Amp-hour silver-cadmium cells connected in series. Each cell is protected against overcharge, overdischarge, and open circuits by cell bypass electronics. The battery supplies pulse loads in excess of RTG capability and may be used for load sharing with the RTGs.

- Central Transformer-Rectifier-Filter (CTRF). The CTRF contains separate transformer-rectifier-filters which operate directly from the AC bus. Each TRF output is regulated either by series dissipative or switching regulators to obtain the multiple regulated DC voltage outputs required by the user subsystems.

- Shunt Radiator. A shunt radiator, consisting of strip resistors mounted on a flat plate, dissipates RTG power in excess of load demand.

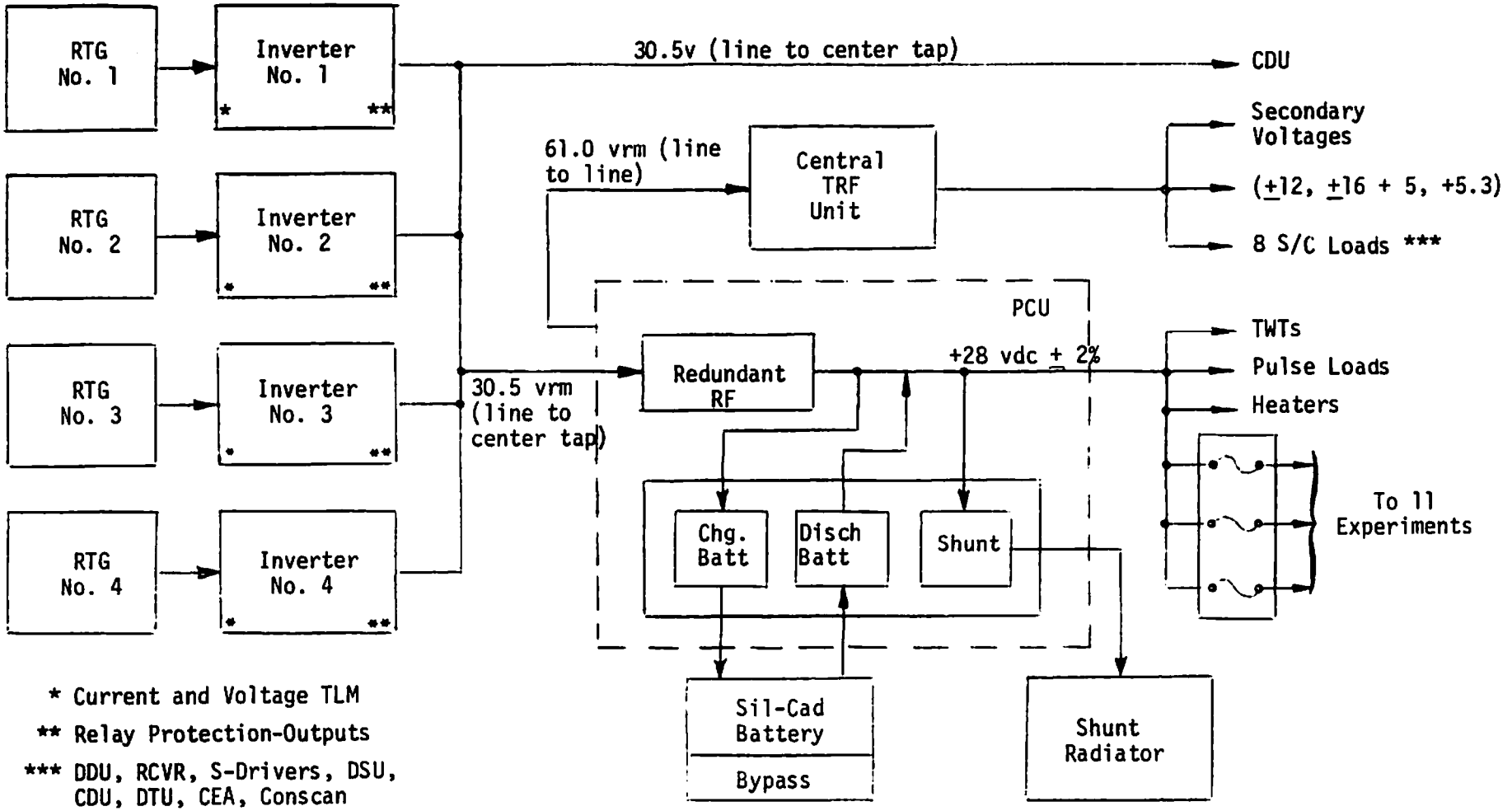


Figure 3.7-1. Pioneer F/G Power Subsystem Block Diagram

Section 3.1 provides detailed drawings of the spacecraft. The IAs are mounted on opposite sides of the spacecraft as close as possible to the RTGs to minimize cable length and losses. The CTRF, PCU and battery are grouped for the same reason. The battery is mounted on an exterior panel to facilitate thermal control implementation. The shunt radiator is located on an exterior panel to minimize impact on the spacecraft thermal design since its power dissipation will vary widely during the mission. A summary of the major power subsystem parameters is presented in Table 3.7-1.

Table 3.7-1. Major Power Subsystem Parameters

RTGs	
RTG Operating Voltage	4.2 vdc ± 7%
RTG Combined Power at Acceptance	175 watt (maximum on S/C)
RTG Power (ea) at Launch	39.2 ± 3 watt
RTG Power (ea) at Three Years	35.0 ± 2 watt (maximum) 32.9 ± 1 watt (minimum)
INVERTERS	
Output Voltage	61.0 vrm ± 3% (line to line)
PCU	
Output Voltage	28 vdc ± 1% (±2% including long term drift) and 61.0 vrm ± 3% (line to line)
Undervoltage Level	26.0 ± 0.5 vdc
Shunt Radiator Power Dissipation	103 watts (maximum)
Shunt Regulator Power Dissipation (inside PCU)	40 watts (maximum)
Shunt Power Capability	118.5 watt (maximum)
BATTERY	
Discharge Current(at 28 v bus)	1.0 amps (maximum)
Discharge Current at Battery	10.0 amps
Charge Current	300 milliamps (maximum)
Capacity	5 amp-hr; 40 wt-hr

The power subsystem operation is characterized by inversion of the low voltage DC power from the RTGs to high voltage AC. The AC from the inverters is sent to the PCU for distribution to the CDU and CTRF. The CDU transforms the AC to high voltage DC which charges storage capacitors for squib firing. The CTRF uses the AC to generate regulated DC voltages for other spacecraft users. Certain CTRF loads are commandable ON/OFF,

others are always ON, while the ACS has automatic sense and switch-over (Section 3.7.2.3 provides details on each CTRF output). The remainder of the inverter AC output is rectified, filtered and regulated in the PCU to form a +28 vdc bus.

3.7.1.2 Power Subsystem Capability. The electrical power subsystem capability can be evaluated based upon its performance in several areas. These can be defined as the capability to:

1) Control and regulate RTG power output during the mission.

2) Support the spacecraft load.

3) Perform battery load sharing.

4) Supply power with a minimum of losses.

3.7.1.2.1 Control and Regulation of RTG Power Output. The power subsystem controls and regulates the RTG power by means of a shunt regulator. The apportionment of the 28 vdc bus $\pm$ 2% regulation band is shown in Figure 3.7-2. The worst case variation in bus voltage prior to launch is $\pm$ 1%. After 2.5 years operation in space, the ageing of components could result in a long term drift of $\pm$ 2%. The nominal voltage levels at which mode changes between shunting, charging, and discharging occur are also shown in the insert.

Note that the variation in bus voltage required to change from the discharge to shunting modes is only about 2 millivolts. Guard bands between each level together with filtering in the PCU control electronics prevent switching between discharge, charge, and shunting due to noise on the bus.

The $\pm$ 2% vdc bus tolerance is reflected back to the output of the inverters through the PCU TRF (see Figure 3.7-3). Additional losses incurred in the TRF produce an AC bus voltage regulation tolerance of $\pm$ 3%. This AC bus tolerance results in a $\pm$ 7% RTG output voltage variation when internal inverter and RTG cable losses are considered (see Figure 3.7-4).

3.7.1.2.2 Support Spacecraft Load. The capability of the power subsystem to support the spacecraft load is dependent upon the RTG power output degradation rate. Figure 3.7-5 shows the total RTG power output for 3 and 4 RTGs as a function of mission time. The spacecraft power demand during cruise and commanding phases of the mission are superimposed on the graph. Note that even with 3 RTGs operative sufficient power is available to support the cruise load at encounter. With 4 RTGs the cruise load can be supported well into the post encounter phase for times which exceed 5 years.

When commanding, the spacecraft the load will rise to 105.5 watts. With three RTGs operative, the spacecraft will process commands with no

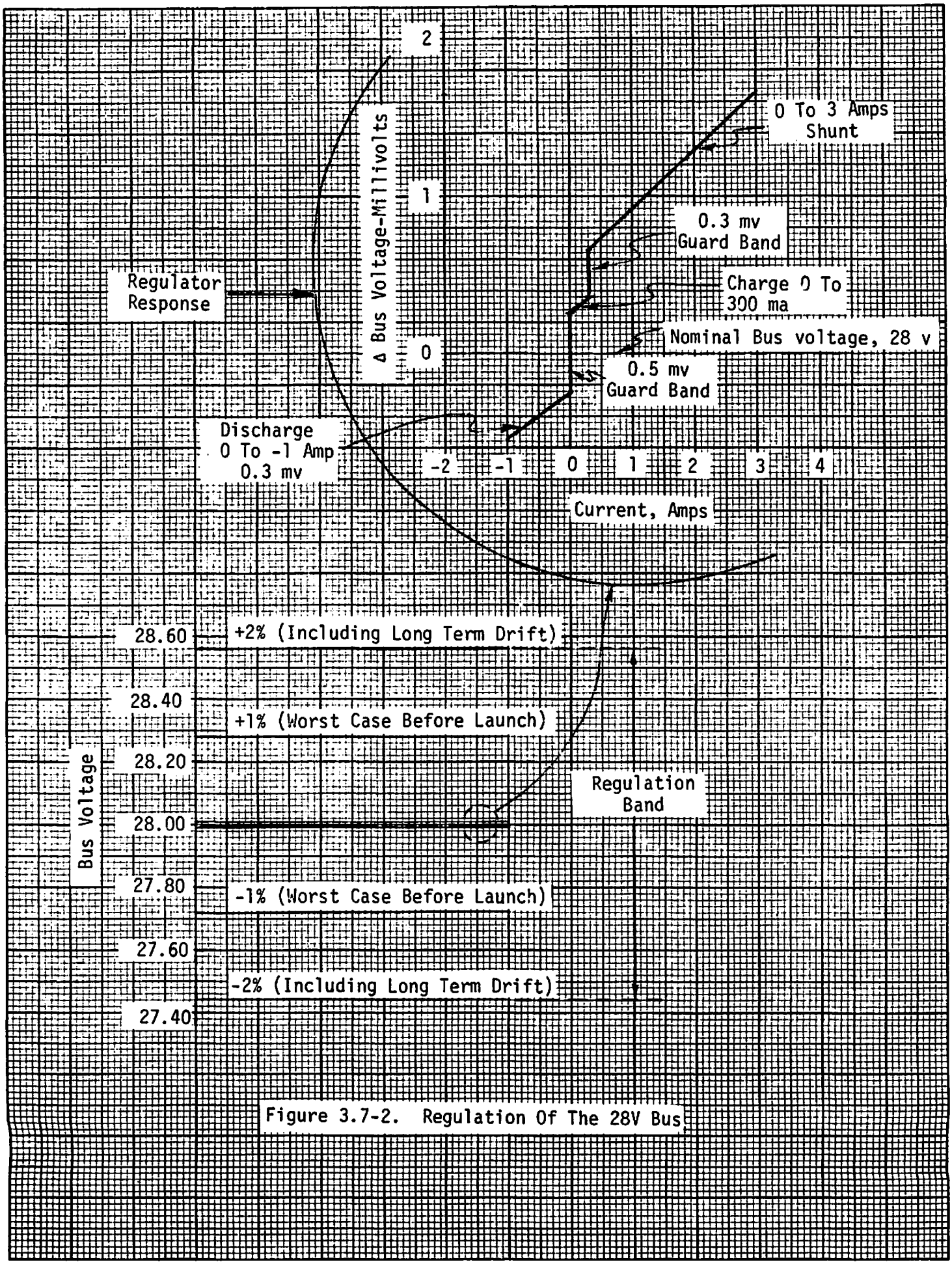


Figure 3.7-2. Regulation Of The 28V Bus

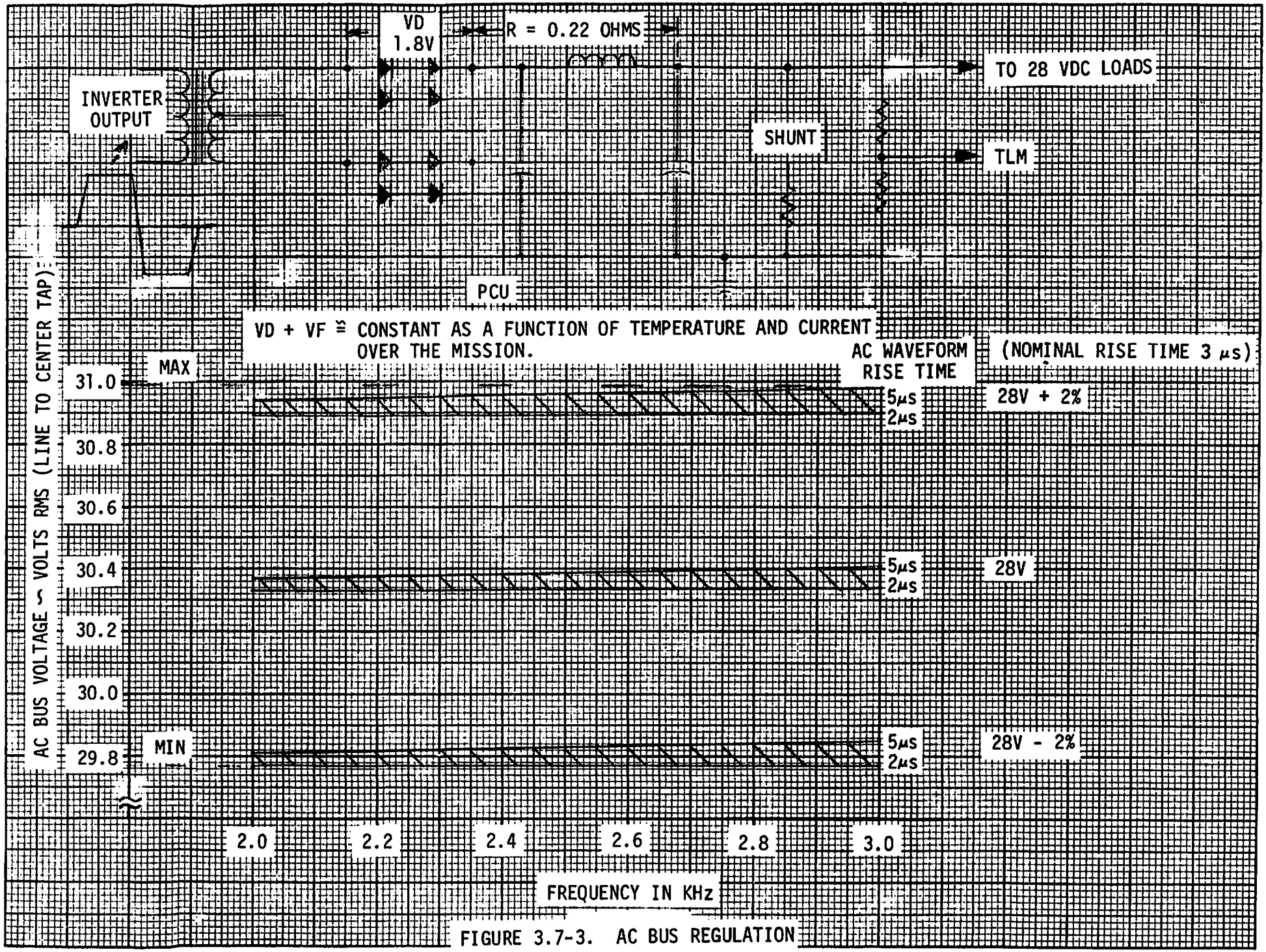


FIGURE 3.7-3. AC BUS REGULATION

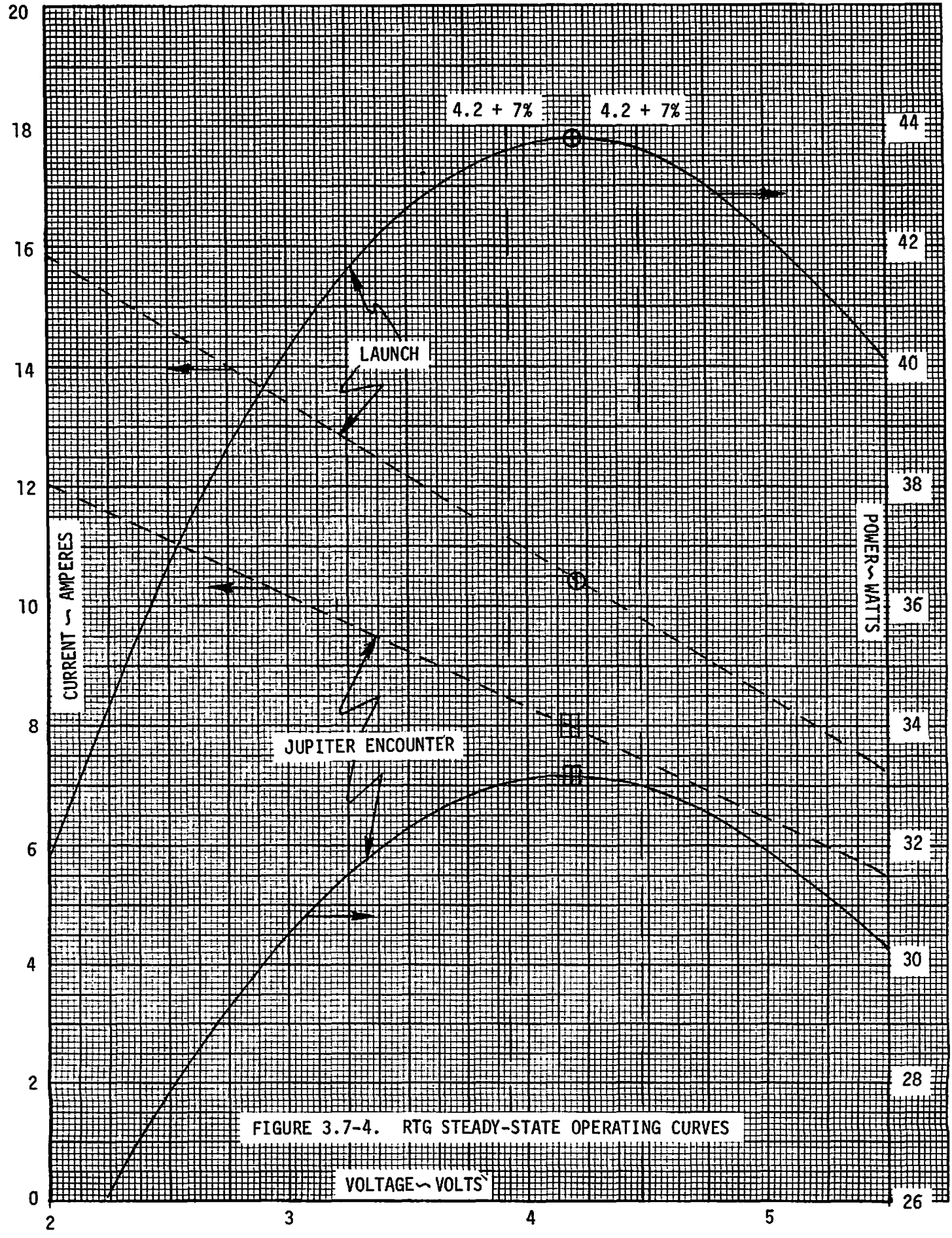


FIGURE 3.7-4. RTG STEADY-STATE OPERATING CURVES

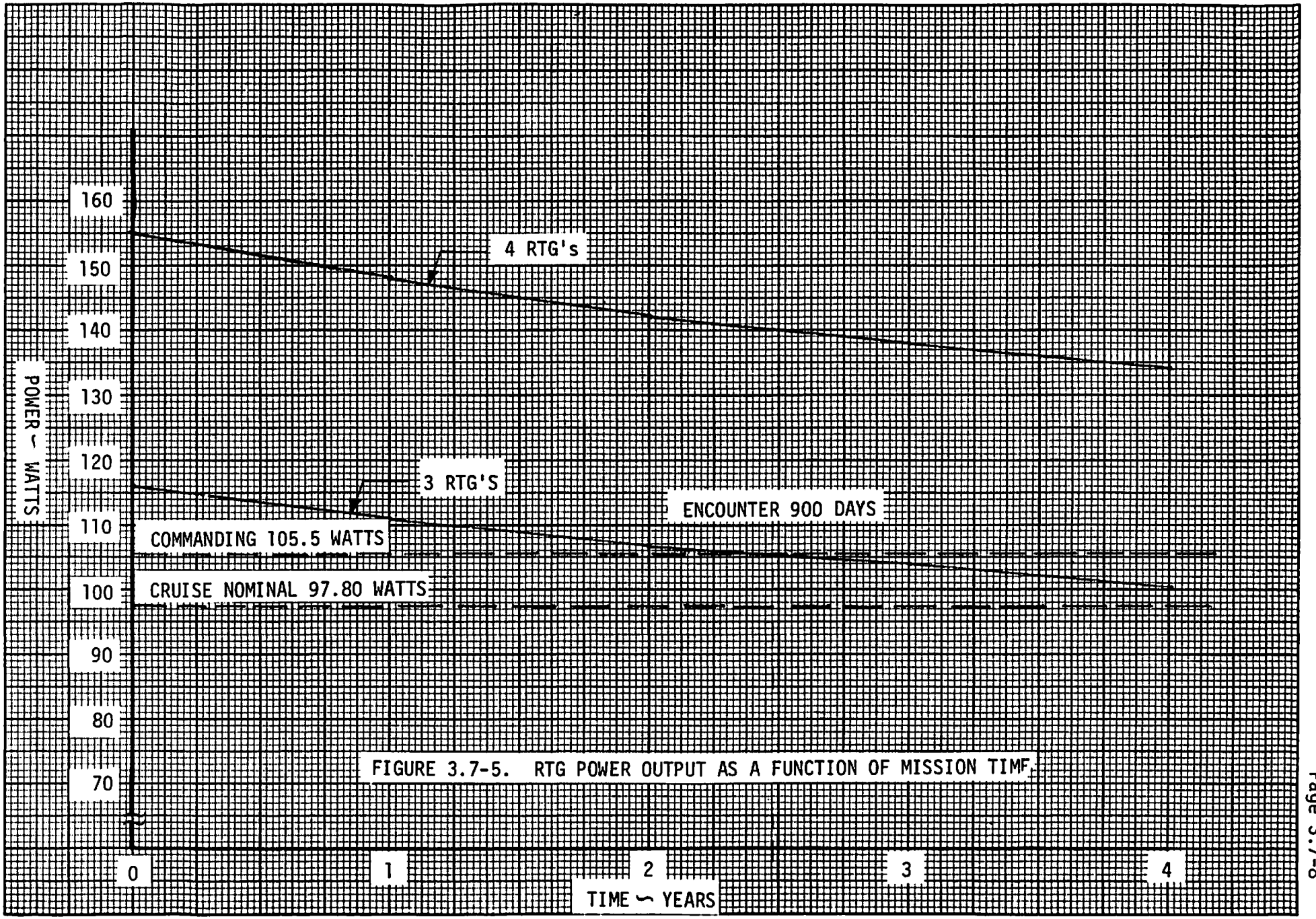


FIGURE 3.7-5. RTG POWER OUTPUT AS A FUNCTION OF MISSION TIME

battery discharge to Jupiter encounter. With the cruise load and 3 RTGs the load can be supported for approximately 4.5 years. It is emphasized that the above performance is predicated on predicted RTG power output data. Flight performance of the RTGs may deviate somewhat from these predicted values with corresponding increases or decreases in the times stated above.

3.7.1.2.3 Battery Load Sharing. When the RTGs have degraded to the point that they are no longer capable of supporting the spacecraft load, the battery may be used to extend the full spacecraft load operating time. A portion of the spacecraft load must be turned on during battery discharge and turned off during battery charge. This mode of operation is denoted as load duty cycling. The capability of the battery to operate in a duty cycle mode is a function of time since battery capacity degradation will be incurred during the mission. The predicted performance of the battery is shown in Figure 3.7-6. Discharger efficiency effects on battery energy delivered to the bus were taken into account. Figure 3.7-7 shows the predicted battery capacity degradation as a function of time. At encounter the battery will have degraded to 50% of its original capacity.

3.7.1.2.4 Supply Power with Minimum Losses. A power flow diagram of the power subsystem with 4 RTGs operative is shown in Figure 3.7-8. The major portion of the loss is in the shunt regulator and radiator which must dissipate excess RTG power to maintain bus regulation. Other large power dissipations occur in the IAs, CTRF and PCU TRF. The power flow diagram with 3 RTGs operative and degraded to the point that they balance load demand is shown in Figure 3.7-9. Note that the shunt losses are reduced to 0.9 watt. The inverter, cable, and PCU TRF losses have been reduced due to decreased RTG output and shunt current. The CTRF losses are essentially unchanged. The overall power subsystem efficiency is approximately 72.5%.

3.7.2 Detailed Unit Description

This section contains detailed descriptions of each major component of the subsystem.

3.7.2.1 Inverters. There are 4 inverters per spacecraft contained in 2 inverter assemblies. Each inverter is connected to an RTG power source through a flat deployable cable. Each pair of inverters within an assembly is driven by a common 2.5 kHz oscillator. The assemblies are connected so that the oscillator in either assembly will drive all 4 inverters. Switchover between oscillators is accomplished automatically and can not be detected in flight. The inverters transform the 4.2 vdc from the RTGs to 61 vrms at 2.5 kHz. The outputs of the inverters are cross-strapped in the PCU to form the AC bus which is regulated to $\pm$3%. The inverter efficiency is approximately 88%.

The inverter input voltage is typically 3.95 vdc at rated load. The input power at BOL is 40 watts nominal and 46 watts maximum. At Jupiter encounter the worst case predicted input power is 33 watts. The

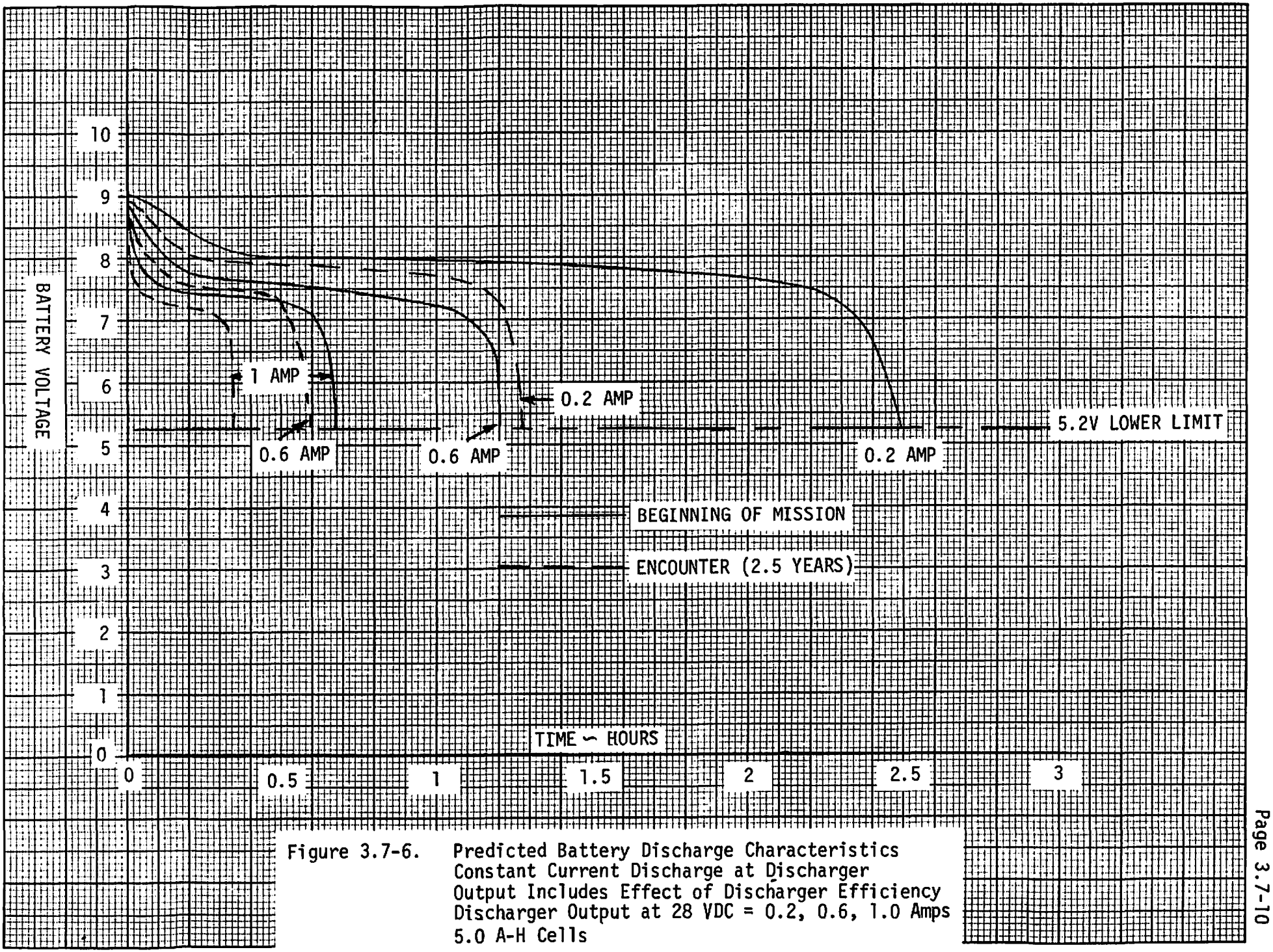


Figure 3.7-6. Predicted Battery Discharge Characteristics
Constant Current Discharge at Discharger
Output Includes Effect of Discharger Efficiency
Discharger Output at 28 VDC = 0.2, 0.6, 1.0 Amps
5.0 A-H Cells

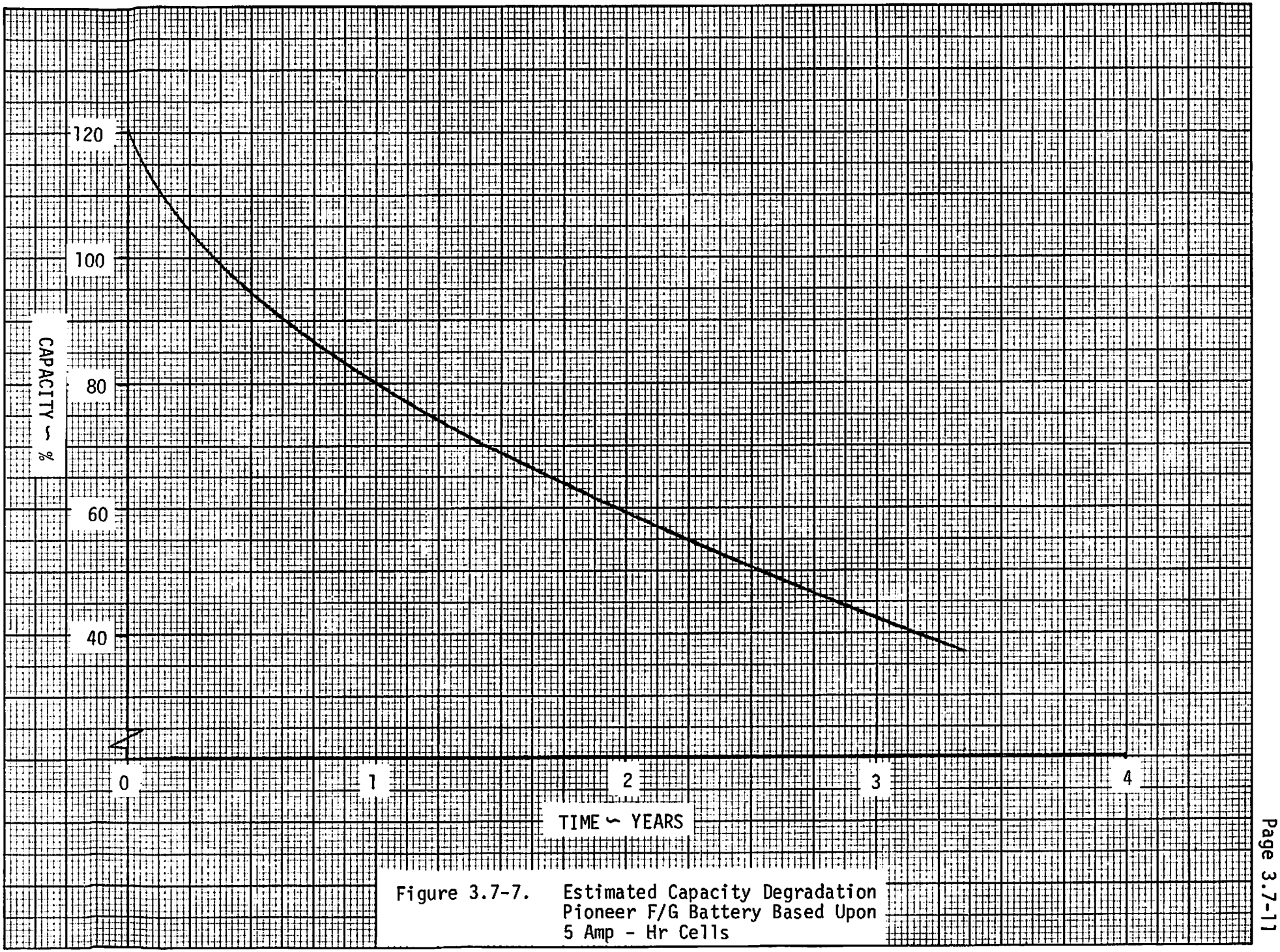


Figure 3.7-7. Estimated Capacity Degradation Pioneer F/G Battery Based Upon 5 Amp - Hr Cells

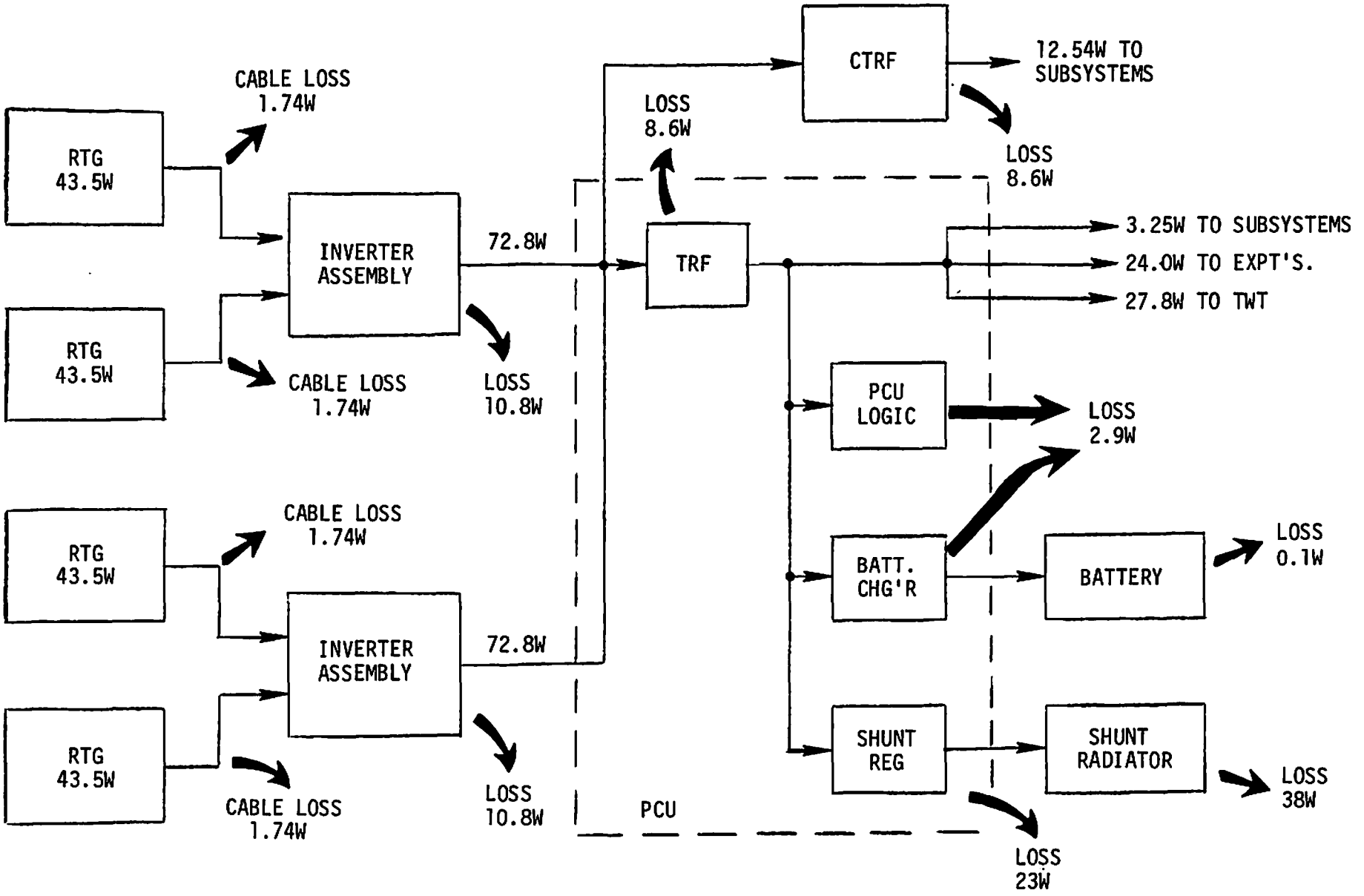


Figure 3.7-8. Power Flow Diagram, 4 TRG's Operative, Beginning of Mission

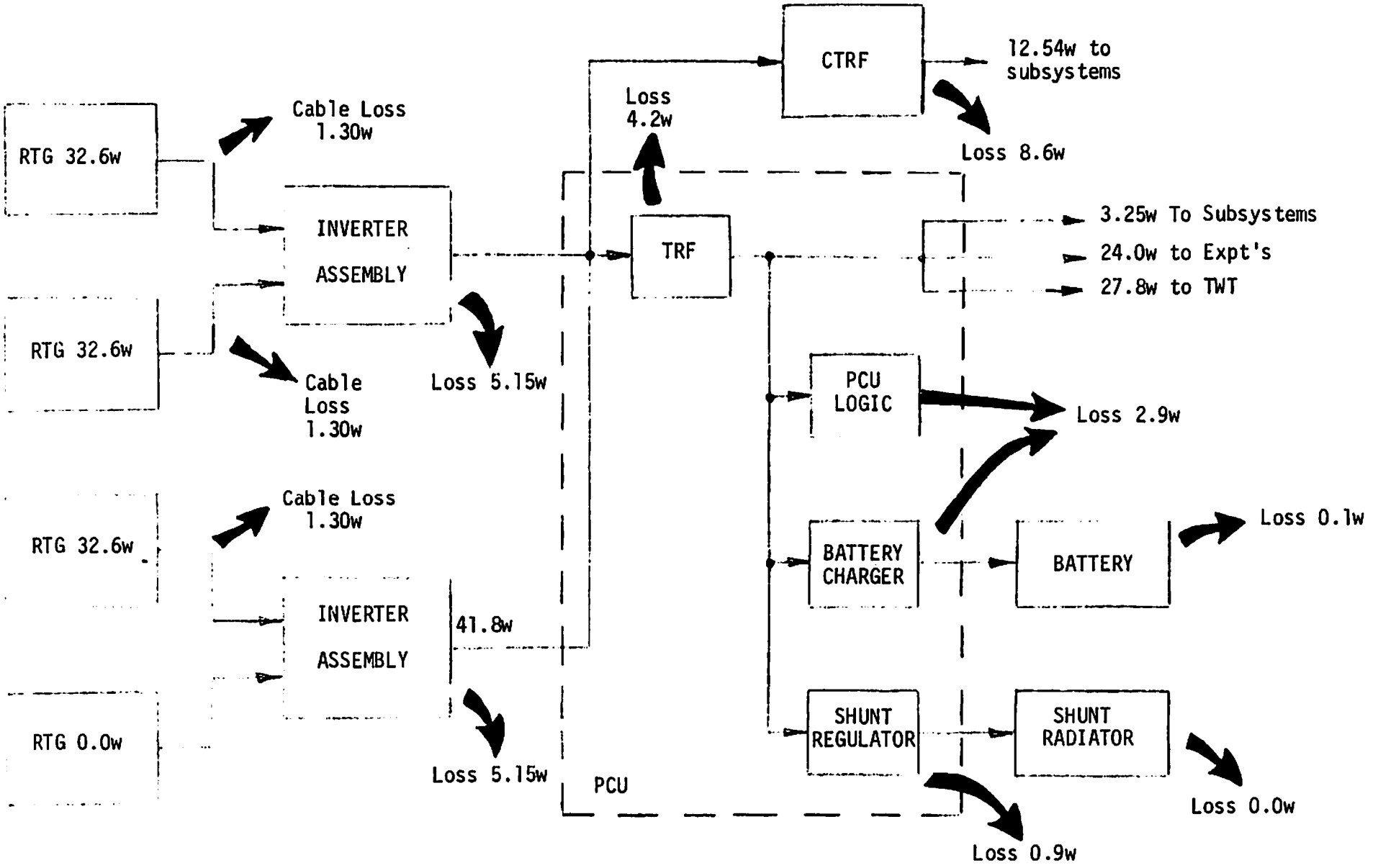


Figure 3.7-9. Power Flow Diagram, 3 RTG's Operative, Encounter

output is a trapezoidal wave with a voltage of 61.0 vrm at a frequency of 2.5 kHz + 0.5 kHz - 0.2 kHz. Figure 3.7-10 is a block diagram of the assembly.

Telemetry is provided for RTG current, voltage, hot junction temperature and Fin-root temperature. Table 3.7-3 at the end of Section 3.7.3 provides a detailed list of the inverter telemetry.

The Inverter Assembly provides protection and isolation against an RTG or RTG cable short as well as an internal inverter failure. If one failure occurs and is isolated, the remaining inverter fault isolation circuits are inhibited. The fault isolation circuits can be reset by a command (PSP1) processed by the PCU. In order to determine if the fault circuit has operated, it is necessary to observe the RTG current telemetry. The RTG current after the fault circuit has operated will be zero. If one sends command PSP1 to reset the fault circuit, the relays (K1-K4) of each inverter assembly will reset. If the fault is true the relays in the appropriate inverter will open again. This can be observed on the RTG current telemetry. If the fault indication is erroneous the inverter will again convert RTG power.

3.7.2.2 Power Control Unit (PCU). The PCU provides central power conditioning, regulation, and distribution for the subsystem as shown in Figure 3.7-11. Specifically the PCU accomplishes the following.

- Cross-straps the square wave outputs of the four inverters to form an AC bus. Distributes the AC bus to the CTRF, CDU, and battery cell bypass electronics.
- Rectifies and filters the AC bus to form a DC bus. Distributes the DC bus to the TWT, experiments, spacecraft pulse loads and heaters. Provides fuse protection for TWT and individual experiment distribution.
- Regulates the DC bus to 28 vdc $\pm$ 1% ($\pm$2% including long term drift) by shunt regulation, and also by control of battery charge and discharge rates.
- Regulates the AC bus to 61.0 volts rms $\pm$ 3% tolerance as a consequence of the DC bus regulation.
- Dissipates excess RTG power in a linear dissipative shunt regulator.
- Charges the battery at current and voltage levels determined by both battery requirements and availability of excess RTG power.
- Discharges the battery at a rate determined by RTG power deficit, up to 26.2 watts.

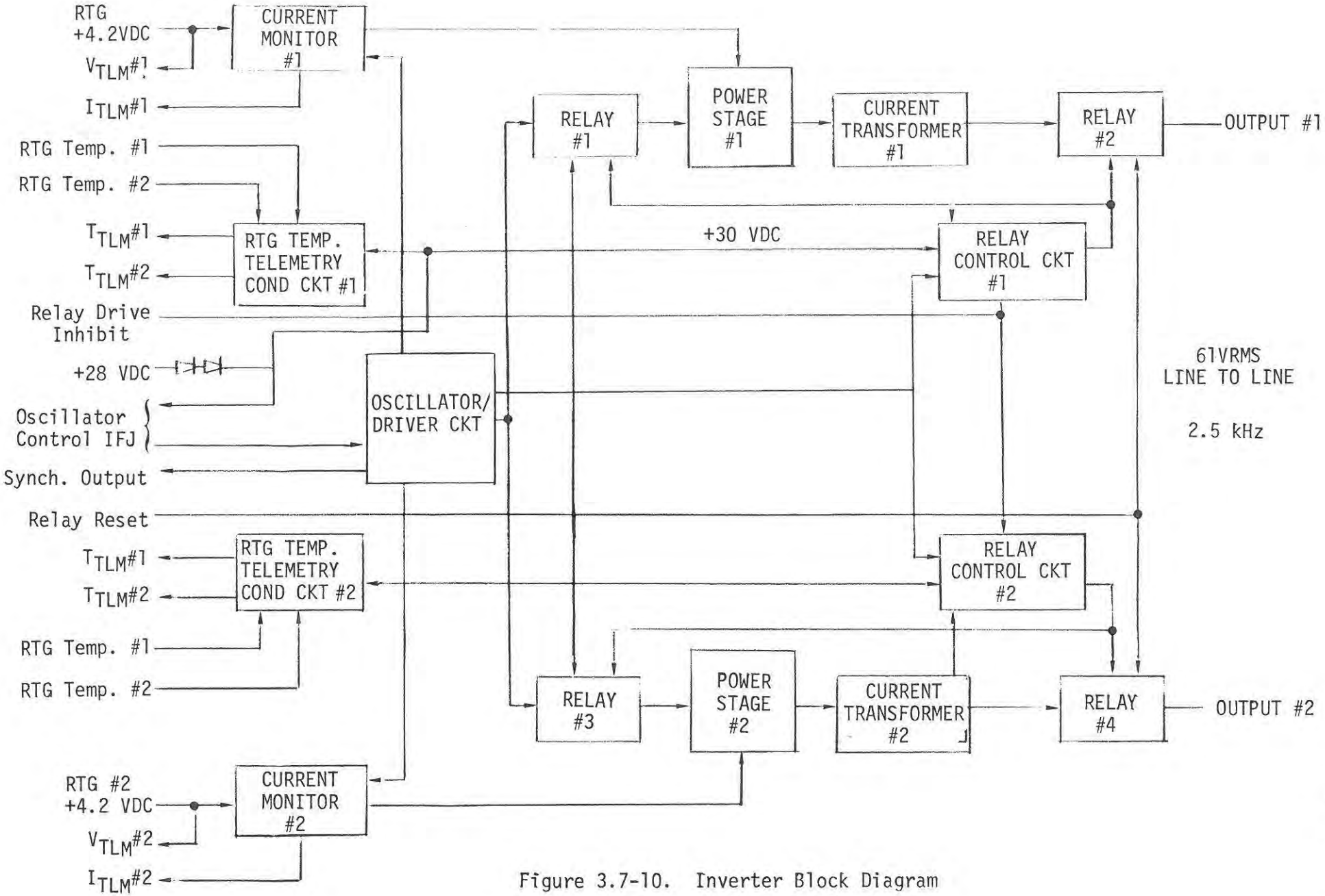
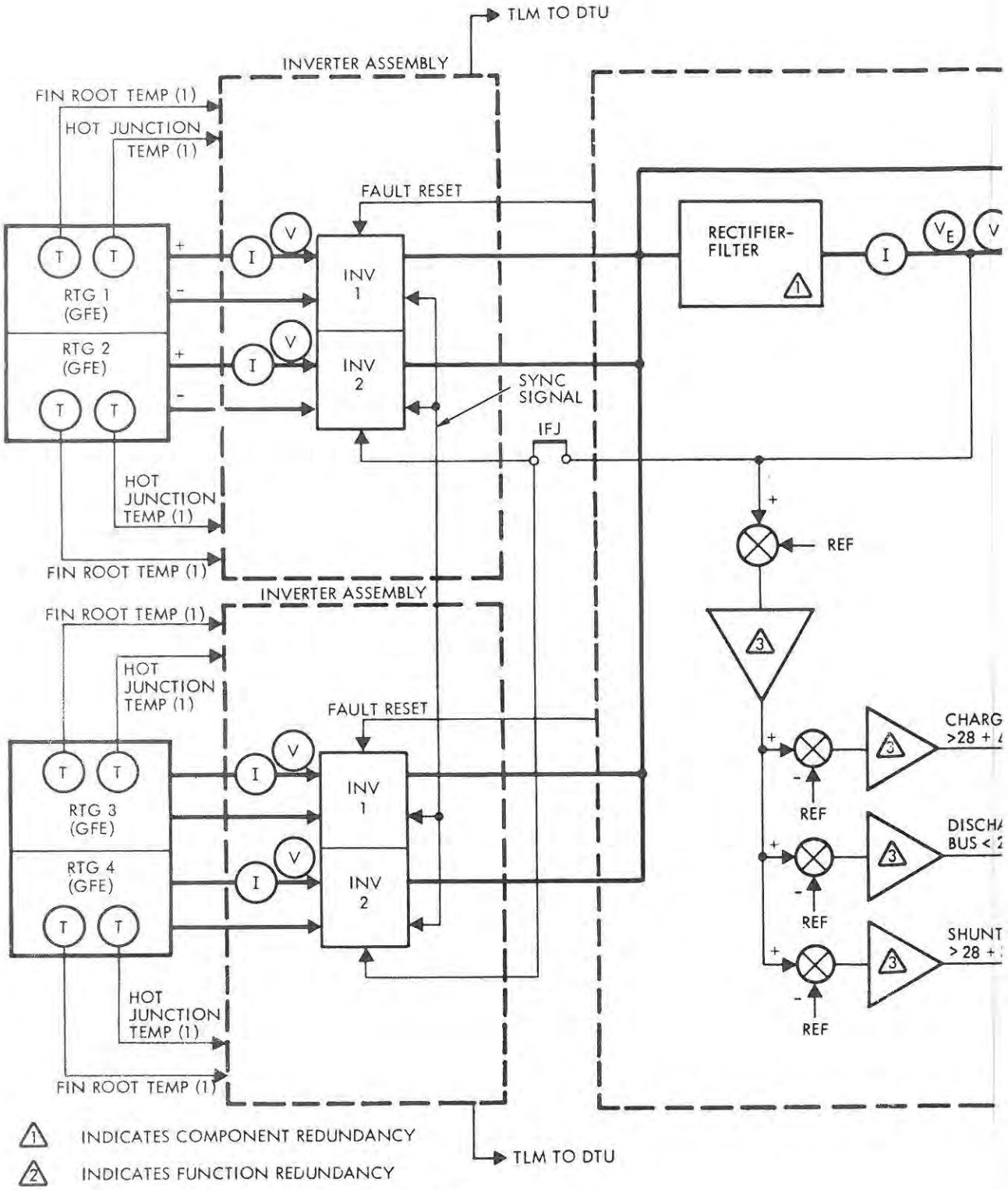


Figure 3.7-10. Inverter Block Diagram



1 INDICATES COMPONENT REDUNDANCY

2 INDICATES FUNCTION REDUNDANCY

3 INDICATES MAJORITY VOTING REDUNDANCY

4 SHORT TERM REGULATION ± 1%, EOL REGULATION ± 2%

I INDICATES CURRENT TLM SIGNAL CONDITIONING

V_E INDICATES EXPANDED SCALE VOLTAGE TLM SIGNAL CONDITIONING

V INDICATES VOLTAGE TLM SIGNAL CONDITIONING

ST INDICATES STATUS TLM SIGNAL CONDITIONING

T INDICATES TEMPERATURE TLM SIGNAL CONDITIONING

INDICATES MAIN POWER FLOW PATHS

INDICATES SIGNAL PATHS

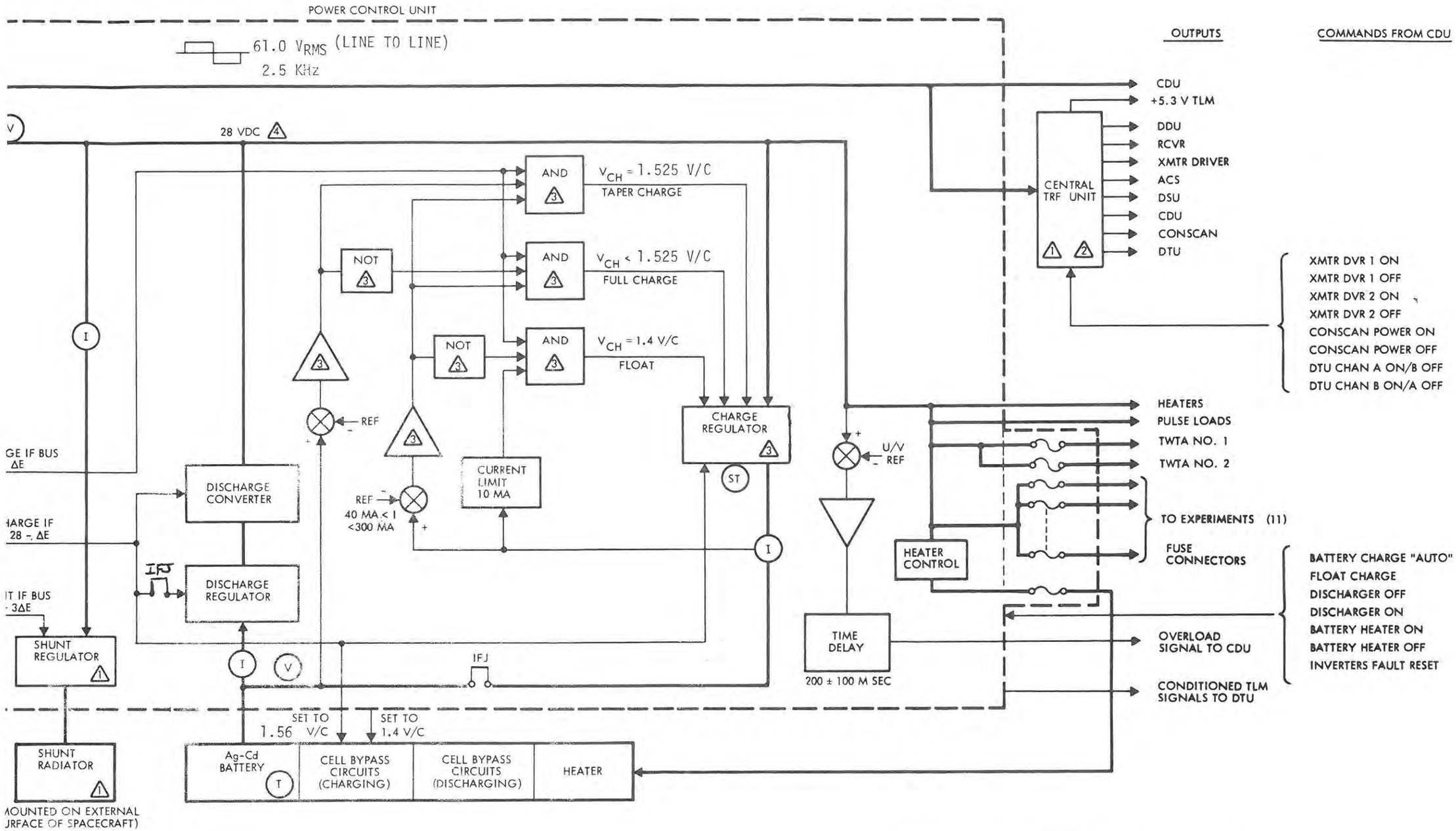


FIGURE 3.7-11 PIONEER F/G ELECTRICAL POWER SUBSYSTEM BLOCK DIAGRAM

- Provides an overload sense signal for use by the CDU in priority load disconnect when the DC bus voltage decays below 26.0 $\pm$ 0.5 vdc.

- Provides diagnostic EPS telemetry outputs.

The discussion which follows concerns the operation of the PCU in response to load changes. These load changes are sensed by a majority voting bus voltage sensor which in essence generates enabling signals causing shunt regulator operation, charge regulator operation or discharge regulator operation. Circuit mechanization is such that: 1) battery discharge may not occur while shunting or charging is enabled and 2) shunting may not occur until current in excess of maximum battery recharge current plus load current is available (see Figure 3.7-2).

The majority voting voltage error amplifier maintains bus regulation by shunting excess power charging the battery or discharging the battery. If the load current demand is greater than the RTG capability at the bus voltage, then the discharger is enabled and the bus voltage is supported by the battery. If load current demand is the same as RTG capability, no battery charging current is supplied from the 28v bus. If RTG current slightly exceeds load current demand and battery recharge is required, the excess current is supplied to the battery. Finally, if RTG current capability greatly exceeds load demand, bus voltage regulation is achieved by charging the battery at the maximum limit allowed by the battery charger (if required) and any current in excess of this level is directed to the shunt radiator via the shunt regulator.

Shunt regulation is utilized since, as RTG power degradation occurs, the dissipation required by the regulation element continually decreases. This results in minimum shunt losses (approximately 0.9 watt) at that point in the mission when the RTG output power equals the total vehicle load. This also results in maximum power availability thereafter since load power can be maintained equal to or less than the source capability by ground command.

The shunt regulator also provides low bus impedance at low frequencies (<1000 Hz) by maintaining bus voltage relatively constant in the presence of load current changes. For the case where RTG power equals the load power demand and the shunt current is zero, all sensitive loads are isolated from a possible rise in bus impedance by regulators contained in the CTRF or TWTA (for loads in excess of RTG power capability, bus impedance is held small by the battery and its discharge regulator).

This source regulation technique is particularly suited to RTG thermal requirements which necessitate operation near the maximum power point voltage or less (to minimize RTG internal heating which causes increased degradation). Regulation of the 28v bus by the shunt is reflected through the inverter assembly transformers to the RTG outputs. The inverter assembly transformer turns ratio has been selected such that when the bus is at 28 vdc, the RTGs will be at their nominal maximum power point voltage.

The PCU 28 vdc power is supplied at ± 1% initially, ± 2% including long term drift. The 28 vdc is supplied to the TWTAs, Scientific Instruments, heaters, and other pulse loads. Other subsystem provisions include: 1) telemetry of important power subsystem parameters for in-flight power system performance analysis; 2) overload condition indication; 3) experiment and TWTA fuses; 4) provisions to implement ground command operational changes and; 5) in-flight jumper control provisions related to prelaunch RTG installation.

The Discharge Regulator Enable (CMD SAT3) may be necessary to insure that the discharge path for the battery is always available if no failure has occurred in the discharge circuit or the battery. A discharge may be required if a failure occurs causing the AC bus to drop below about 30 vrm (line to line). The battery would turn on to keep the inverter oscillators switching. Therefore, it is suggested that the discharge status telemetry, TLM word C-128, be checked before the start of long periods when the spacecraft is not being monitored.

During battery discharge, telemetry is provided for battery current (TLM word C-126). The current supplied to the load must be calculated using the discharger efficiency curve, Figure 3.7-12.

3.7.2.2.1 DC Bus Regulation (4 RTGs). With four normal RTGs there should always be an excess of RTG power over load power for several years past encounter (Figure 3.7-5). Therefore, the PCU will regulate the 28 vdc bus exclusively by means of shunt regulation.

The shunt regulation is accomplished by a closed loop regulation scheme which utilizes an Error Amplifier (EA) and a dissipative shunt regulator. The EA senses the 28 vdc bus and provides a linear transconductance signal to the shunt regulator depending on the amount of excess RTG power. The shunt regulator in turn shunts the excess power resulting in 28 vdc bus regulation.

3.7.2.2.2 DC Bus Regulation (3 RTGs). With three degraded RTGs, the PCU uses the battery charge and discharge regulators along with the shunt regulator to maintain regulation of the 28 vdc bus. During the majority of the mission, RTG power will be greater than load power, and DC bus regulation will be accomplished by implementing the shunt regulation scheme described in the preceeding section. However, near encounter excess RTG power may not always be greater than load power. In these situations DC bus regulation will be accomplished by supporting the bus through battery discharge. This RTG/battery load sharing is accomplished in the following manner:

- If RTG power is greater than load power and if the battery is not fully charged, the battery is charged at a current limit up to 250 ± 50 ma. If the battery is fully charged, all excess power is directed to the shunt regulator.

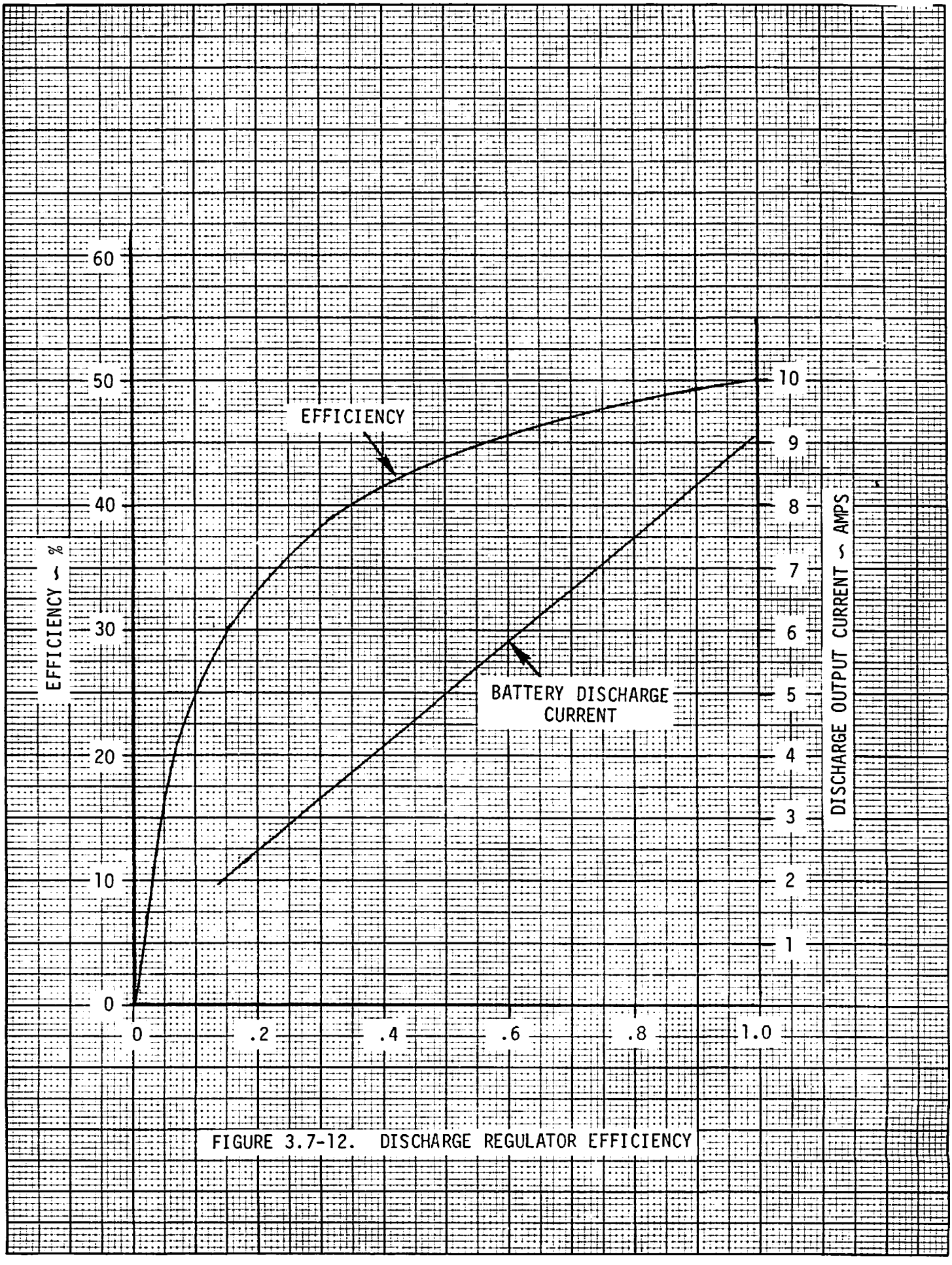


FIGURE 3.7-12. DISCHARGE REGULATOR EFFICIENCY

- If RTG power is less than load power, the discharge regulator provides up to 26.2 watts of battery power to the 28 vdc bus to maintain regulation.

3.7.2.2.3 AC Bus Regulation. The PCU cross-straps the outputs of the four inverter assemblies to form a common AC bus. This bus is then rectified and filtered within the PCU to form the DC bus. Since the DC bus is regulated as described above, the AC bus, being separated only by a rectifier-filter, will also be regulated. The tolerance on the AC bus regulation is 61.0 vrms ± 3%. This tolerance is greater than that of the DC bus due to the in-line tolerance of the rectifier-filter assembly.

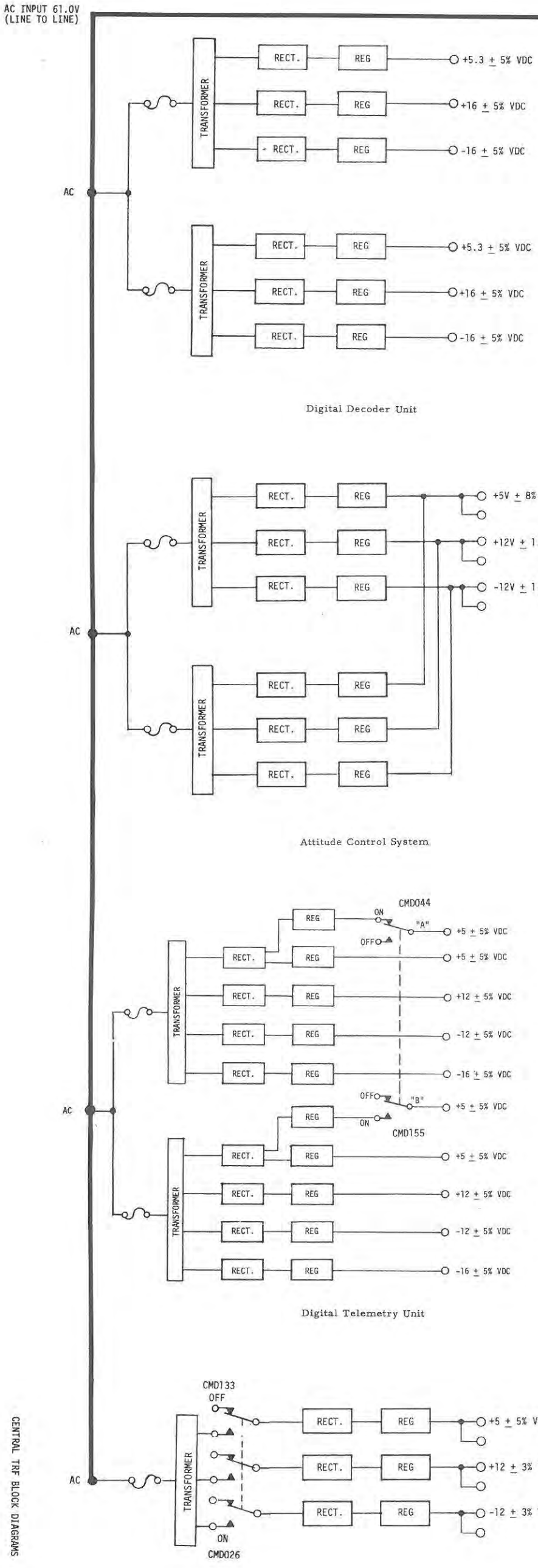
3.7.2.2.4 RTG Output Regulation. The RTG output voltage is regulated at a maximum power point of approximately 4.2 volts ± 7%.

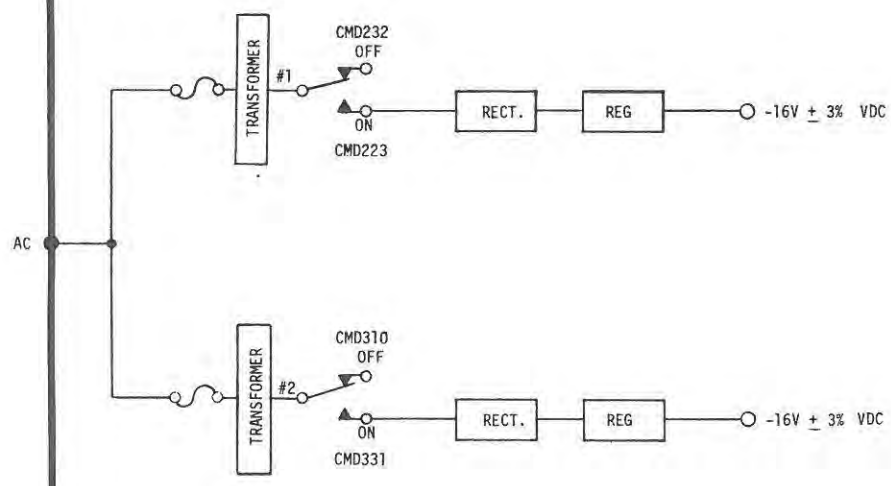
The outputs of the RTGs are connected to individual inverter assemblies which form the AC bus. Since the AC bus is regulated as described above, the individual RTG outputs (being separated from the AC bus by the turns ratio of the inverter assembly transformers, the inverter saturation drop and cable IR drops) will also be regulated. The value of the regulated voltage is in turn strictly a function of the turns ratio of the inverter transformers plus IR drops.

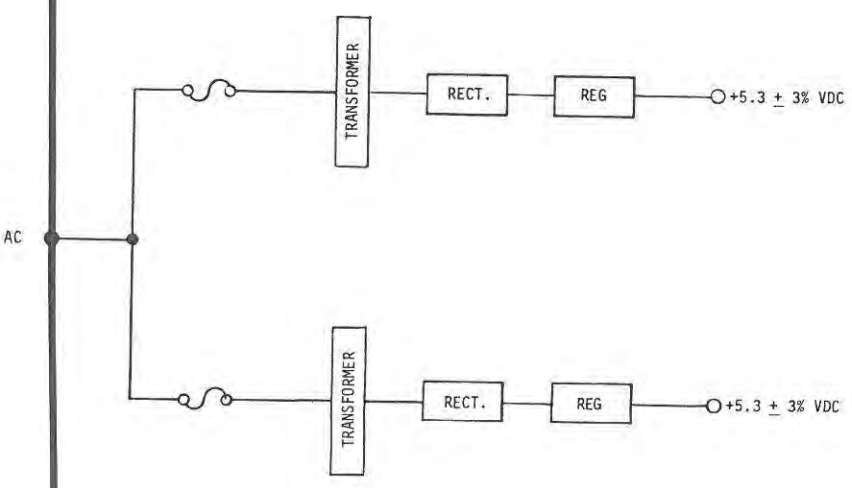
3.7.2.2.5 Undervoltage Provisions. The PCU senses the 28 vdc bus and provides an undervoltage signal to the Command Distribution Unit (CDU) with output characteristics as described below when the bus voltage decays at a maximum rate of 5 volts/second, below a level of 26.0 ± 0.5 vdc for a minimum duration of 0.2 ± 0.1 seconds. The PCU provides a degraded overload signal with increased delay time and decreased output voltage, when DC bus voltage decays down to 19.3 vdc minimum at a rate greater than 5 volts/second. The CDU sends a disconnect signal to all loads not essential to spacecraft survival. Reactivation of these loads is by ground command.

3.7.2.3 Central TRF. The central TRF contains transformer-rectifier-filters whose outputs are regulated either by series dissipative or switching regulators. Each voltage output of the CTRF (with the exception of the Conscan and DSU outputs) has separate redundant TRF-regulators. A combination of automatic or command switching between redundant TRF-regulators is provided.

The CTRF converts the 61 vrms AC bus to conditioned and regulated voltages (±12v, ±16v, +5.3v, and +5v) for the various spacecraft loads. The CTRF provides voltage telemetry for the CDU +5.3v (words C-117 and C-118). It provides ON/OFF control for the transmitter driver and conscan, and DTU "A" and "B" ON/OFF. The AC input to each CTRF slice is protected by a fuse and each output is current limited. The block diagram, Figure 3.7-13, shows the user, the voltages, and switching. The power budget given in Appendix D, shows the power for each output. When calculating the input power, use 66% efficiency for the CTRF if the load is above nominal power.


AC INPUT 61.0V
(LINE TO LINE)
AC
TRANSFORMER
RECT.
REG
+5.3 ± 5% VDC
+16 ± 5% VDC
-16 ± 5% VDC
+5.3 ± 5% VDC
+16 ± 5% VDC
-16 ± 5% VDC
Digital Decoder Unit
AC
TRANSFORMER
RECT.
REG
+5V ± 8% VDC
+12V ± 1.5% VDC
-12V ± 1.5% VDC
Attitude Control System
AC
TRANSFORMER
RECT.
REG
CMD044
ON
OFF
"A"
"B"
CMD155
+5 ± 5% VDC
+12 ± 5% VDC
-12 ± 5% VDC
-16 ± 5% VDC
Digital Telemetry Unit
AC
TRANSFORMER
CMD133
OFF
ON
CMD026
RECT.
REG
+5 ± 5% VDC
+12 ± 3% VDC
-12 ± 3% VDC
CONSCAN Signal Processor (Digital)


AC
TRANSFORMER
#1
#2
CMD232
OFF
ON
CMD223
CMD310
OFF
ON
CMD331
RECT.
REG
-16V ± 3% VDC
-16V ± 3% VDC
Transmitter Driver


AC
TRANSFORMER
RECT.
REG
+5.3 ± 3% VDC
+5.3 ± 3% VDC
Command Distribution Unit

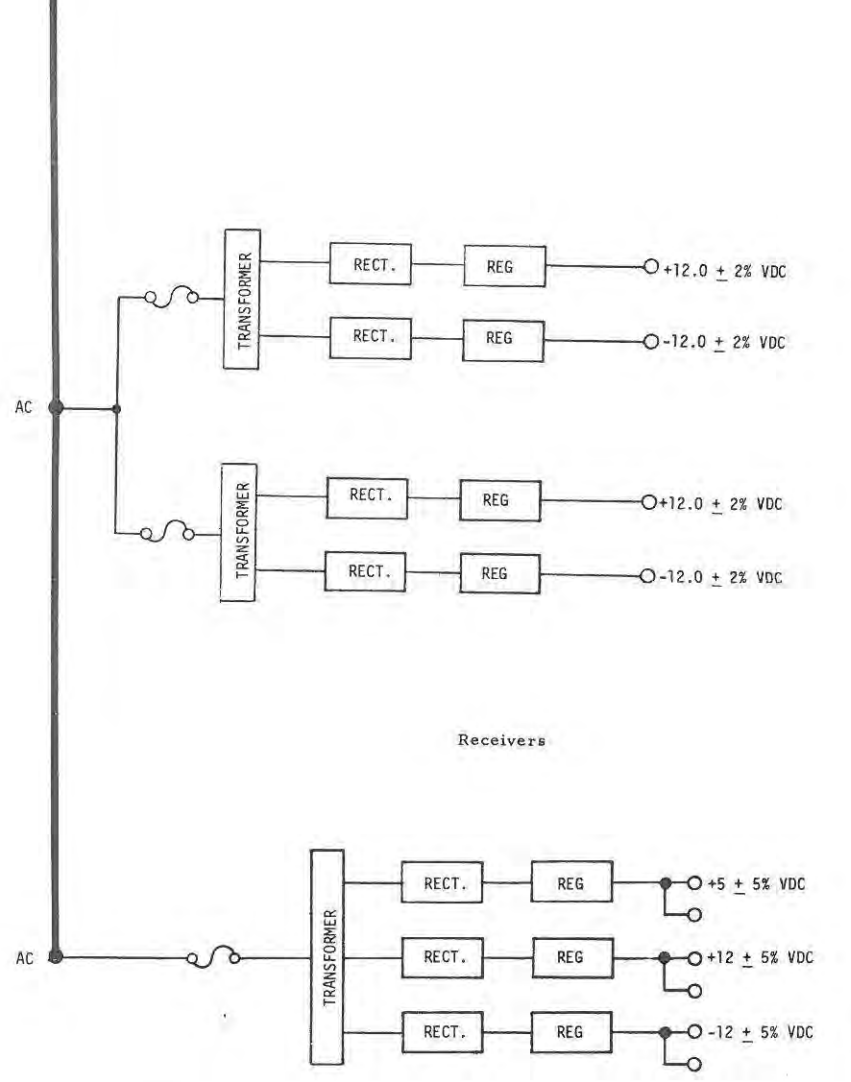

AC
TRANSFORMER
RECT.
REG
+12.0 ± 2% VDC
-12.0 ± 2% VDC
+12.0 ± 2% VDC
-12.0 ± 2% VDC
Receivers
AC
TRANSFORMER
RECT.
REG
+5 ± 5% VDC
+12 ± 5% VDC
-12 ± 5% VDC
Data Storage Unit

Figure 3.7-13

The response characteristics of the CTRF cannot be observed on telemetry. For reference the following characteristics are given:

Input	
Voltage	61.0 vrm ±3% (line to line)
Source Impedance	25 ohm nominal
Frequency Response	2.5 kHz (+0.5, -0.2) kHz
Max. Input Transient	+20% for 10 milliseconds - 5% for 150 milliseconds
Output	
DC Voltage and Regulation	See Figure 3.7-13
AC Component	a. 140 mvp-p from 30 Hz to 100 MHz except transmitter drive. b. Transmitter driver 40 mvp-p from 30 Hz to 200 Hz, 140 mvp-p from 1.5 kHz to 100 MHz
Maximum Power	24.98 watts to users

The CTRF provides two types of fault isolation. In the event of a secondary fault each output to each user is current limited to limit the maximum power that can be consumed. In addition the input to each slice is fused to remove CTRF shorts from the AC bus. Where commands are available, faults can be removed from the system. In non-command situations the fault will draw additional power from the system.

When commanding ON or OFF a CTRF load, the operator should observe shunt current telemetry (words C-122 or C-209). Since the shunt regulator absorbs the power of the unit when it is OFF, its telemetry can be a useful indicator when commanding these loads ON or OFF.

3.7.2.4 Battery. The battery chosen for the Pioneer F/G mission is silver-cadmium (silver-cadmium is used because of low magnetic fields) composed of eight, 5 ampere-hour cells in series. Each cell is bypassed in both the charge and discharge direction with cell protection circuitry. Discharge bypass circuitry insures battery survival if a cell fails open or fails to a very low capacity value and reverses voltage on discharge. Cell charge bypass circuitry is included on the battery to protect against cell divergence or overcharge during charging or cell failure by limiting the maximum cell voltage. Charge protection is necessary because the PCU can control only battery terminal voltage. Without cell protection, if a cell fails in the shorted mode (or diverges significantly) the remaining cells could experience a greater than tolerable voltage on charge, resulting in worsened stress and more rapid degradation. The charge bypass is designed with two commandable voltage bypass limits which correspond

to the charge and flat levels required to minimize overstress and prevent excessive pressure.

The battery discharge current level is limited to about 1 amp by the discharge regulator. The battery recharge technique is discussed in Section 3.7.3. The battery charge cycle can be controlled by ground command if required (BAT1 and BAT2). A heater in the battery compartment can be controlled by command (BAT0 and BAT9) to raise the battery temperature when a discharge is anticipated or to maintain the battery above the minimum allowable temperature.

Figure 3.7-14 shows variation in battery voltage and current during a typical charge/discharge cycle. The battery is charged approximately 250 ma until the 12.2v full charge voltage is reached. The battery is maintained at 12.2 volts while the charge current tapers to 50 ma. At this point the charge limit is switched to 11.3 volts (which corresponds to the open circuit voltage of the battery) and the charge current drops to practically zero.

When the transition from float to discharge occurs the battery voltage drops from 11.3 volts to approximately 9 volts at the beginning of discharge. The battery "plateau" voltage during discharge is about 8.5 volts.

A simplified block diagram of the battery is shown in Figure 3.7-15. Cell reversal protection limits the reverse potential of each cell to -0.2 volts. The circuit operates only during deep discharge when cells having the lowest capacity may discharge fully and reverse voltage due to being driven by the remaining higher capacity cells. The reverse protection circuits are capable of bypassing 10 amps. The cell reversal circuits also provide a current path around open circuited cells.

The overcharge protection electronics limits the voltage of each cell to 1.56 volts during the maximum rate portion of the charge cycle. The cell charge bypass level is commandable to 1.41 volts per cell. During periods of extended float charging, the 1.41 v/cell bypass should be commanded to minimize cell divergence.

3.7.3 Power Management

This section has been divided according to the possible power subsystem operational modes which are:

1) Shunting

2) Battery Charge/Discharge

3) Undervoltage

4) Failure Operation

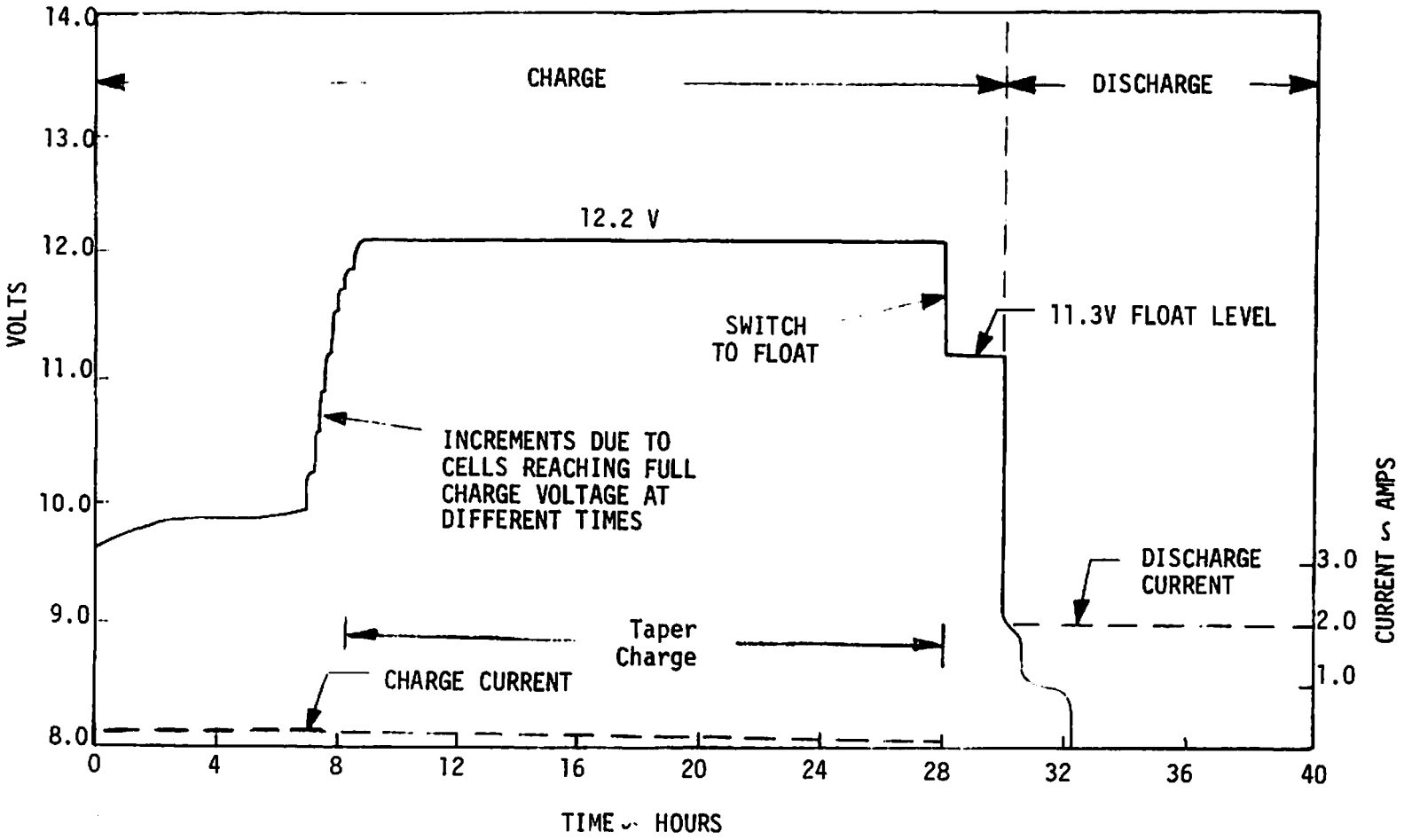


FIGURE 3.7-14. BATTERY VOLTAGE AND CURRENT DURING CHARGE/DISCHARGE CYCLE

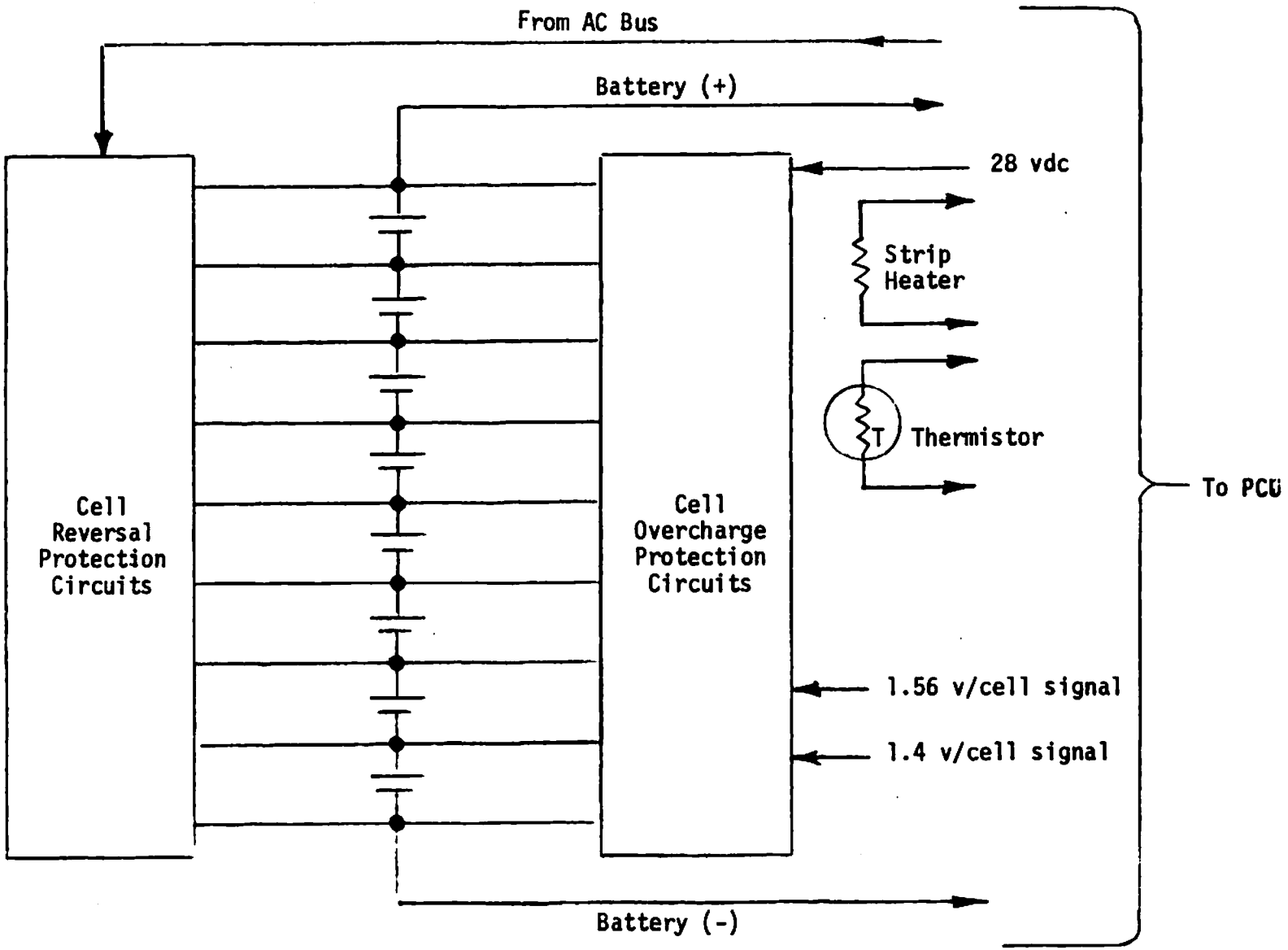


Figure 3.7-15. Battery Block Diagram

3.7.3.1 Shunting. If RTG degradation is as predicted and no RTG failures occur, the power subsystem will operate in the shunting mode for essentially the entire mission (see Figure 3.7-5) including several years in the post encounter phase. Excess RTG power capability is readily determined from the shunt current telemetry sensor since shunt current multiplied by the known bus regulation voltage of 28 vdc is the excess RTG power (words C-122 and C-209). Figure 3.7-16 shows the predicted variation in shunt current as a function of time with the spacecraft in the cruise mode. Even with 3 RTGs operative, the shunt current is predicted to be above zero after 4 years. The shunting function is performed by a shunt regulator (located in the PCU) in concert with a shunt radiator (located on spacecraft exterior panel). The power dissipated inside the PCU will vary with shunt current flowing (Figure 3.7-17). Shunt current greater than approximately 2 amp should be present only during modes of operation when the RTGs are new and the spacecraft is operating with a reduced load. The PCU temperature is strongly dependent upon the magnitude of the shunt current since up to 40 watts may be dissipated in the PCU due to shunt regulator operation. The shunt radiator is simply a resistor and its dissipation is proportional to the square of the shunt current present.

3.7.3.2 Battery Charge/Discharge. The PCU battery charge regulator charges the battery at current and voltage levels determined by both battery needs and RTG power available. The battery charger characteristics are as follows.

1) Charge battery at 0 - 250 $\pm$ 50 ma current limit until V_{BAT} = 12.2 v (8 cells · 1.525 v/cell).

2) Charge battery at V_{BAT} = 12.2 v until I_{BAT} = 50 $\pm$ 10 ma.

3) Charge battery at V_{BAT} = 11.28 v (8 cells · 1.41 v/cell). Also, the battery charge regulator must:

4) Provide ground charge capability by use of automatic and float charge commands.

5) Provide 10 ma current limit to battery when ground commanded to float charge.

6) Provide bi-level telemetry status for automatic and float charge commands.

Three modes of battery operation are possible. These are.

1) Automatic Charge

2) Command Float Charge

3) Discharge

Figure 3.7-18 is a logic flow diagram of the battery control technique.

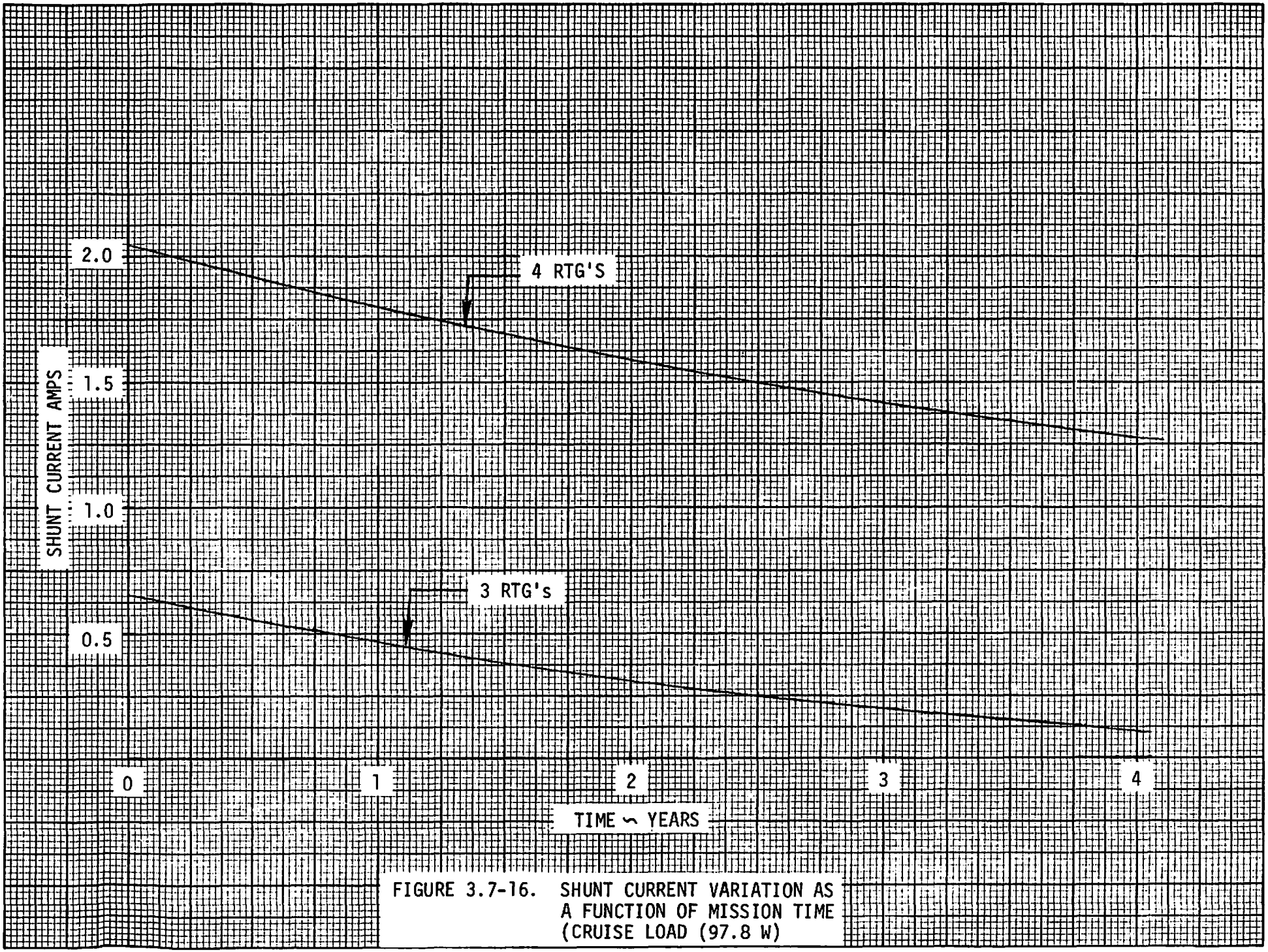


FIGURE 3.7-16. SHUNT CURRENT VARIATION AS A FUNCTION OF MISSION TIME (CRUISE LOAD (97.8 W)

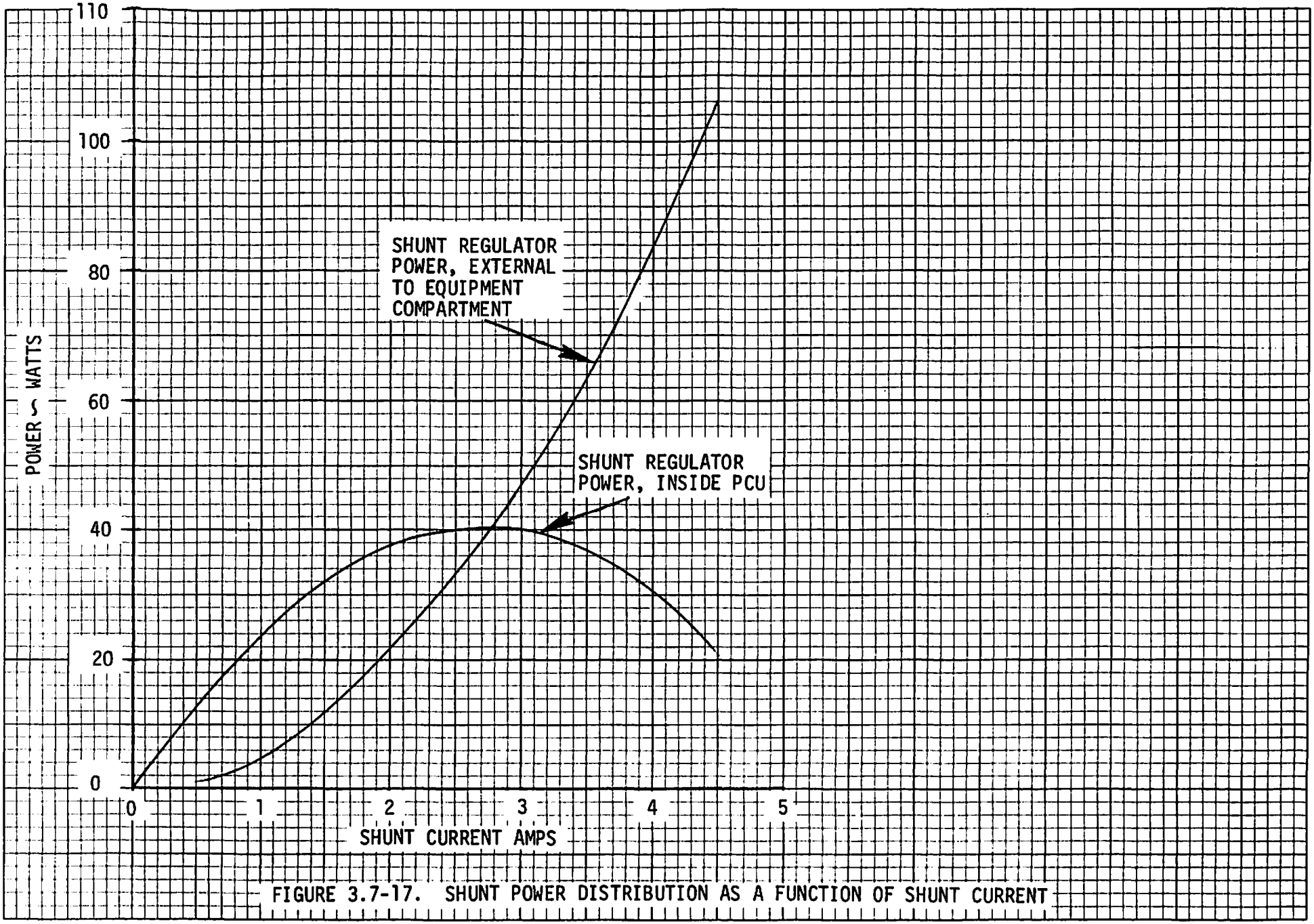
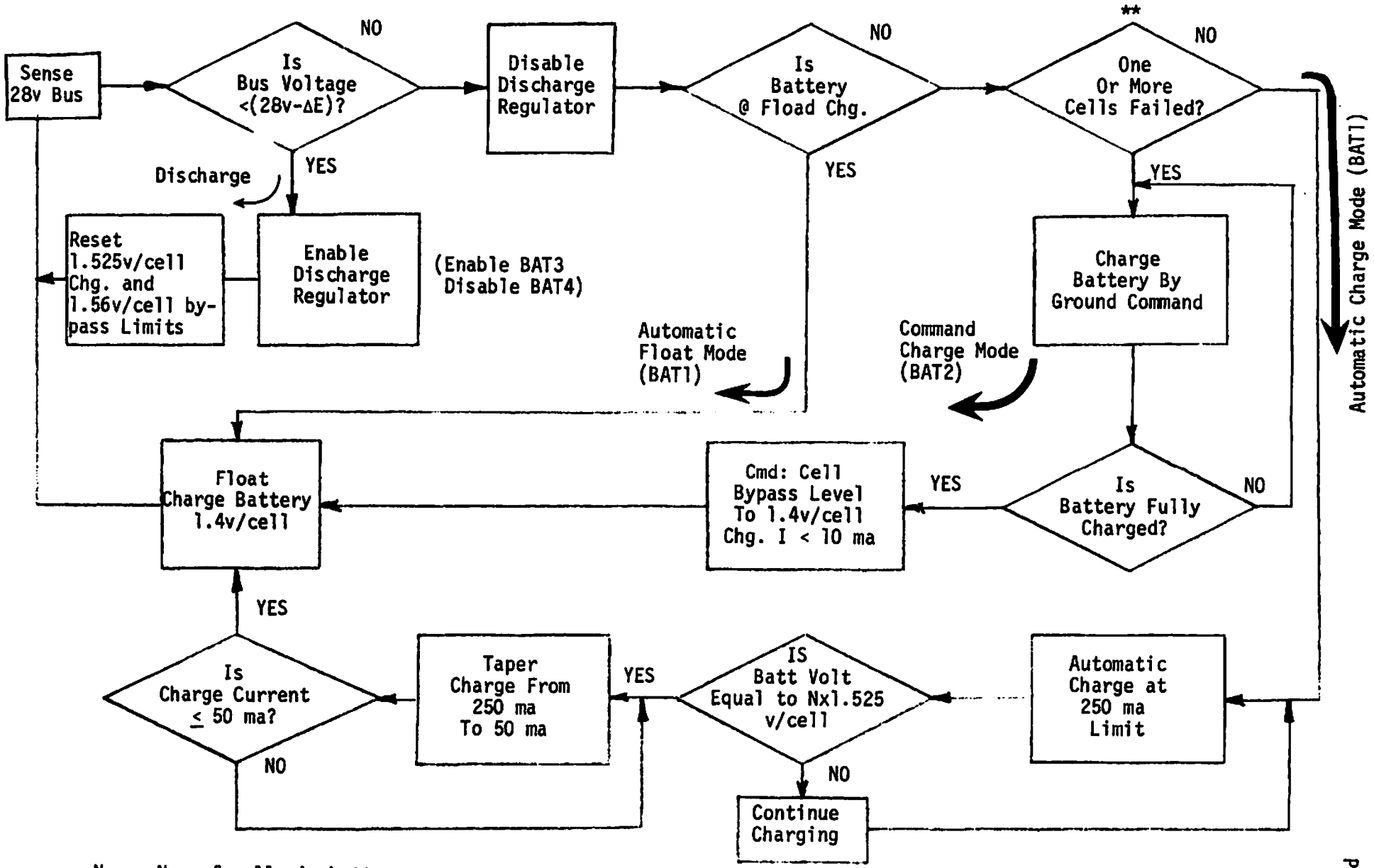


FIGURE 3.7-17. SHUNT POWER DISTRIBUTION AS A FUNCTION OF SHUNT CURRENT



N = No. of cells in battery
** = TLM monitor required and decision by command

Figure 3.7-18. Pioneer Battery Control Logic Flow Diagram

3.7.3.2.1 Automatic Charge Mode (see Figure 3.7-18). Conditions for the automatic charge mode are: 1) Battery Discharge, 2) RTG Power > Load Power, and 3) Automatic Charge Mode (Command BAT1). At this time, the battery charge regulator controls the charge current into the battery at 0 to 250 ± 50 milliamperes. The battery will be charged at this rate until the battery voltage reaches 12.2 vdc ± 1% (8 x 1.525 v/cell). The charge regulator will then be in the voltage limit mode and the charge current will start to decrease as the battery voltage increases. When the charge current drops to 50 ± 10 milliamperes, the charge regulator will switch to the 11.28 ± 1% voltage limit mode (1.41 v/cell). The charge regulator will keep the battery fixed at this voltage (Float Mode) until the next battery discharge. At this time, the autocharge cycle will repeat itself. When the battery voltage is held to 11.28v the current will automatically taper off to essentially zero. Hence, the term "float" is used to describe this condition.

3.7.3.2.2 Command Float Charge. The command charge mode is entered by ground command for one of the following three possible reasons.

1) A failed battery cell which prevents normal operation of the battery charge regulator in the automatic mode.

2) A priority load condition where it is decided to redirect the excess power for charging the battery to one of the spacecraft loads.

3) Extended period of float charge during mission.

A battery cell can fail in two different ways: 1) Open and 2) short. A discussion of each of these failure modes with respect to failure detection and charge control is presented in the following paragraphs.

1) Open Cell. Because of the operation of the cell bypass circuitry, detection of an open cell can be accomplished only during battery discharge. Also, an open cell requires no need for manual charge operation of the battery.

   With an open cell, the cell bypass electronics provides a voltage across that cell equivalent to a fully charged cell (1.56 v/cell). At the same time, the bypass electronics passes the charge current to the other cells (0 - 250 ± 50 ma). Hence, an open cell looks normal to the battery charge electronics, and the battery charge cycle is similar to that described in the automatic charge mode. However, since the battery is now actually seven cells, a battery discharge will present a lower than normal battery voltage for a known load. At this time, the ground station can monitor the battery voltage (word C-106) and detect the presence of the open cell.

2) Short Cell. A shorted cell results in erroneous operation of the automatic charge mode of the battery charge regulator and necessitates charging of the battery in the manual charge mode. Its detection is imperative since battery operation in the automatic mode with a shorted cell could cause cell failure due to pressure buildup and battery overtemperature. It is anticipated, based upon past observations, that a cell short will not occur instantaneously but will be manifested as a gradual decrease in cell voltage. Battery telemetry evaluations every seven days should be sufficient for detecting an impending shorted cell condition and initiating corrective action.

With a shorted cell the maximum charge voltage the battery can attain is 10.85 volts (7 cells · 1.56 v/cell). This limit is dictated by the cell bypass electronics. Since this voltage is less than the 12.2 volt limit imposed by the battery charge regulator, the battery will never reach its normal full charge voltage. The result is that the charge regulator views the battery as being uncharged, and continues supplying charge current to the battery. When the battery voltage reaches 10.85 volts, the charge current will bypass the cells through the bypass electronics and the charge current will stay constant. Dissipation in the bypass circuits will cause the battery temperature to rise. It is this constant flow of charge current for an extended period of time ($\geq$ 2 days) and temperature increase which indicates a shorted cell on telemetry (words C-109 and C-115). After diagnosing this condition, the ground station should send the float charge command. From this time to the end of the mission the battery should be charged by manual control as described in the following paragraphs.

Assume that a shorted cell has been detected by telemetry diagnosis and the battery must be recharged by command. The cell bypass level is reset to 1.56 v/cell by a discharge (every time a discharge occurs an automatic function of the PCU resets the 1.525 v/cell logic and the 1.56 v/cell bypass levels.) The battery is then allowed to charge automatically (BAT1) at up to 300 ma until an average cell voltage of 1.525 v/cell is indicated by the battery voltage on telemetry. (During the automatic charge the 1.56 v/cell bypass limit is in effect on and cell voltages are limited to preclude undesirable gassing.) A command (BAT2) is now sent to float the battery at 1.41 v/cell. This command has a dual function. It limits the charge current to 10 ma (in the PCU) and sets a relay in the battery bypass circuitry, which changes the cell bypass level from 1.56 v/cell to 1.41 v/cell. The relay changes the bypass level by switching to a second set of biasing resistors on the bypass transistors. Therefore, the desired float level voltage

of 1.41 v/cell is accurately controlled by the bypass circuits instead of the PCU logic and is not affected by cell failures.

When prolonged periods of float charge are anticipated, the ground station has the option of either commanding float charge or allowing the PCU automatic circuits to control the float. Commanding float charge is desirable in this case because the bypass circuits limit each cell to 1.41 v whereas, the PCU logic senses the battery terminal voltage which can result in cell divergence.

Since the battery and charge control circuits can draw up to 9 watts from the bus during maximum charging it may be desirable to make this power available to other loads on a priority basis. This can be done by commanding the battery to float (BAT2).

3.7.3.2.3 Battery Discharging. The PCU automatically discharges the battery when RTG power is less than load power demand. The battery discharge regulator provides a current output to the DC bus in the range of 0 to 1 amp, as determined by the input signal received from the central error amplifier. The design battery voltage range for discharge regulator operation is 5.2 to 11.3 vdc.

The battery discharge regulator can be enabled or disabled by ground command. The battery discharge disable command (BAT4) turns the discharge regulator off, preventing any discharge of the battery. The battery discharge enable command (BAT3) places the discharge regulator into an operable condition. However, the ground station cannot discharge the battery with this command. It only enables the discharge regulator for subsequent use when RTG power is less than load power.

Status telemetry indicating the mode (enable/disable) of the battery discharge regulator is provided.

3.7.3.2.4 Battery Temperature Control. Battery temperature control is critical to battery life and capacity. Experience gained by TRW and NASA-Goddard during spacecraft flights using silver-cadmium batteries shows that battery capacity deterioration and battery life are both a function of temperature.

Current electrical load profiles for the spacecraft indicate the battery will remain dormant through most of the mission, and will be required to cycle only in the case of a failure of an RTG. Immediately after launch, the battery will be at 60 $\pm$ 10°F. Over the next 900 days the battery will decrease slowly to 10° $\pm$ 10°F. Battery temperature may vary due to compartment dissipation and side sun maneuvers. In laboratory tests of silver cadmium cells, it was established the battery can be charged at 20°F, but not at 0°F, and that there is a limited power output available at 0°F. At the same time it was determined through NASA-Goddard flight experience that an average temperature of less than 50°F was

necessary to achieve a 900 day life. Consideration of these two facts and the thermal limitations of the spacecraft, resulted in a compromise battery temperature profile in which high post launch temperatures (detrimental to life) are avoided by limiting the battery-spacecraft thermal coupling. The difficulty in battery charging at temperatures below 20°F is avoided by the addition of a commandable 1.6 watt heater. During the later phases of the mission the unheated battery temperature will fall below 20°F; below this temperature, the battery is still available for limited emergency discharges. Following the emergency discharge, or for a planned cycle, the battery heater should be commanded on whenever the temperature is below 20°F. The heater will increase the battery temperature approximately 20°F (see Figure 3.7-19).

The nominal mission profile results in an average battery temperature of less than 50°F. The only requirement for ground control is to command the heater on prior to battery charging if the temperature has fallen below 20°F (BAT9). The heater can be commanded off if desired (BAT0). Unless there are prolonged periods of planned non-cycling after the heater actuation, the heater should be left on.

3.7.3.3 Overload Protection. If at any time the power available from the RTGs and battery is insufficient to supply the load, the PCU senses this condition and sends a signal to the CDU which initiates a sequential turn off of non-essential spacecraft loads. When the 28 bus is overloaded, its voltage will drop. If the bus decays to 26.0 $\pm$ 0.5 v for more than 200 $\pm$ 100 milliseconds, the overload sensor will be triggered.

The overload sensor could be triggered due to:

1) RTG degradation, failure of RTG cable, or inverter.

2) Low impedance faults in loads

3) Overload sensor failure

If the RTGs degrade to the point that the load can no longer be supported and the load is not reduced, the battery will discharge. Conceivably, the overload may be light enough that when the battery is depleted the overload may not be tripped. The bus could stabilize between 26.5 and 28 volts. Since RTG degradation is predictable, this condition can be prevented by reducing the load such that sufficient shunt current is present between ground station contacts.

Failure of an RTG, RTG cable, or inverter during the post encounter phase would cause the overload sensor to trigger and reduce the load. Following the overload condition, spacecraft load reductions would be necessary to continue the mission. A low impedance load fault which does not activate the fault isolation circuitry (fuses) would probably trip the overload sensor after battery depletion. This failed load would be identified and removed permanently from the bus through ground command control of the various users (one at a time).

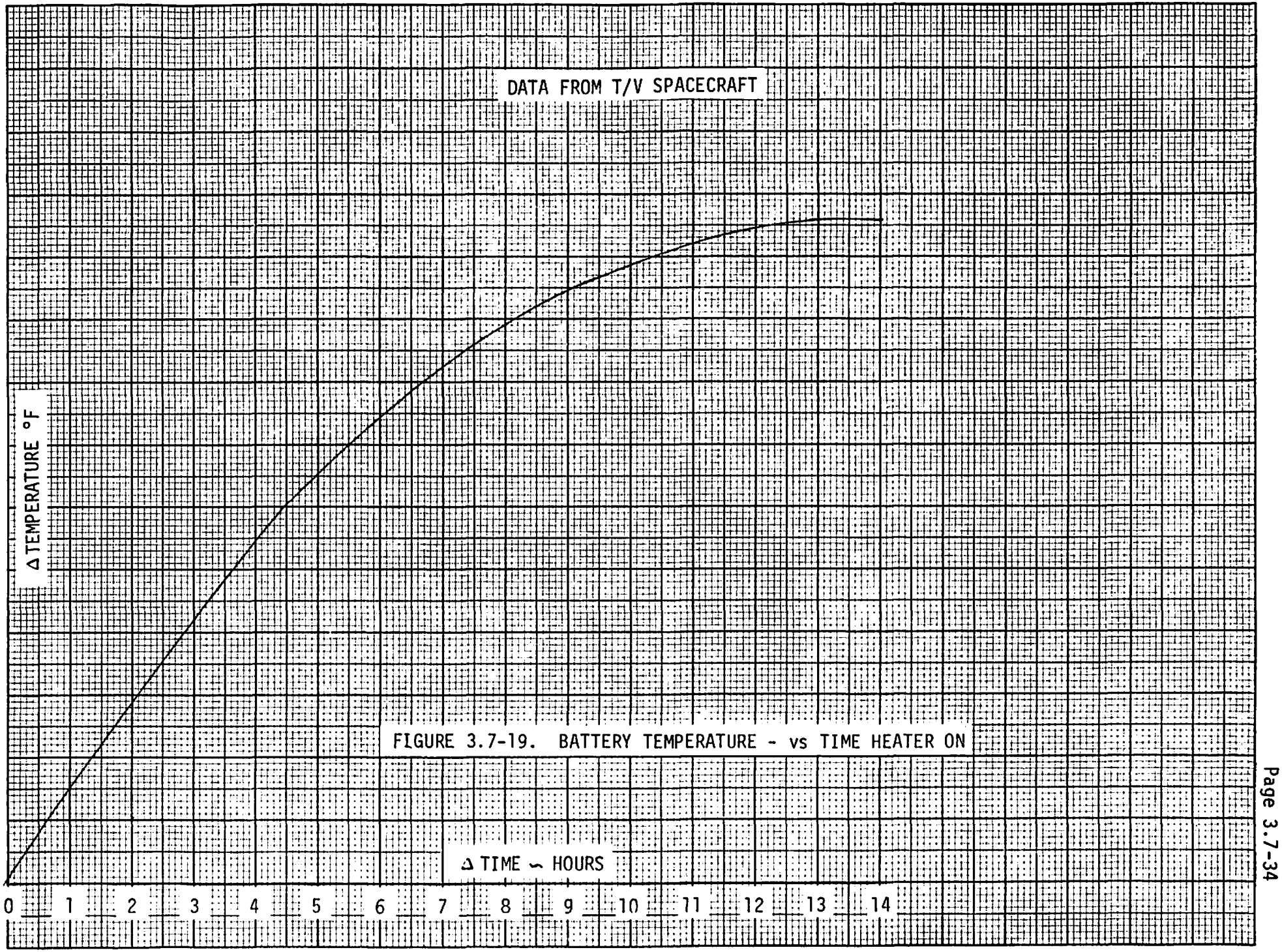


FIGURE 3.7-19. BATTERY TEMPERATURE - vs TIME HEATER ON

The overload sensor could fail and send an erroneous signal to the CDU. The signal can be inhibited by command in the CDU (PSP2).

3.7.3.4 Commands and Telemetry. Tables 3.7-2 and 3.7-3 present summary lists of the power subsystem commands and telemetry sense points. The telemetry list includes the expected normal ranges for each telemetry point and, in some cases, possible causes of off-nominal readings.

3.7.3.5 Failure Mode Operation. The power subsystem incorporates redundancy and fault isolation which preclude catastrophic mission failures due to any single failure in the power subsystem. Table 3.7-4 lists failures which may occur and corrective action that may be taken by ground commands to eliminate or minimize the effect of failures on the spacecraft mission.

Table 3.7-2. Power Subsystem Commands

Command	Octal	Designator	Description
Conscan Power ON	026	CNS9	Turns on power to conscan via CTRF relay closure.
Conscan Power OFF	133	CNS0	Turns off power to conscan via CTRF relay opening.
Xmtr. DVR. A ON	223	TRA9	Turns on xmtr. driver 1 via CTRF relay closure.
Xmtr. DVR. A OFF	232	TRA0	Turns off xmtr. driver 1 via CTRF relay opening.
Xmtr. DVR. B ON	310	TRB9	Turns on xmtr. driver 2 via CTRF relay closure.
Xmtr. DVR. B OFF	331	TRB0	Turns off xmtr. Driver 2 via CTRF relay opening.
DTU Redundancy A ON/B OFF	044	DTA1	Selects DTU 5.0v redundancy A via CTRF relay (power applied to A)
DTU Redundancy B ON/A OFF	155	DTB1	Selects DTU 5.0v redundancy B via CTRF relay (power applied to B)
Battery Auto Charge	043	BAT1	Places battery in automatic charge mode.
Battery Float Charge	164	BAT2	Places battery in float mode, limits charge current to 10 ma.
Discharger Enable	004	BAT3	Allows discharge of battery to bus if required.
Discharger Disable	115	BAT4	Prevents battery discharge to bus.
Battery Heater ON	015	BAT9	Turns on battery heater via relay closure.
Battery Heater OFF	126	BAT0	Turns off battery heater via relay opening.
Relay Reset	204	PSP1	Resets inverter assembly fault isolation relays.
Overload Protection OFF	126	PSP2	
Overload Protection ON	015	PSP3	

Table 3.7-3. Power Subsystem Telemetry List

Word	Measurement	Type	Sensor Range	Normal Range	Remarks
C-105	RTG 2 Current	A	0 to 11 amp	6 to 10 amp	RTG current of zero indicates open circuit, inverter relay should be reset by command.
C-106	Battery Voltage	A	0 to 15v	5.2 to 12.3v	Battery voltage greater than 12.3v could damage battery. Voltage less than 5.2 results in regulated bus out of tolerance condition during discharge.
C-107	DC Bus Voltage (Exp)	A	26 to 30v	27.4 to 28.6	When out of normal range indicates main bus regulation out of spec.
C-109	Battery Charge Current	A	0 to 0.3 amp	0 to 0.3 amp	When current remains at >0.2 amp for extended period and battery temperature rises, could indicate cell failure.
C-110	RTG 1 Voltage	A	0 to 6 vdc	3.9 to 4.5v	Voltage of 6v indicates RTG open circuit. Reset inverter relay by command.
C-113	RTG 4 Voltage	A	0 to 6 vdc	3.9 to 4.5	Same as C-110
C-114	RTG 3 Voltage	A	0 to 11 amp	6 to 10 amp	Same as C-105
C-115	Battery Temperature	A	-20 to +120°F	0 to +70°F	Excessive battery temperature could indicate cell failure if current does not taper. Battery temperature less than +20°F during charge could result in insufficient charge.
C-117	TRF 5.3v Output A	A	0 to 6v	5.3v $\pm$ 3%	When out of tolerance switch to redundant unit.
C-118	TRF 5.3v Output B	A	0 to 6v	5.3v $\pm$ 3%	When out of tolerance switch to redundant unit.
C-119	DC Bus Voltage	A	0 to 30 vdc	28 $\pm$ 2%	Provides additional data during undervoltage condition beyond range of exp. scale monitor (C-107).

Table 3.7-3. Power Subsystem Telemetry List (continued)

Word	Measurement	Type	Sensor Range	Normal Range	Remarks
C-122	Shunt Bus Current	A	0 to 3 amp	0 to 3 amp	Indicates excess RTG power, should be maintained above 0 by load reductions, if necessary.
C-123	RTG 4 Current	A	0 to 11 amp	6 to 10 amp	Same as C-105
C-125	RTG 2 Voltage	A	0 to 6 vdc	3.9 to 4.5 vdc	Same as C-110
C-126	Batt. Disch. Current	A	0 to 10 amp	0 to 10 amp	Battery current should approach 10 amp when battery voltage is near 5.2 volts during maximum discharge.
C-127	RTG 1 Current	A	0 to 11 amp	6 to 10 amp	Same as C-105
C-128	Battery Charge Status (BIT1)	B	Auto/CMD	----	0 = Auto, 1 = Command
C-128	Batt Disch Reg Status (BIT2)	B	Enable/Disable	----	0 = Enable, 1 = Disable
C-129	DC Buss Current	A	0 to 6 amp	1.1 to 4.5	6 amp reading could indicate load fault.
C-131	RTG 3 Voltage	A	0 to 6 amp	3.9 to 4.5v	Same as C-110
C-209	Shunt Bus Current	A	0 to 3 amp	0 to 3 amp	From same sensor as C-122

Table 3.7-4. Summary of Failure Modes for which Corrective Ground Commands may be Utilized

Type of Failure or Anomaly	Method of Detection	Action to be Taken by Ground Station
Inverter, RTG or RTG cable failure	Open circuited RTG, reduced shunt current increased RTG temperature.	Inverter relay reset command may be sent (Com'd 204). However, RTG can be open circuited only a short time without permanent damage. If necessary, reduce load to maintain shunt current greater than zero.
Battery cell failure	Extended period of time during charge at high current (>200ma) battery temperature increase.	Command float mode (Com'd 164), battery charge control by ground command if load duty cycling required.
Excessive battery temperature	Battery temperature telemetry	Command float mode (Com'd 164)
Battery temperature below 0°F	Battery temperature telemetry	Command battery heater on (Com'd 015)
28v bus voltage outside regulation band on high side (shunt failure)	28 vdc bus expended scale telemetry monitor	Increased spacecraft load to to reduce RTG voltage.
28v bus voltage outside regulation band on low side.	28 vdc bus expanded scale telemetry monitor	Reduce spacecraft load.
Excessive PCU temperature	-X axis Hex. platform temperature monitor	If high temperature causing anomalous PCU operation, adjust shunt current by load change to reduce PCU dissipation (see Figure 3.7-17, Section 3.7.3.1).
DTU - A + 5v TRF failure	DTU A inoperative (loss of telemetry	Command switch to redundant CTRF B output (Com'd 155)
DTU - B + 5v TRF failure	DTU B inoperative (loss of telemetry)	Command switch to redundant CTRF A output (Com'd 044)

Table 3.7-4. Summary of Failure Modes for which Corrective Ground Commands may be Utilized (continued)

Transmitter driver failure	Transmitter inoperative	Command switch to redundant CTRF output (Com'ds 223 #1 ON, 232 #1 OFF, 310 #2 ON, 331 #2 OFF)
Discharge regulator failure in enabled mode	Increase in shunt current, decrease in battery voltage, battery discharge current present during shunting.	Disable discharger by command (Com'd 115).
Overload sensor erroneous signal	Overload indication, when turnedon all loads drawing normal power, shunt current greater than zero.	Command override of overload signal (Com'd 246)
DDU A TRF failure	Spacecraft does not respond to command.	Repeat command through DDU B.
DDU B TRF failure	Spacecraft does not respond to command.	Repeat command through DDU A.
Receiver TRF failure	Spacecraft does not respond to command.	Select redundant receiver frequency.
CDU TRF 5.2 vdc failure	Spacecraft does not respond to command, 5.3 vdc telemetry monitor.	Select redundant CDU by ground command. (B ON/A OFF Com'd 315, A ON/B OFF Com'd 262)

SECTION 3.7 NOTATION

A	Analog
AC	Alternating Current
ACS	Attitude Control System
A-H	Ampere-Hours
B	Bi-Level
BAT	Battery
BOL	Beginning of Life
CDU	Command Distribution Unit (Described in Section 3.9)
CEA	Control Electronics Assembly
CMD	Command
CTRF	Central Transformer-Rectifier-Filter
D	Digital
DC	Direct Current
DDU	Digital Decoder Unit (Described in Section 3.9)
DSU	Data Storage Unit (Described in Section 3.5)
DUT	Digital Telemetry Unit (Described in Section 3.5)
EA	Error Amplifier
GFE	Government Furnished Equipment
IA	Inverter Assembly
IFJ	In-flight Jumper
KHz	KiloHertz, cycles per second
ma	milliamps
p-p	peak-to-peak
mv	millivolts
PCU	Power Control Unit
RTG	Radioisotope Thermoelectric Generator (Described in Section 3.8)
RCVR	Receiver
S/C	Spacecraft
TLM	Telemetry
TRF	Transformer-Rectifier-Filter
TWT	Travelling Wave Tube
T/V	Thermal-Vacuum (Test)
U/V	Undervoltage
VRMS	Volts Root Mean Squared
VDC	Volts Direct Current
VD	Voltage Drop
XMTR	Transmitter

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3.8 RTG - RHU DESCRIPTION

3.8.1 General Description of RTG

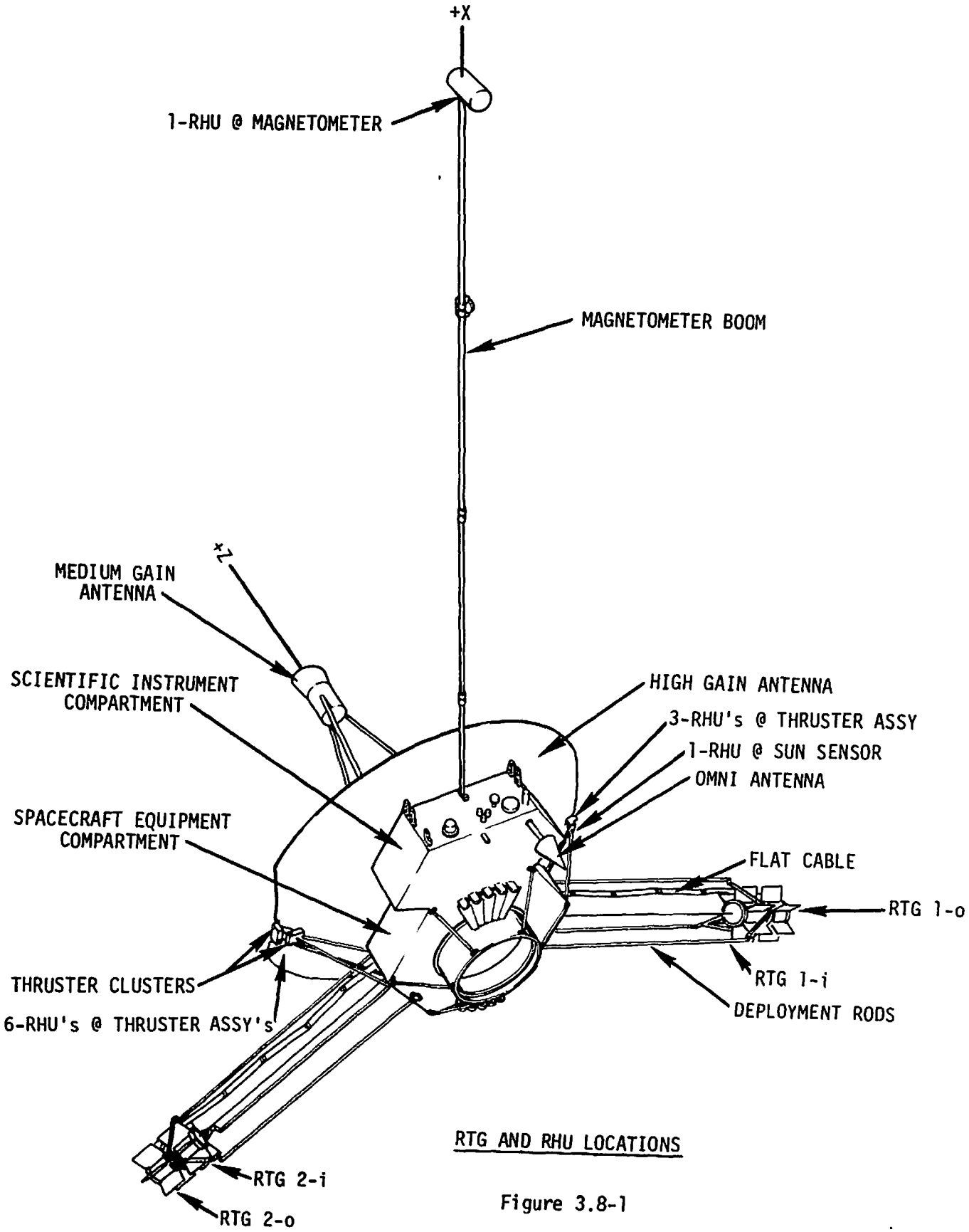
The electrical power supply for the Pioneer F/G spacecraft consists of four SNAP-19 radioisotope thermoelectric generators (RTG's). The four RTG's are grouped in pairs and are electrically connected in parallel at the output of the inverters. After launch, each RTG pair is deployed on extendable booms perpendicular to the spacecraft spin axis and 120° apart. The configuration minimizes the effects of RTG radiation on the spacecraft scientific compartment, and aids in the spacecraft dynamics (see Figure 3.8-1).

An RTG (see Figure 3.8-2) weighs 29.4 pounds nominally (30 pounds maximum) exclusive of bracketry and electrical harnessing. It has an overall length of 11.10 inches and a fintip to fintip diameter of 15.68 inches. The two basic subassemblies consist of the internal nuclear heat source which is fueled with plutonium 238 isotope fuel and the surrounding thermopile assembly. The heat source is designed with an ablative graphite heat shield so that it can survive a possible earth reentry without loss of fuel. The thermopile assembly is composed of six thermoelectric modules containing a total of 90 thermoelectric couples connected electrically in two parallel strings of 45 couples in series. These are all internal to the converter body. The six radial fins on the converter radiate the waste heat away from the RTG to space.

3.8.1.1 Identification. The RTG's are designated by NASA as numbers 1-o, 1-i, 2-o and 2-i on the Pioneer F/G vehicle. The NASA designation "i" stands for "inboard" and "o" stands for "outboard" with respect to the spacecraft Z axis. Figure 3.8-1 shows the location of each RTG. In terms of spacecraft/RTG integration, each RTG is designated by a permanent serial number, and after magnetic measurements have been made, they are assigned to each of the four spacecraft locations. This procedure is an attempt to cancel the magnetic moments (dipole) at any point in space, thereby reducing the B field at the magnetometer detector. Radiation is also a factor in assigning RTG locations. Table 3.8-1 should be filled in, after the RTG's have been assigned to particular spacecraft locations. There are two alternate RTG designations for the RTG's, which are shown on the table.

Table 3.8-1. RTG Identification

NASA ARC Designation for RTG	1-i	1-o	2-i	2-o
TRW Designation for RTG	3	4	1	2
Telemetry Reference Designation	0412	0413	0410	0411
Actual Serial Number of SNAP-19 used	___	___	___	___
Actual Inverter Used With RTG	___	___	___	___

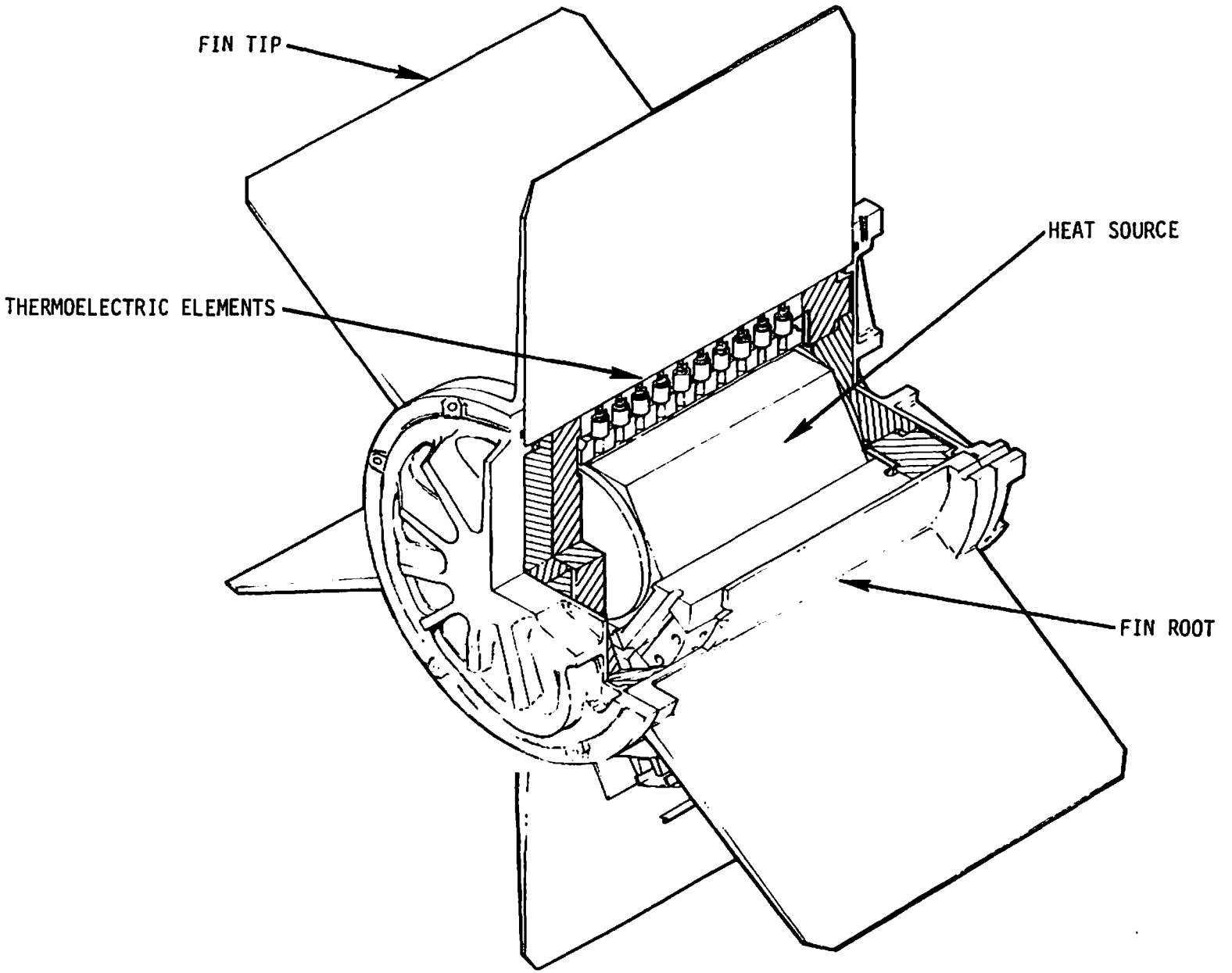


RTG AND RHU LOCATIONS

Figure 3.8-1

SNAP-19 RTG

Figure 3.8-2



3.8.1.2 Interfaces with Spacecraft.

- Mechanical. Each pair of RTGs is mounted to the spacecraft as shown in Figure 3.8-1. There is no mechanical control or adjustment which can be made to the RTGs once they are deployed.

- Electrical. Each RTG pair is connected to the spacecraft via a 26 pin, twist lock type electrical power connector. The RTG supplies DC power to the spacecraft in accordance with the estimated power versus time curves shown in Figures 3.8-3 a, b, c, and d. Operational characteristics in flight may be plotted directly on these curves. At constant load resistance or at constant voltage it is not possible to regulate the amount of power output by an RTG. This is solely a function of the temperature distribution and the electrical characteristics of the converter thermoelectric elements.

- Thermal. Due to the distance between the RTGs and spacecraft after RTG deployment, the thermal interactions during interplanetary flight will be very small. No special provisions have been made to limit this thermal interaction. The design value for the total amount of heat rejection from each RTG while functioning in the space environment is 610W at Beginning of Life (BOL). The temperature of the external surface of the RTG outer housing, end covers, and fin roots is 225-360°F.

- Magnetic. Each pair of tandem mounted RTGs is wired so that the fields resulting from RTG current flow, oppose each other. The magnetic field along the axis of symmetry of an RTG pair and at right angles to that axis, is shown in Figure 3.8-4.

- Nuclear Radiation. Neutrons are produced by (α, n) reactions with low Z impurity elements in the fuel, spontaneous fission in the fuel and neutron induced fission. Figure 3.8-5 shows the Neutron Fluence for the spacecraft, resulting from the deployed RTGs and the RHUs.

  Primary gamma radiation results from radioisotope decay, fission products and prompt fission gammas. Figure 3.8-6 shows the gamma radiation ionizing dose, considerating the RTGs (deployed), the RHUs and the gamma radiation from the planet Jupiter.

3.8.2 General Functional Operation

3.8.2.1 Operating Principles. The RTG provides a continuous source of DC electrical power. Heat from the decay of Pu-238 is emitted from the source. This heat is converted to electricity via the thermoelectric effect in the thermopile assembly of the RTG. There is no mechanism for controlling the amount of heat emitted nor the subsequent current flow load control.

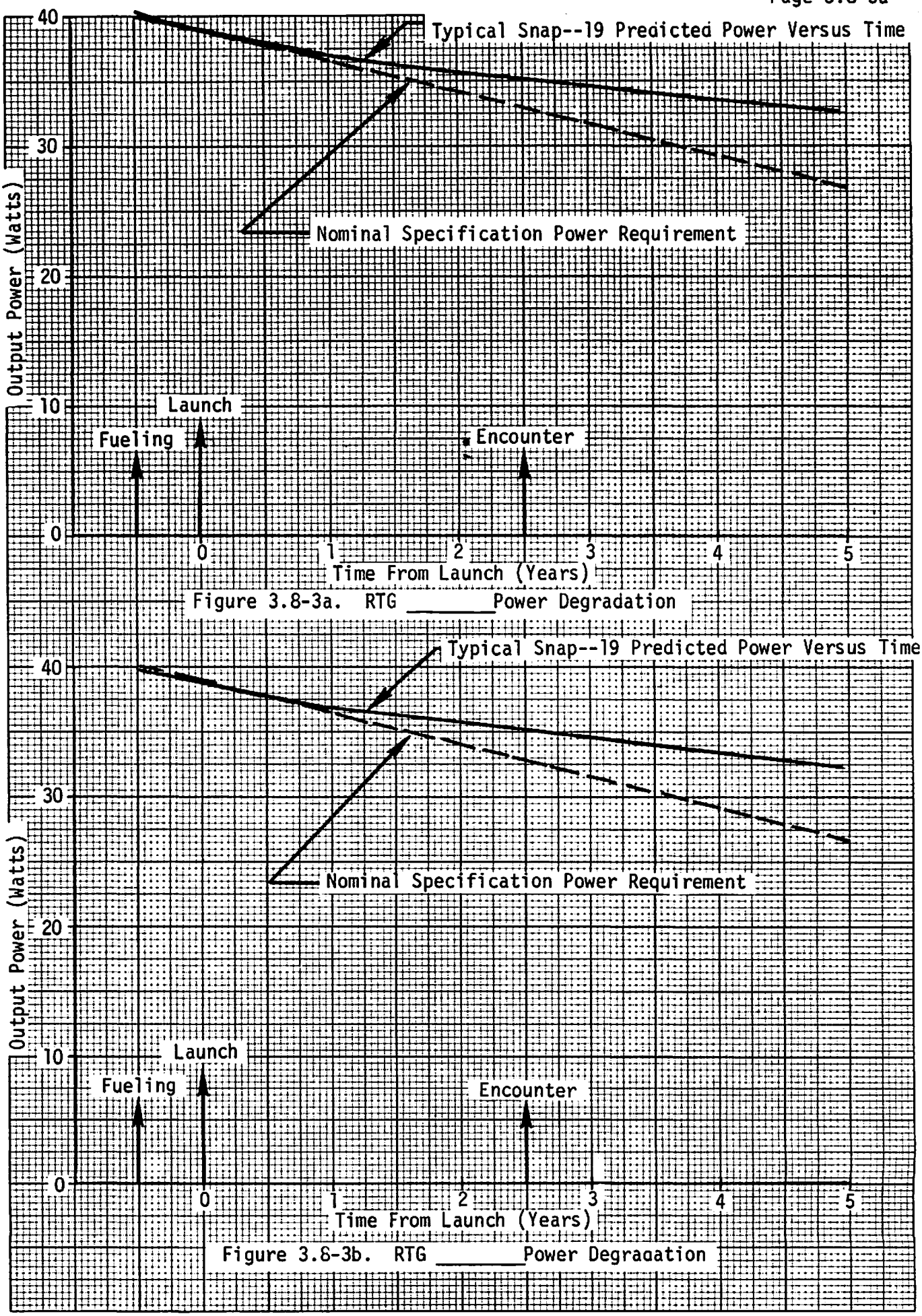


Figure 3.8-3a. RTG _____ Power Degradation

Figure 3.8-3b. RTG _____ Power Degradation

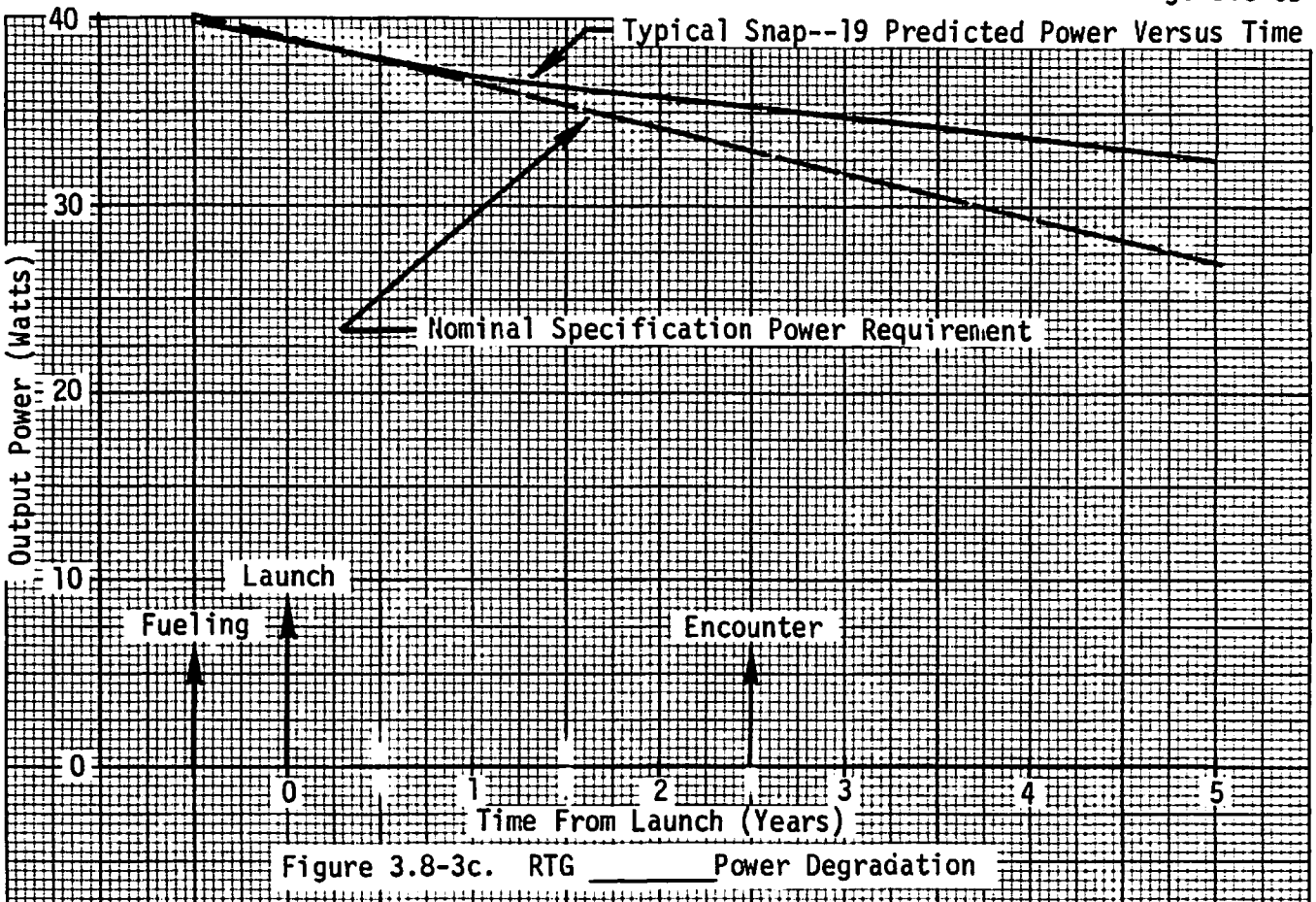


Figure 3.8-3c. RTG _____Power Degradation

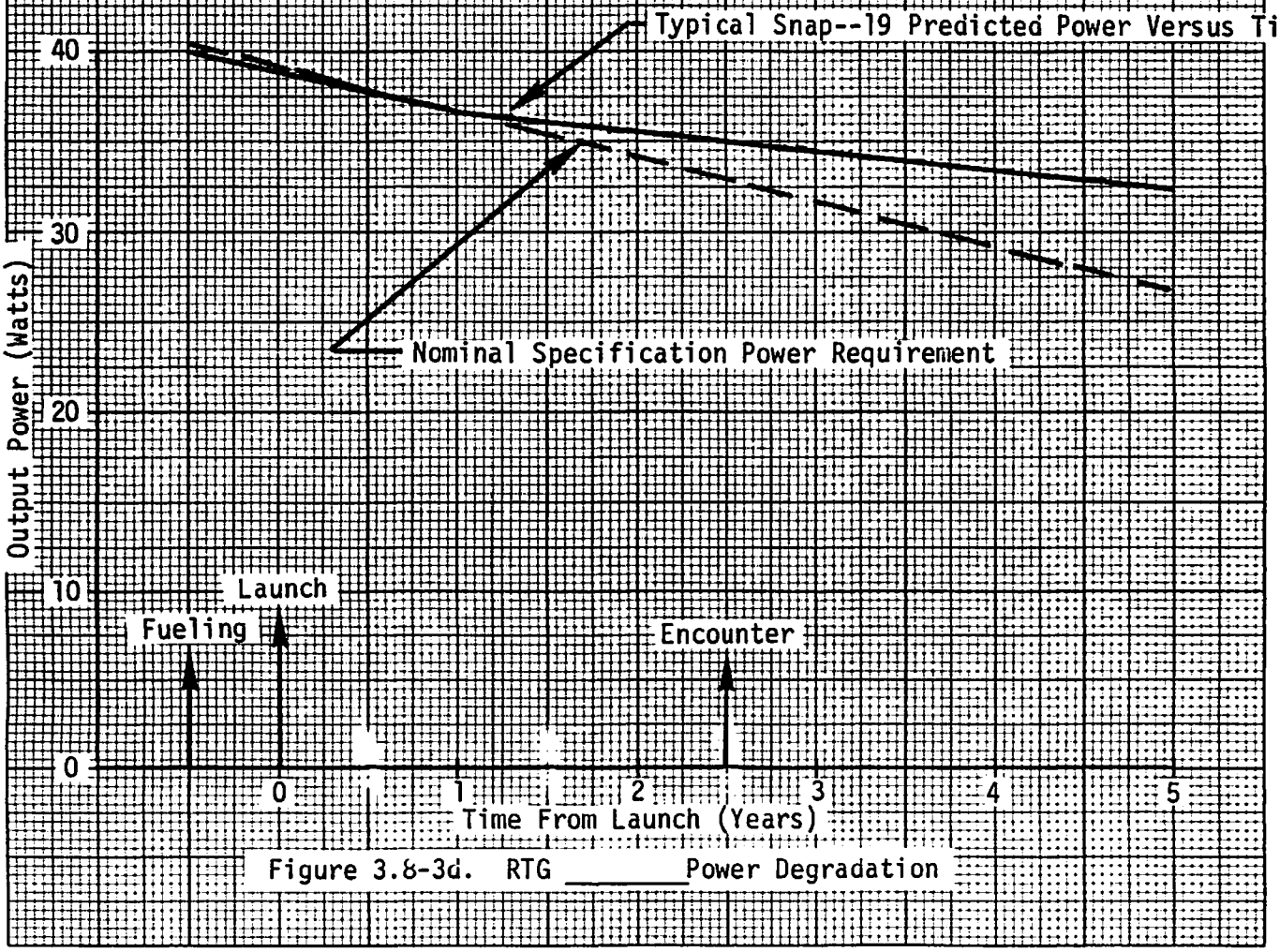


Figure 3.8-3d. RTG _____Power Degradation

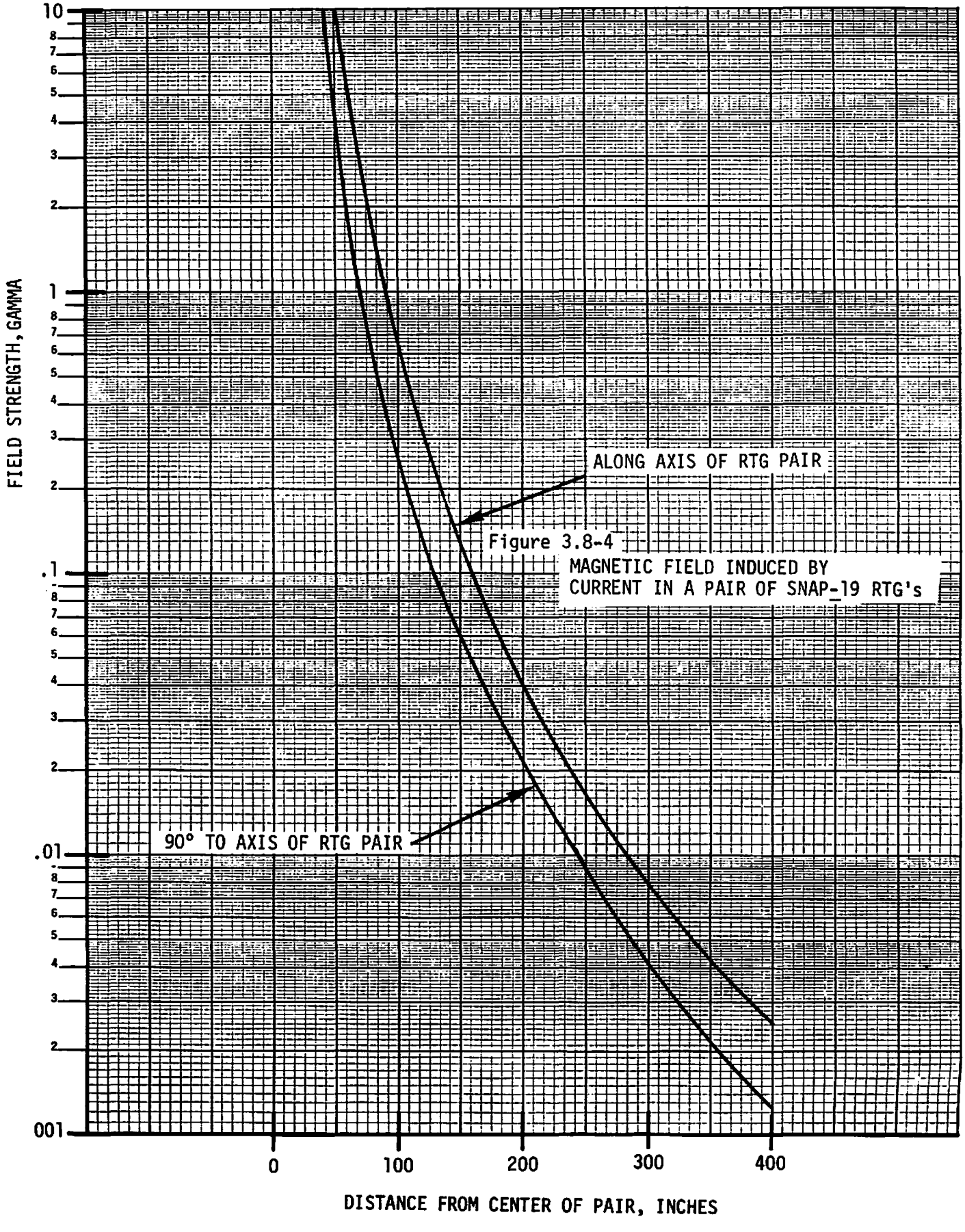


Figure 3.8-4
MAGNETIC FIELD INDUCED BY
CURRENT IN A PAIR OF SNAP-19 RTG's

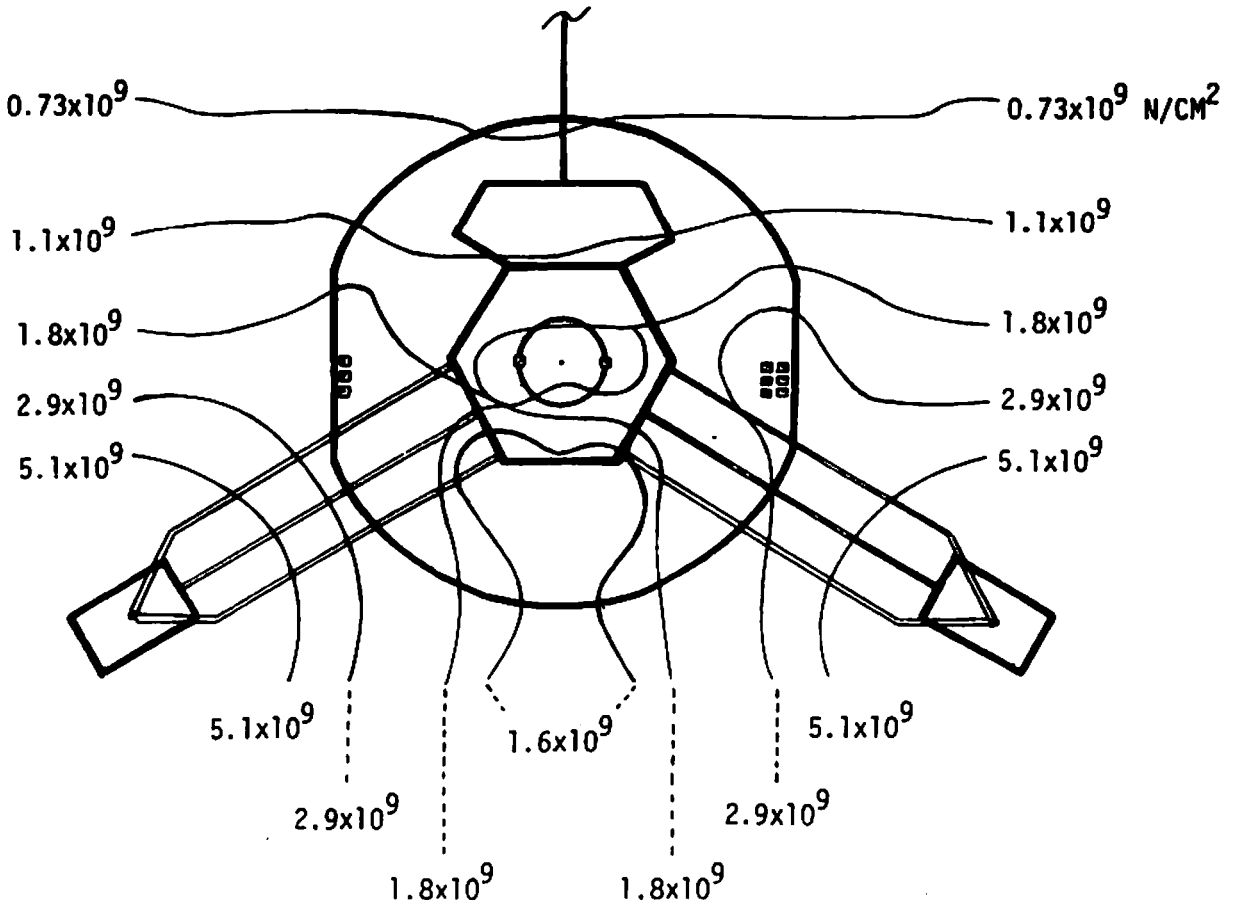
RTG/RHU ENVIRONMENTS
NEUTRON FLUENCE - DEPLOYED (850 DAYS)

Figure 3.8-5

LEGEND:
- RTG PAIR
- RHU

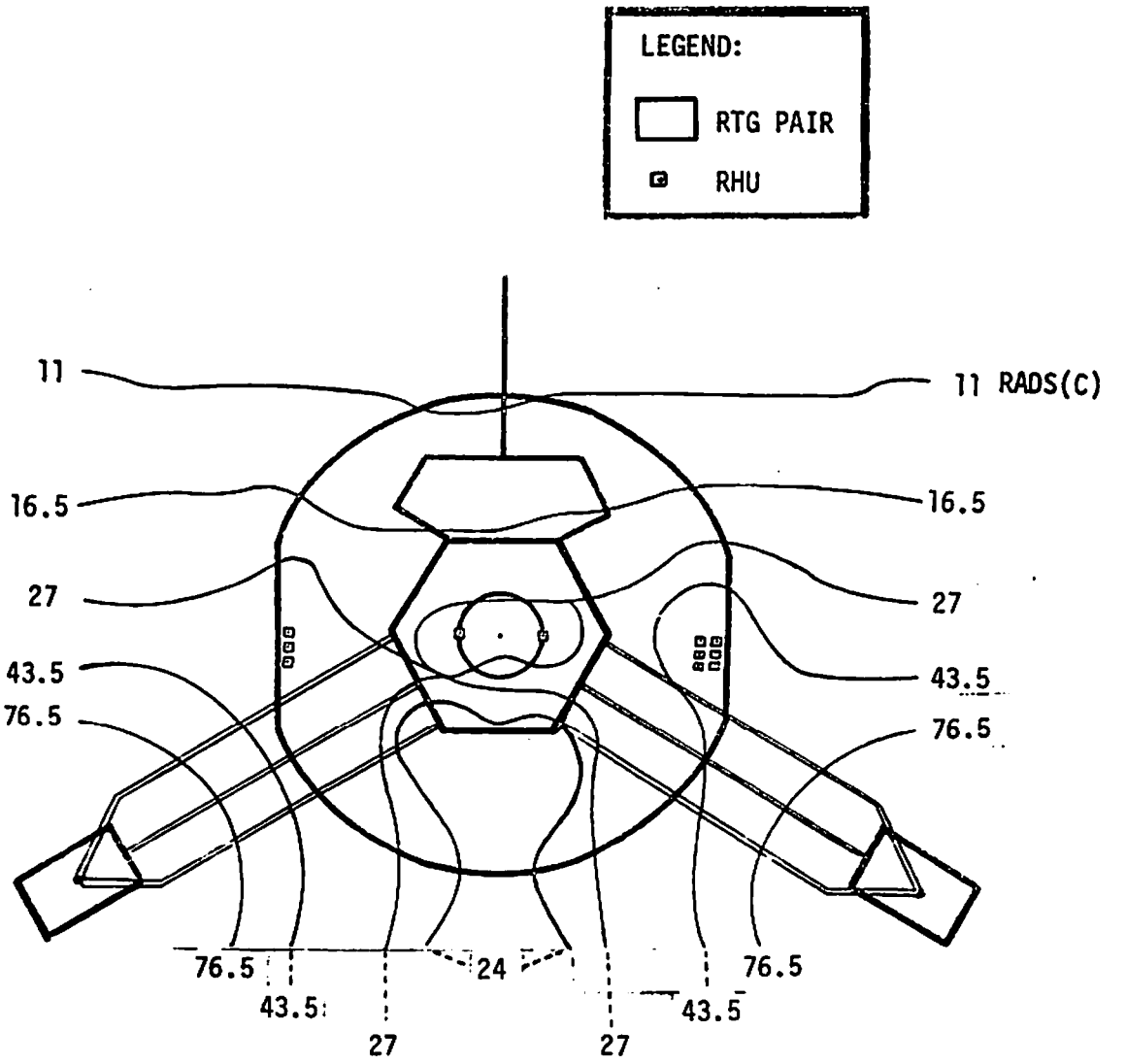
NEUTRON ACTIVITY

Energy (Mev)	Neutrons/Sec g ^{238}Pu Product
0 - 0.5	3.4×10^{2}
0.5 - 1.0	7.0×10^{2}
1.0 - 2.0	5.4×10^{3}
2.0 - 3.0	9.7×10^{3}
3.0 - 4.0	4.3×10^{2}
4.0 - 5.0	4.5×10^{1}
5.0 - 6.0	7.2×10^{1}
6.0 - 7.0	2.4×10^{1}
7.0 - 2.0	1.7×10^{1}
8.0 - 10.0	6.3×10^{0}
10.0-13.0	$3.1 \times .10^{0}$



CALCULATED GAMMA PHOTON INTENSITIES IN Pu-238 DECAY

	Photons/Sec-gm ^{238}Pu	
Energy (Kev)	1 Yrs.	5 Yrs.
17	8.2×10^{10}	7.9×10^{10}
99.8	5.0×10^{7}	4.9×10^{7}
153	6.3×10^{6}	6.1×10^{6}
277	7.8×10^{3}	9.2×10^{4}
511	2.5×10^{3}	3.0×10^{5}
583	9.4×10^{3}	1.1×10^{4}
727	1.4×10^{2}	1.6×10^{3}
785	2.2×10^{3}	2.6×10^{4}
860	1.3×10^{2}	1.6×10^{4}
1075	1.2×10^{2}	1.5×10^{3}
1620	3.5×10^{2}	4.2×10^{3}
2614	1.1×10^{4}	1.3×10^{5}



GAMMA RADIATION

Figure 3.8-6

The electrical power output will decrease over time as a result of degradation of the converter thermoelectric elements, plus fuel inventory loss due to isotope decay.

3.8.2.2 Modes of Operation. The electrical power distribution from the RTGs is shown in Figure 3.8-7. The prime energy source consists of the four RTGs which convert thermal energy directly into electrical energy at a maximum equivalent voltage of 4.2 vdc at maximum power. This voltage is conditioned to a principal distribution voltage of 30.5 vrms at 2.5 kHz in four inverter assemblies. The point of interface between inverters and RTG is considered to be the RTG power output connector. Control of the voltage operating point is accomplished indirectly by the action of the shunt regulator, which is part of the Power Control Unit (Section 3.9).

Actually the equivalent RTG voltage deviates from a true DC voltage since the signal is being operated at 2.5 kHz in an approximate square wave shape from zero to some peak value. The DC RMS value of the waveform is 4.2 volts. The effect of this deviation on the power output is negligible.

There are three definable operating modes of the RTGs to which the Electrical Power System, RTG cables, and Inverter Assemblies are designed to operate. They are:

a) Mode I (Maximum Power)--Beginning of Life; total power is approximately 33 watts per RTG x 4 = 160 watts.

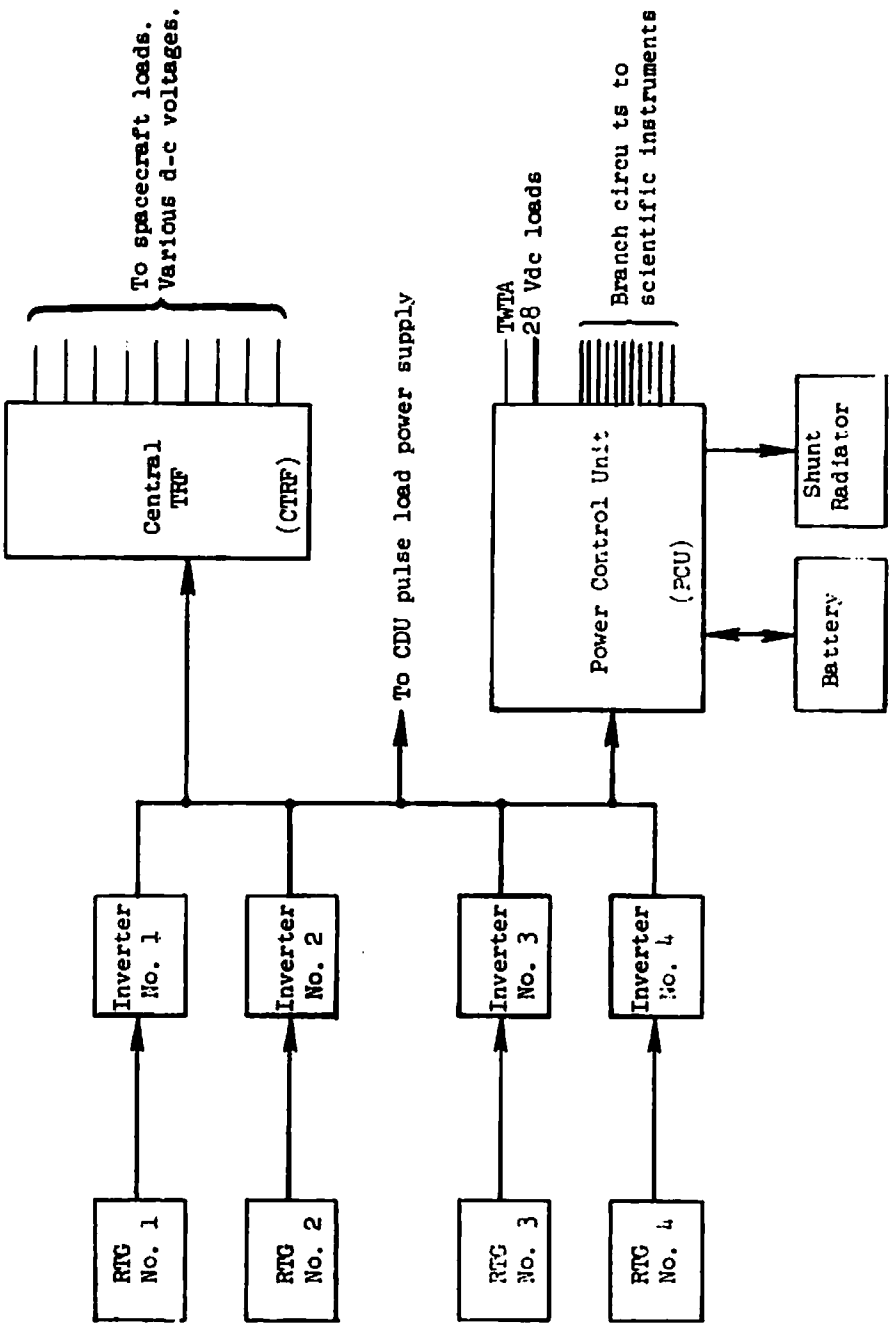
b) Mode II (Minimum Power)--Jupiter Encounter; total power is approximately 33 watts per RTG x 3 = 99 watts.

c) Mode III (Intermediate Power)--Degraded Operation of 4 RTGs; total power is between 160 watts and 99 watts.

Note that the Output Voltage is 4.2 $\pm$ 7.0% DC for all of the above modes.

The computed values of power output should be plotted on the degradation of power over time curves (Figures 3.8-3a through 3.8-3d). The curves are presented in a format suitable for continuous plotting of power. Fin root temperature, hot junction temperature, RTG voltage and RTG current to the inverter are available for each RTG from the telemetry data. Figure 3.8-8 shows typical anticipated hot junction temperature from telemetry data. This can then be compared to the nominal expected degradation.

The electrical power output of the SNAP-19 RTG is dependent on the RTG operating voltage and fin root temperature. Variation of these parameters during the flight from the nominal value of 4.2 volts and 342°F, or variation of the BOL parameters will affect the power output. Additional considerations are the effect of the decay of the isotope fuel and thermoelectric degradation. To estimate changes in electrical power output normalized to a referenced load voltage (BOL or 4.2 volts and 342°F or 352°F root temperature respectively), see the sample calculation on Pages 3.8-12 through 3.8-15.



ELECTRICAL POWER SUBSYSTEM BLOCK DIAGRAM

Figure 3.8-7

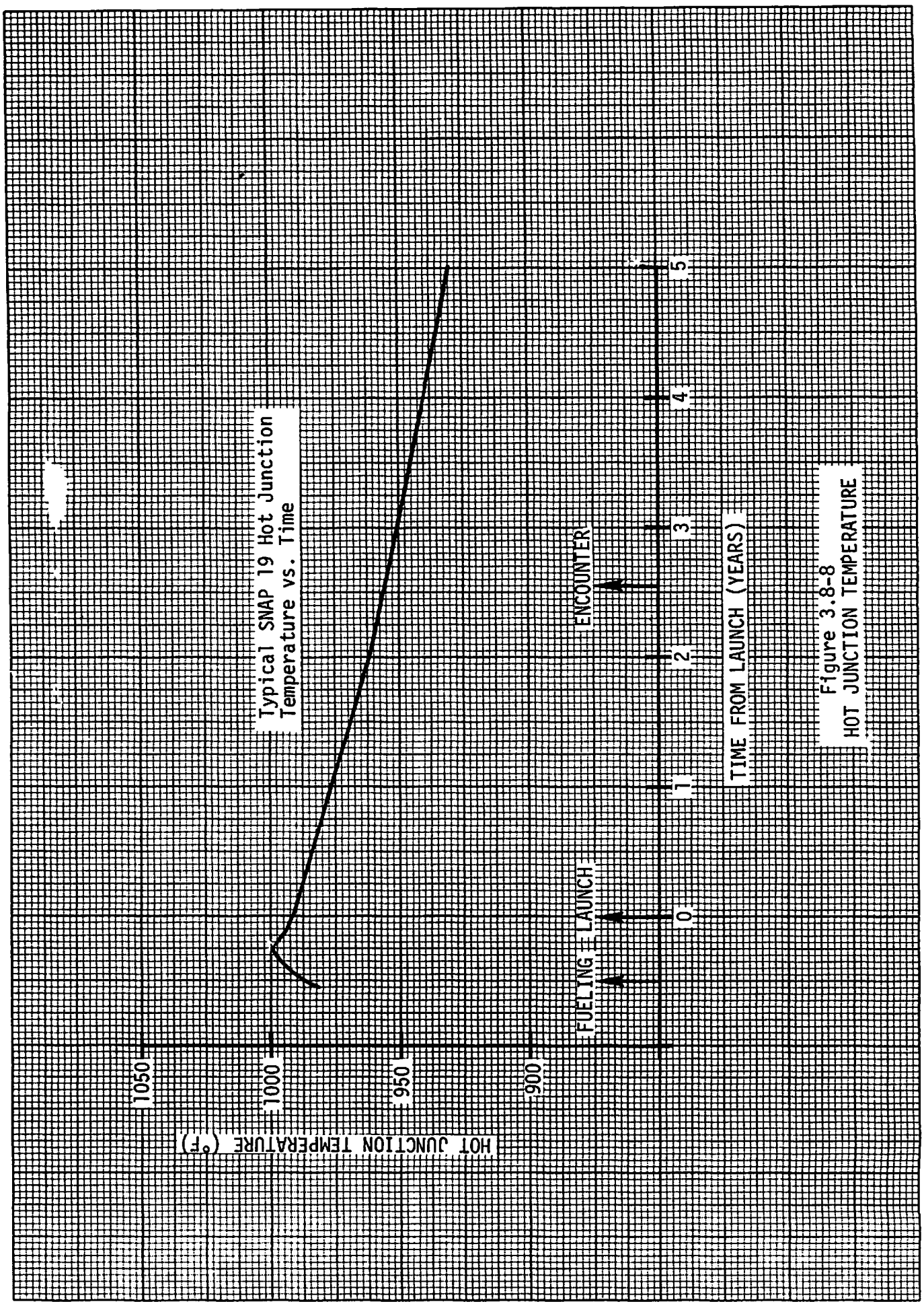


Figure 3.8-8
HOT JUNCTION TEMPERATURE

The variation in electrical output, at 4.2 volts DC, for fin root temperatures between 240°F and 330°F, is -0.030 watts/°F. That is, an increase of one °F in fin root temperature causes a decrease in power output of 0.030 watts. It is anticipated that inboard RTGs will have a fin root temperature approximately 15°F hotter than outboard RTGs and that the variation between launch and encounter will be a decrease of approximately 10°F. This implies that the inboard RTG will have approximately 0.45 watts less power output than the outboard RTG, and that approximately 0.30 watts power output will be gained due to the decrease in fin root temperature. Unless greater temperature variation is experienced during flights, these effects should be relatively small compared to telemetry accuracy and/or long-term degradation effects.

The anticipated degradation effect of a single SNAP-19 is approximately 0.2 watts/month (see Figure 3.8-4).

The load voltage is expected to vary during the flight over the range 4.2 volts $\pm 7\%$ (± 0.29 volts) from launch to encounter. To determine the RTG output power, the inverter input power can be calculated from telemetry. Correction for RTG flat cable power loss can be made by using the nominal cable resistances (≈0.018 ohms) for each flat cable assembly. The anticipated RTG output voltage will vary from a nominal operating point of 4.22 volts at launch to 4.19 volts at encounter. There are additional uncertainties associated with power conditioning equipment component tolerances. These uncertainties result in the following anticipated operating bands for the RTG voltages:

	Anticipated Operating Range of RTG Voltage		
Time of Flight	Nominal	Maximum	Minimum
Launch	4.22	4.37	3.99
Encounter	4.19	4.22	3.83

The variations in RTG operating voltages are anticipated to result in a power output variation between 0.1 watts to 0.5 watts. In general, this effect can be ignored relative to the power output changes due to thermoelectric converter degradation over long periods of time.

An example of a typical in-flight power calculation is given below:

Assume the following data is obtained from the telemetry system after data reduction:

Measurement	Value
#1 RTG Fin Root Temperature	340°F
#2 RTG Fin Root Temperature	325°F
#3 RTG Fin Root Temperature	351°F
#4 RTG Fin Root Temperature	335°F
#1 RTG Voltage	4.06 Volts
#2 RTG Voltage	4.05 Volts
#3 RTG Voltage	3.81 Volts
#4 RTG Voltage	3.69 Volts
#1 RTG Current	7.66 Amps
#2 RTG Current	8.22 Amps
#3 RTG Current	7.82 Amps
#4 RTG Current	8.38 Amps

The values for the parameters should be plotted on graphs to show the telemetry data as a function of time. The inverter input power should be computed as follows:

No. 1 Inverter Input Power = 7.66 x 4.06 = 31.1 watts
No. 2 Inverter Input Power = 8.22 x 4.05 = 33.3 watts
No. 3 Inverter Input Power = 7.82 x 3.81 = 29.8 watts
No. 4 Inverter Input Power = 8.28 x 3.69 = 30.9 watts

Next, the RTG power should be computed by adding the cable power loss. In the event individual cable resistances are not available an average value of 0.018 ohms can be used for this calculation.

No. 1 RTG Power = $(7.66)^2$ (0.018) + 31.1 = 32.2 watts
No. 2 RTG Power = $(8.22)^2$ (0.018) + 33.3 = 34.5 watts
No. 3 RTG Power = $(7.82)^2$ (0.018) + 29.8 = 30.9 watts
No. 4 RTG Power = $(8.38)^2$ (0.018) + 30.9 = 32.1 watts

If desired, the power correction to a nominal fin root temperature of 342°F (inboard) and 357°F (outboard) can be calculated as follows: (Note telemetry readings are 10°F below average fin root temperature. Add 10°F to telemetry readings to get average fin root temperatures.) Influence coefficient is -0.03 watts/°F fin root.

No. 1 RTG = (340-347) (-0.03) = +0.2 watts
No. 2 RTG = (325-332) (-0.03) = +0.2 watts
No. 3 RTG = (351-347) (-0.03) = -0.1 watt
No. 4 RTG = (335-332) (-0.03) = -0.1 watt

The corrected RTG power at nominal RTG fin root temperature, would be:

No. 1 RTG = 32.2 + 0.2 = 32.4 watts
No. 2 RTG = 34.5 + 0.2 = 34.7 watts
No. 3 RTG = 30.9 - 0.1 = 30.8 watts
No. 4 RTG = 32.1 - 0.1 = 32.0 watts

The RTG voltage is computed by adding the cable drop to the measured inverter input voltage.

No. 1 RTG Voltage = (0.018) (7.66) + 4.06 = 4.20 volts
No. 2 RTG Voltage = (0.018) (8.22) + 4.05 = 4.20 volts
No. 3 RTG Voltage = (0.018) (7.82) + 3.81 = 3.95 volts
No. 4 RTG Voltage = (0.018) (8.38) + 3.69 = 3.84 volts

Corrections of RTG power to a nominal operating voltage will be examined during detailed flight analysis of the RTG. The telemetry data and the corrected power and voltage for the inverters and RTGs should then be recorded and plotted as a function of time.

3.8.3 Outputs

3.8.3.1 Telemetry Sensor Signals. Four telemetry signals carry information relative to the performance of each of the four RTGs. There is one inverter assembly associated with each RTG pair, and there are two inverters per each inverter assembly. Reiterating, there are four RTGs, two inverter assemblies, four inverters and sixteen telemetry signals.

Table 3.8-2 shows the signals for a typical RTG. The range of each signal is 0 to 3 volts.

Table 3.8-2. Telemetry Signals For Typical RTG

Measured Quantity	Calibration
RTG Current to Inverter	2.7 volts per 10 amps
RTG Voltage	3 volts per 6 volts (@ inverter)
Hot Junction Temperature	0 to 3 volts = 880°F to 1200°F
Fin Root Temperature	0 to 3 volts = 160°F to 360°F

A summary of RTG telemetry signals showing word numbers ranges, and measurement identifiers is presented in Table 3.8-3. It should be noted that 10°F must be added to the telemetry value of "fin root temperature" in order to obtain the average value of the fin root temperature.

3.8.4 Anomaly Detection

The possible failure modes of an RTG while in the deployed,or flight condition are summarized in Table 3.8-4.

Table 3.8-3. RTG Pioneer Telemetry

Measurement	Word	Range	Measurement Identifier	Location
#1 RTG Fin Root Temperature	C-201	160 to 360°F	T059	Internal to RTG, approximately 2.6 inches away from connector end of RTG. Located on inside of cylindrical shell, directly under an RTG fin. (Second fin clockwise from connector looking at connector end of RTG.)
#2 RTG Fin Root Temperature	C-202	160 to 360°F	T060	
#3 RTG Fin Root Temperature	C-203	160 to 360°F	T058	
#4 RTG Fin Root Temperature	C-204	160 to 360°F	T055	
#1 RTG Hot Junction Temperature	C-220	880 to 1200°F	T061	Internal to RTG, approximately 2.6 inches away from connector and midway between fins. Measures hot junction at corner of heat block which is internal to RTG.
#2 RTG Hot Junction Temperature	C-219	880 to 1200°F	T062	
#3 RTG Hot Junction Temperature	C-218	880 to 1200°F	T056	
#4 RTG Hot Junction Temperature	C-217	880 to 1200°F	T057	
#1 RTG Voltage	C-110	0-6 VDC	T042	Input to Inverter Assembly
#2 RTG Voltage	C-125	0-6 VDC	T043	Input to Inverter Assembly
#3 RTG Voltage	C-131	0-6 VDC	T044	Input to Inverter Assembly
#4 RTG Voltage	C-113	0-6 VDC	T045	Input to Inverter Assembly
#1 RTG Current	C-127	0 to 10 Amps	T036	Input to Inverter Assembly
#2 RTG Current	C-105	0 to 10 Amps	T037	Input to Inverter Assembly
#3 RTG Current	C-114	0 to 10 Amps	T034	Input to Inverter Assembly
#4 RTG Current	C-123	0 to 10 Amps	T035	Input to Inverter Assembly

Table 3.8-4.

Condition	Effect On Spacecraft	How Detected (Per RTG)	Possible Remedial Action
Degradation of thermoelectric elements	Decreased power	Lowered input current to inverter assembly	None
Loss of Argon Gas	Decreased power	Initial increase followed by rapidly decreasing input current to inverter assembly	None
Complete electrical shorting of thermopile assembly circuitry	Decreased power	Input current to inverter assembly drops to zero	None
Open circuit	Decreased power	Input current to inverter assembly drops to zero	None
Shorted couples	Decreased power	Lowered input current	None

A short circuit would actuate a relay at the particular inverter assembly paired with the malfunctioning RTG. This opens the circuit to the malfunctioning RTG. Automatically, an inhibit signal would then be sent to the other three RTGs so they retain their power producing capability. The malfunctioning RTG is permanently lost at this point. An open circuited RTG would cause the following changes in the telemetry data:

- The shunt current would decrease
- The inverter input current for the open circuited RTG would become zero.
- The hot junction temperature of the open circuited RTG would increase.
- The fin root temperature of the open circuited RTG would increase slightly.

Any malfunction of an RTG, less catastrophic than a short circuit, would be manifested by:

- Decrease in inverter input current of the malfunctioning RTG, below that predicted by normal RTG degradation versus time curves.

- Changes in the hot junction and fin root temperatures for the malfunctioning RTG.

3.8.5 General Description of Radioisotope Heater Unit

In order to reduce the total spacecraft electrical power requirement, eleven RHUs are utilized to provide heat to critical components. The locations of the RHUs are as follows: (see Figure 3.8-1.)

3 RHUs at each of the three Thruster Assemblies	= 9
1 RHU at the magnetometer	= 1
1 RHU at the sun sensor	= 1
	11

The spin/despin thruster cluster and the velocity/precession thruster cluster use two RHUs to heat the valves and one to heat the thrust chambers. All thruster RHUs are mounted in the tube which runs between the two thruster assemblies in each cluster.

A cutaway section of a typical RHU is shown in Figure 3.8-9. Each RHU, fueled with Pu-238 O_2 cermet (Plutonia-Molybdenum Cermet) fuel, produces 1 thermal watt. The external dimensions are 0.87 inches in diameter by 1.85 inches in length and the weight is 0.12 pounds.

- Performance. Each RHU is required to deliver from 0.925 to 1.025 watts for six years. The expected lifetime is seven years.

- Output. Due to the simplicity of its construction, it is essentially impossible for an RHU to malfunction. There is no direct telemetry data output which gives information relative to the RHU performance. If RHU malfunction occurred it would most likely be explosive and there would be no way to restore the RHU.

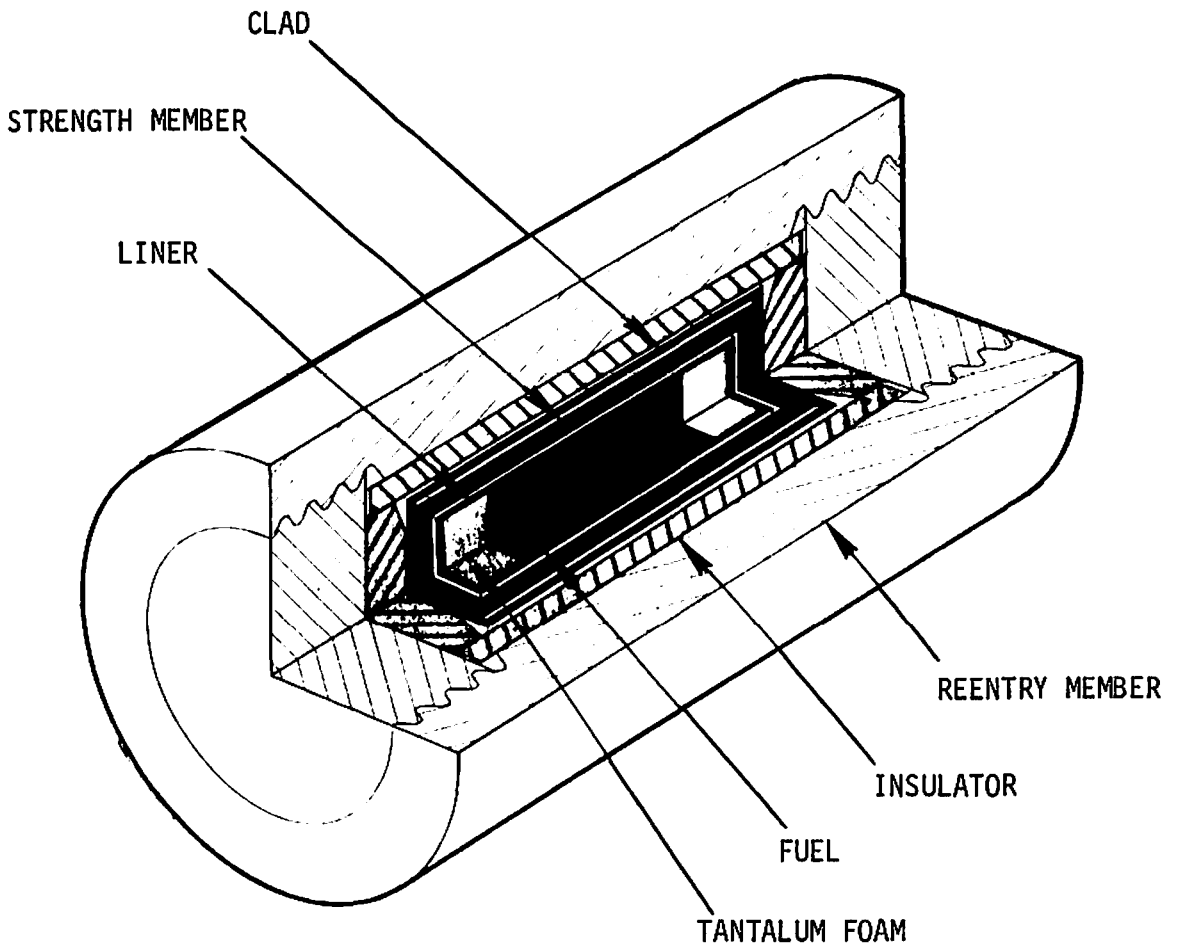


Figure 3.8-9. One Watt Radioisotope Heater Unit

SECTION 3.8 NOTATION

ARC	Ames Research Center
BOL	Beginning of Life
CDU	Command Distribution Unit (Described in Section 3.9)
CTRF	Central Transformer-Rectifier-Filter (Described in Section 3.7)
DC	Direct Current
KEV	Electron Volts x 10^3
KHz	KiloHertz
MEV	Electron Volts x 10^6
N/CM^2	Neutrons per square centimeter
PCU	Power Control Unit (Described in Section 3.7)
Pu	Plutonium
RHU	Radioisotope Heater Unit
RMS	Root mean squared
RTG	Radioistope Thermoelectric Generator
TWTA	Travelling Wave Tube Amplifier (Described in Section 3.6)
VDC	Volts, Direct Current
VRMS	Volts Root Mean Squared

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3.9 ELECTRICAL DISTRIBUTION SUBSYSTEM

3.9.1 Description of CDU and Harness

The Electrical Distribution Subsystem consists of the Command Distribution Unit (CDU) and the spacecraft harness. The purpose of the CDU is to process, store and route to other subsystems and experiments ground commands which are provided to the CDU via the spacecraft antenna, receiver and digital decoder unit. The purpose of the spacecraft harness is to distribute required electrical power, data signals and commands among spacecraft units and experiments. The total spacecraft harness is made up of eleven individual harnesses. These include the following:

- The main interconnection harness of spacecraft units and experiments.
- The harness to the meteroid experiment under the antenna dish.
- The ordnance harness.
- The battery cable.
- The +Y outrigger harness to the sun sensor, thruster cluster and propellant line heaters.
- The -Y outrigger harness to two thruster clusters, and the propellant line heaters.
- The shunt regulator cable.
- Two ordnance in-flight jumpers to provide test panel access to ordnance circuitry.
- A rapid command, in-flight jumper to allow speed up of commands to the spacecraft during test and a battery oscillator in-flight jumper for use when RTGs are being connected or disconnected.
- An in-flight jumper to provide a means of controlling the oscillators in the two inverters when RTGs are being connected or disconnected.
- The RTG power cables to carry the DC power from the RTGs to the inverters.
- The Magnetometer Boom Harness to carry signals and power from Magnetometer Electronics to the sensor.

The functions and operations associated with the CDU will be discussed in the paragraphs that follow.

3.9.1.1 General Functional Operation of CDU. The general purpose of the CDU of the Electrical Distribution Subsystem is to functionally integrate the various subsystems of the spacecraft. Integration is primarily accomplished by the CDU distributing commands. The CDU accepts a digitally coded command signal from the Digital Decoder Unit (DDU), decodes, processes, and distributes the signal as a serial command or a discrete command. In addition to the real time processing of a command, the CDU has provisions to store up to five (5) commands and their related delay times. Special functions required by the spacecraft operation are also provided, i.e., sequencing of commands upon a spacecraft separation signal, ordnance firing, signal conditioning, etc. The full complement of functions provided by the CDU is listed and discussed in the following paragraphs.

3.9.1.1.1 CDU Functions. The CDU is designed to perform the following functions:

a) Accept, process, and distribute command signals from/to various units of other interfacing subsystems (real time command processing).

b) Provide for the storing of up to five (5) commands to be executed at precise time intervals (also stored) after the unit receives an "initiate" ground command signal (command memory).

c) Provide a sequencer operation which activates a sequence of commands at preset time intervals after the unit receives a signal upon separation of the spacecraft (sequencer).

d) Provide means of sequentially turning off power to various subsystems in the event of an undervoltage condition of the primary bus (overload control).

e) Provide for a signal present detector to automatically switch spacecraft receiver inputs when the signal is not present at the receiver outputs (signal present detection).

f) Provide for a thruster firing counter to detect and store the number of firings of a thruster engine, and to route the stored information to the DTU (Thruster Firing Counter).

g) Provide for the firing of the spacecraft ordnance devices (Ordnance Firing Operation).

h) Provide for bi-level and analog signal conditioning of telemetry signals not compatible with Digital Telemetry Unit (DTU) input requirements (Telemetry Signal Conditioning).

i) Provide "OR-ing" and overvoltage sensing capabilities of the redundant +5VDC power input voltages (+5.3V OR Switching and Overvoltage Sensing).

j) Provide initialization of the bi-stable functions during initial application of power to the unit and during momentary outage of the power input (Power Reset)

3.9.1.1.2 CDU Functional Block Diagram. The CDU Functional Block Diagram is presented in Figure 3.9-1. Each function as listed in Paragraph 3.9.1.1.1 is depicted in a simplified functional block form and will be discussed in detail in the following paragraphs.

3.9.1.1.3 CDU Modes of Operation. A brief description of the functions' mode of operation is given in Table 3.9-1. The primary purpose of Table 3.9-1 is to present each function's operating time and show their independence of operation from each other. Except for the command memory, thruster firing counting, signal conditioning, and +5.3V OR switching and overvoltage sensing functions, the circuits are power gated and only dissipate the power listed when operating.

3.9.1.2 CDU Operating Description

3.9.1.2.1 Real Time Command Processing

3.9.1.2.1.1 Command Processing Description. The function of the real time command processor is to decode, process, and distribute real time commands upon receiving signals from the Digital Decoder Unit (DDU). The signals from the DDU consist of the following (see Figure 3.9-2 for the timing diagram).

1) Enable Pulse

2) Routing Address Clock

3) Command Data Clock

4) Command Data (Serial NRZ Signal)

5) Execute Signal

The Command Processor decodes, processes, and distributes real time commands of three types. They are:

1) Serial Commands (routed to the digital telemetry unit, attitude control subsystem, and command memory--see Paragraph 3.9.1.2.2).

2) Discrete Pulse Commands

3) Discrete State Commands

3.9.1.2.1.2 Real Time Command Processing Operations (see Figure 3.9-3)

Real time Command Processing is first initiated by the receipt of the Enable Pulse (see Figure 3.9-2). The Enable Pulse activates the Power Gating Switch which supplies power to the rest of the Command Processor decoding

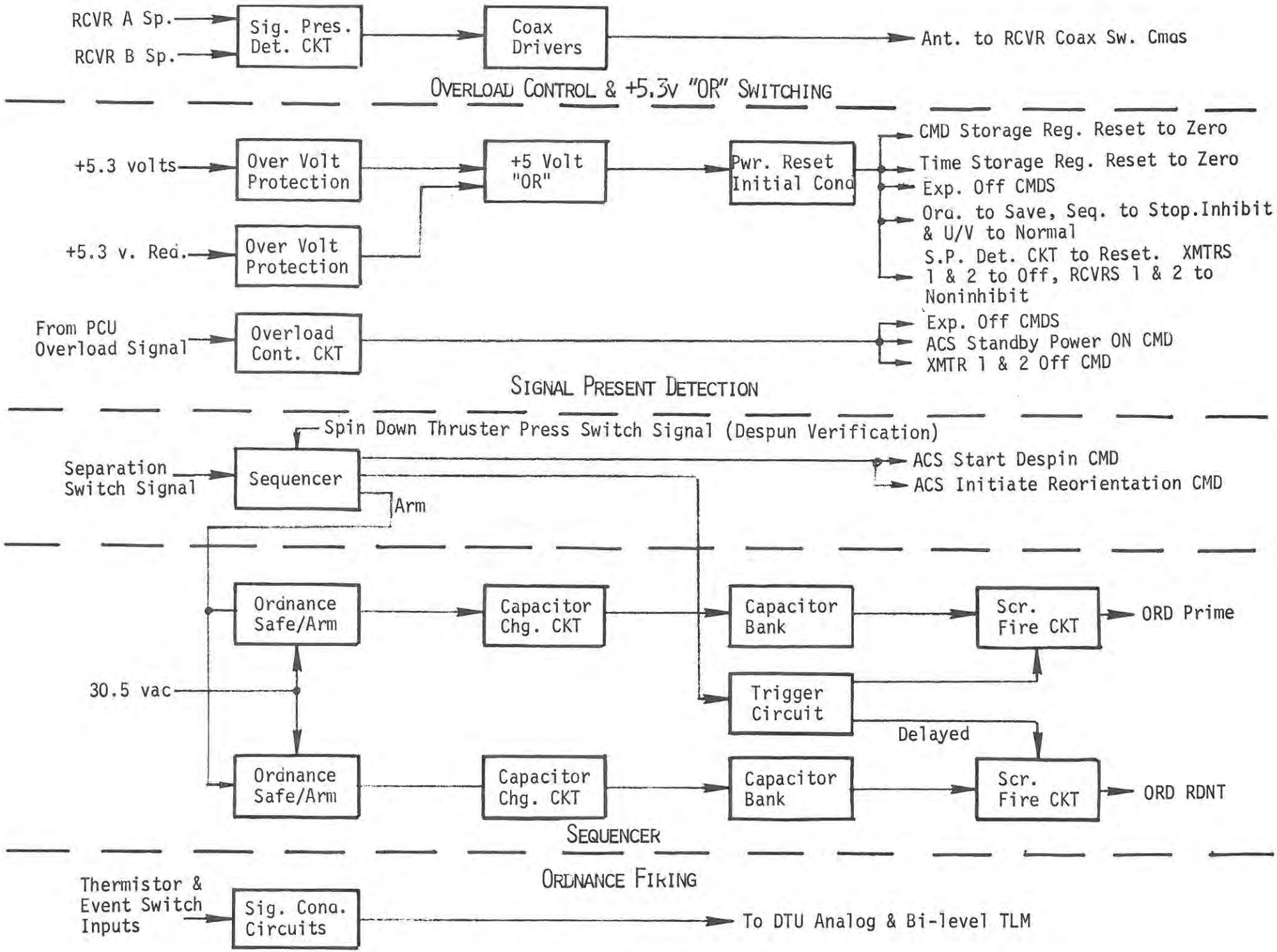
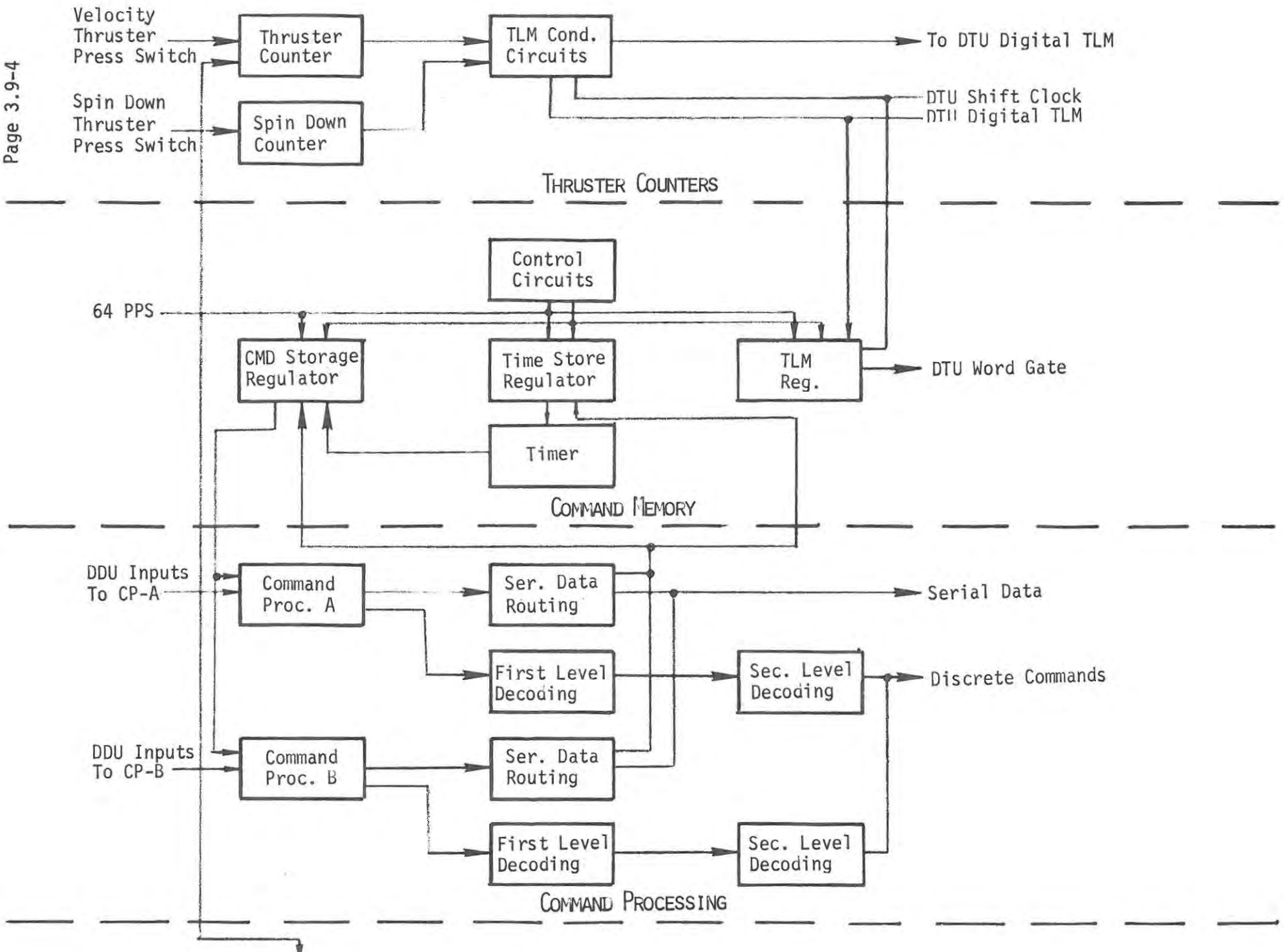


FIGURE 3.9-1. CDU FUNCTIONAL BLOCK DIAGRAM


Velocity Thruster Press Switch
Thruster Counter
TLM Cond. Circuits
To DTU Digital TLM
Spin Down Thruster Press Switch
Spin Down Counter
DTU Shift Clock
DTU Digital TLM
THRUSTER COUNTERS
Control Circuits
64 PPS
CMD Storage Regulator
Time Store Regulator
TLM Reg.
DTU Word Gate
Timer
COMMAND MEMORY
DDU Inputs To CP-A
Command Proc. A
Ser. Data Routing
Serial Data
First Level Decoding
Sec. Level Decoding
Discrete Commands
DDU Inputs To CP-B
Command Proc. B
Ser. Data Routing
First Level Decoding
Sec. Level Decoding
COMMAND PROCESSING

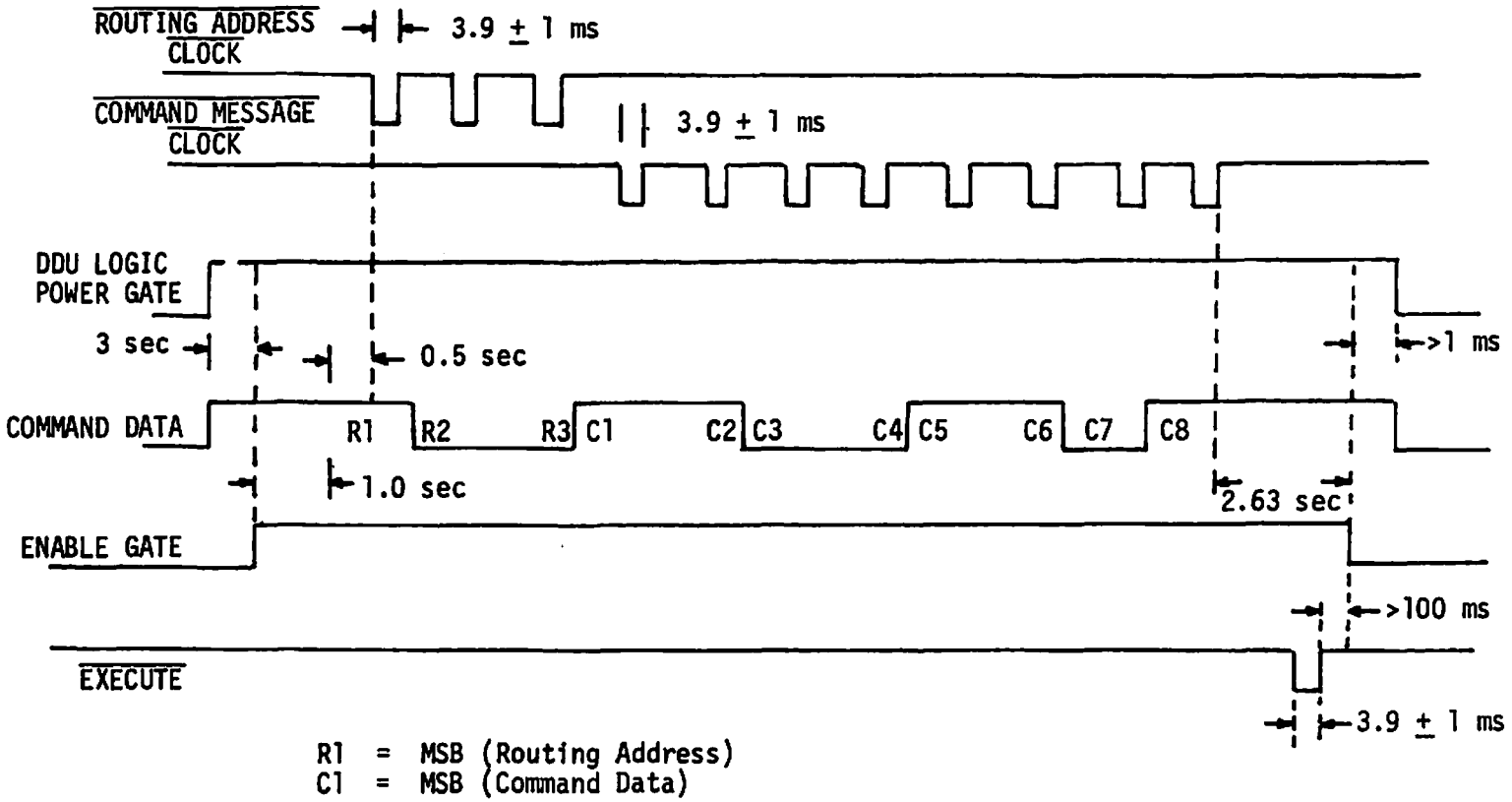
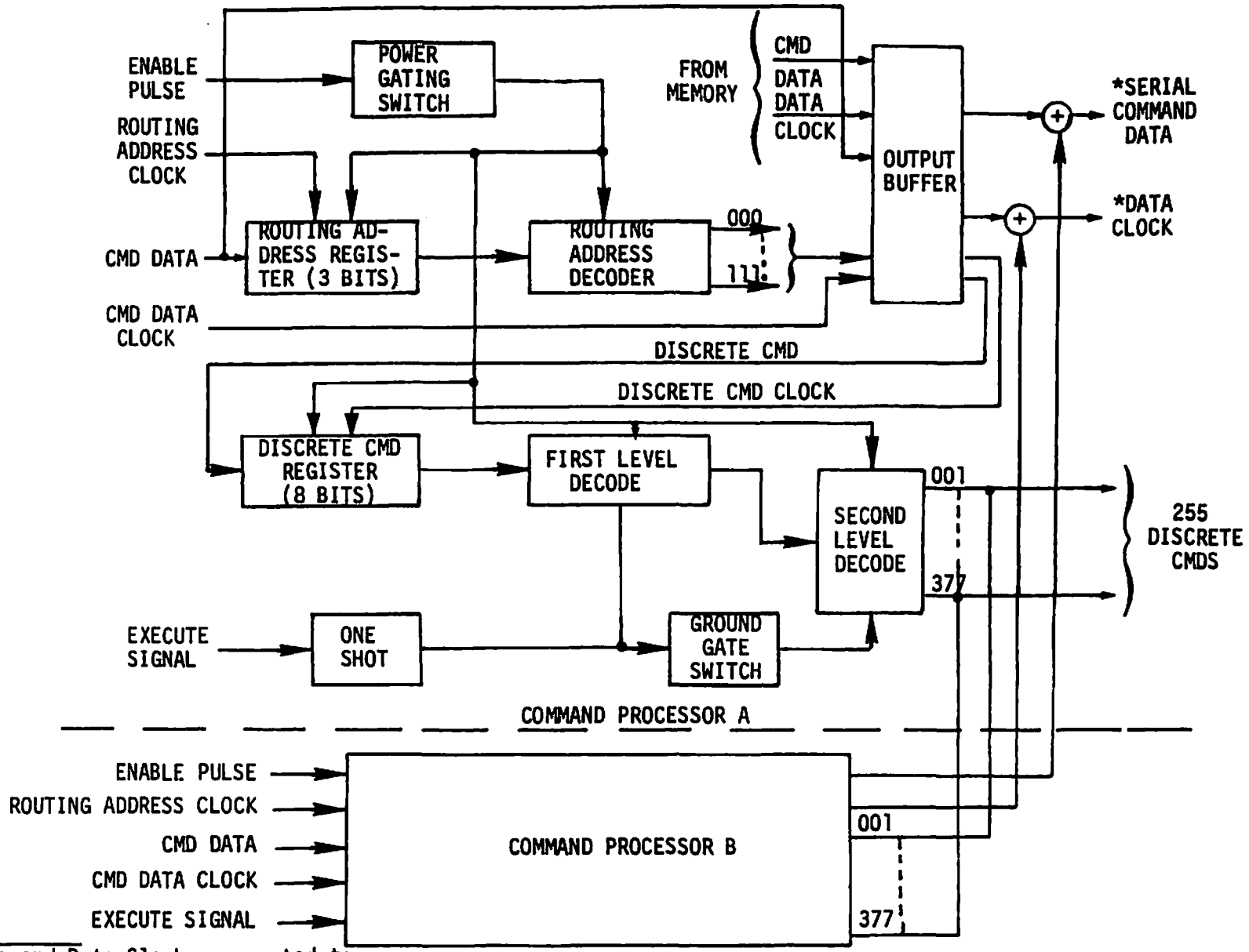


Figure 3.9-2. DDU Output Timing Diagram



* Cmd Data and Data Clock are routed to:

- o Cmd Storage DTU Serial (Memory--CDU)
- o Cmd Storage Discrete (Memory--CDU)
- o Time Storage (Memory--CDU)
- o ACS
- o DTU

Figure 3.9-3. Command Processing Block Diagram

TABLE I

MODES OF OPERATION

ITEM No.	FUNCTION	INITIATION	OPERATING TIME	OPERATING +5.3V POWER (mw)	REMARKS
1	Command Processing	Upon receipt of CMD signal from DDU via CMD link	15.2 seconds max.	650	Only one CMD processor operating at any time when ground commanding
2	Command Memory	Via Command Processor A or B (Gnd CMD "Execute Stored Sequence")	Dependent on total of 5 time delays stored in time registers	1660	CMD memory circuits are always "ON." Added power is due to CMD processor executing each CMD for 70 ms max and execution control function
3	Sequencer	Separation signal from 3rd stage/ spacecraft separation	2560 seconds max.	180	Sequencer automatically turns itself off
4	Overload Control	Signal from primary bus sensor in PCU	420 seconds	38.4	Operating time will be shorter if bus returnsto normalvolt.
5	Signal Present Detection	Absence of the two RCVR inputs indicating loss-of-lock	36.4 hours	150	Detection of ground transmission presence will terminate timing operation
6	Thruster Firing Counting	Thruster pressure switch closure	Continuous	340	Operating power is for total of 4 VCTs and 2 SCTs
7	Ordnance Firing	Via sequencer signals or ground commands	During armed period till receipt of safe CMD ordnance firing pulse ~ 5 ms capacitor delay	+5.3V=14.3 +28V=3000 30.5VAC=500	+5.3V power is continuous. +28V power during arm and ord firing only. 30.5VAC power during arm only.
8	Signal Conditioning	Continuous monitoring	Continuous	28	Circuits powered from +5.3V power input.
9	+5.3V OR switching and overvoltage sensing	+5.3V OR switching via gnd CMD. Overvoltage sensing is automatic	+5.3V OR switching 70 ms max. overvoltage sensing continuous	+5.3V=198 +28V=808	+5.3V OR switching done only upon a failure of power input being used

circuits. Following the leading edge of the Enable Pulse, the first three Command Data Units are clocked into the Routing Address Register by the Routing Address Clock. The information stored in the Routing Address Register is decoded by the Routing Address Decoder which selects one of eight routing destinations. They are:

	Octal Code	
1.	*000	Spare
2.	001	CDU Real Time Commands (Discrete Command)
3.	010	Command Memory Discrete Command Storage
4.	011	Command Memory Serial Command Storage (DTU only)
5.	100	DTU Serial Command Data (Prime and Redundant Paths)
6.	101	ACS Serial Command Data (Prime and Redundant Paths)
7.	110	Command Memory Time Message Storage
8.	111	Spare

*First data bit into the CDU

The following eight command data bits are routed through the Ouput Buffer to the particular destination selected by the Routing Address Decoder. The two Serial Command data destinations, octal code 100 and 101 (see above), are routed to users, DTU and ACS, respectively, outside the CDU. Three serial command data, octal code 010, 011 and 110, are routed to the Command Memory circuits within the CDU (see Paragraph 3.9.1.2.2). The remaining destination is the Discrete Command Processor octal code 001, within the CDU.

For discrete command decoding, the eight command data bits are routed and clocked into the Discrete Command Register consisting of 8 flip-flops. The outputs of the Discrete Command Register are decoded in two levels. The First Level Decode circuitry takes the 16 output lines from the register and forms 20 combinations, including the "and-ing" with the execution signal from the one shot. The 20 output lines are fed into the Second Level Decode circuit to complete the decoding (see Appendix 3.9-A). The outputs of the Second Level Decode produce a possible total of 256 commandable output states. The Second Level Decode circuitry is activated by means of the Power Gating Switch (Enable Pulse signal) and the Ground Gate Switch (Execute Signal) which applies the circuit ground. The Execute Signal is a pulse that is received after the last data clock pulse indicating the verification by the DDU of the spacecraft command message. The Ground Gate switch and Power Gating Switch provide the necessary isolation of a Command Processor, in the event of a failure, from interfering with the redundant command processor.

The decoded commands are transformed into octal notation such that octal commands 001 through 377 are available (command 000 is not implemented, so only 255 commands are actually available). The octal decoding and numbering method used is described in Appendix 3.9-A.

The octal commands are distributed in a four layer matrix (see Appendix A, "Pioneer Command Requirements"). Matrix layers 1, 2, 3 and 4 contain octal commands 000 through 077, 100 through 177, 200 through 277, and 300 through 377 respectively. Complementary commands have been assigned different vertical and horizontal elements of the matrix and/or different matrix layers. Assignment of commands in this manner eliminates single point failures from inhibiting proper commanding operation of a function due to multiple commands. For example, Transmitter No. 1 "ON" command (octal 223) is on layer 3 horizontal line 20 and vertical line 3; while Transmitter No. 1 "OFF" command (octal 232) is on layer 3, horizontal line 30 and vertical line 2. Another example is Ordnance Arm Command which (octal 243) is on layer 3, horizontal line 40 and vertical line 3; while Ordnance Safe Command (octal 365) is on layer 4, horizontal line 60 and vertical line 5. All critical commands have been assigned octal numbers equal to or greater than 200.

The design of the command processor is such that under real time commanding (exluding memory operating) only one (1) matrix layer is active (powered) at any one time, thereby consuming as little power as possible.

3.9.1.2.1.3 Operational Options. In the event of a failed command processor (A or B), the alternate command processor must be addressed (through the digital decoder unit) to obtain a serial and/or a discrete command out of the CDU.

During the execution of the memory stored sequence of commands, the CDU is still commandable via ground commands. It is important to note that ground commands during this mode must be sent through the command processor not being used by the memory circuits; otherwise, if during the sending of the ground command the processing becomes coincident with an execution of a command from the memory, an erroneous command may occur.

3.9.1.2.2 Command Memory

3.9.1.2.2.1 Command Memory Description. The function of the command memory is to provide the spacecraft with the capability to store commands for execution at a later time. The execution of the stored command is initiated by an "execute stored sequence" command sent from earth. A "stop stored sequence" command is provided to halt the command memory execution operation at any time in the sequence.

The command memory has the capability of storing up to five command messages and their associated time delay periods for later sequential execution. The command message shall be stored in five registers and the time delay period (time message) in another five registers. The registers utilize flip-flops as storage elements. Each command message register consists of nine flip-flops, eight for the command message and one to identify whether or not the command is a CDU discrete command or a DTU serial command data output. Each time message register consists of seven flip-flops.

Verification of the contents of the stored command messages and associated stored time delay periods is provided by means of a serial digital telemetry readout to the DTU. Continuous telemetry readout is provided even during the stored command execution period, and is interrupted only if coincident with an actual command message readout to the command processor or time message readout to the coincidence counter or if coincident with ground command loading of the command and/or time register.

3.9.1.2.2.2 Command Memory Operation (see Figure 3.9-4). The storing process for the command message and time delay period is accomplished by the command processors routing the command data information from the DDU to each memory register by means of the data clock (see Paragraph 3.9.1.2.1.2 for Routing Addresses). Before a stored data message is sent, a discrete ground command must be sent to the command memory to select the desired command message/time message register pair by means of the memory slot select circuitry. This process enables a gate to be opened for loading of the next two commands routed as a command message and/or time delay period. The slot select circuitry keeps the ground command addressed register pair "opened" until the end of the time message command or if a "Reset Slot Select Register" ground command (octal number 267) is received. At this time the slot select control circuitry returns to a non-active waiting state.

The normal sequence to load the memory storage registers is:

Steps	Process
1	Send "Select Register" command
2	Send command to be stored octal routing number 010 or 011 (see Paragraph 3.9.1.2.1.2)
3	Send Time Delay command to be stored octal routing number 110
4	Verify by telemetry the right messages were received and stored

If only the command is to be stored, then the following steps should be followed:

Steps	Process
1	Send "Select Register" command
2	Send command to be stored octal routing number 010 or 011 (see Paragraph 3.9.1.2.1.2)
3	Send "Reset Slot Select Reg" command octal number 267
4	Verify by telemetry the right message was received and stored

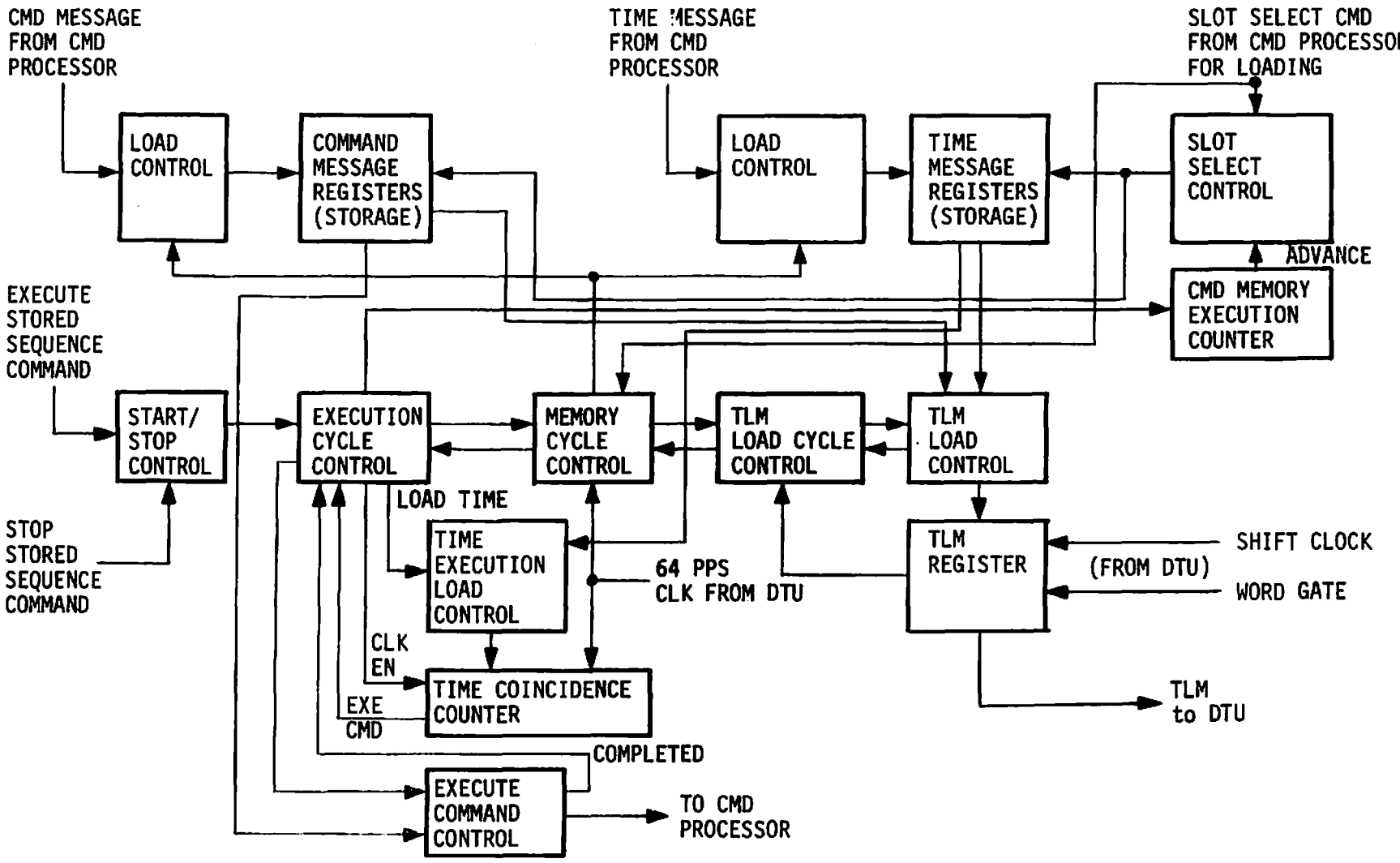


Figure 3.9-4

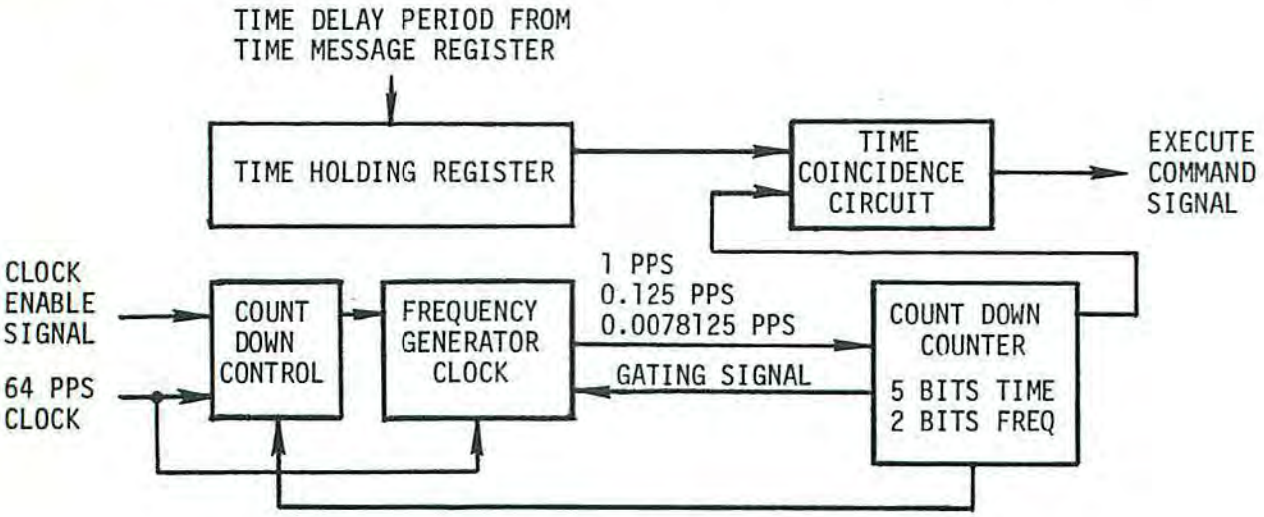
CDU COMMAND MEMORY BLOCK DIAGRAM

If only the time delay message is to be stored, then the following steps should be followed:

Steps	Process
1	Send "Select Register" command
2	Send Time Delay command to be stored octal routing number 110
3	Verify by telemetry the right message was received and stored

To initiate command memory operation, an "execute stored sequence" (start signal) command is sent from the command processor. This command energizes a flip-flop start/stop control which in turn initiates the execution cycle control circuitry to step through the execution cycle. Upon receipt of this command, the memory slot select is set to position "one" enabling the first register pair to be activated.

At this time, the time message data of the first register pair is clocked out to a time holding register. At the same time, a 1 pps controlling gate in the countdown control is enabled so that the clock can be fed into a seven bit countdown counter (see figure below). The output of the Frequency Generator clock are gated by the gating signal from the countdown counter such that three frequencies are generated: 1) 1 pps; 2) 0.125 pps; and 3) 0.0078125 pps.



TIME COINCIDENCE
COUNTER BLOCK DIAGRAM

Each of the above frequencies is selected in order, 1 pps being first, 0.125 pps next, and 0.0078125 pps, until the information in the five bit time counter and the two bit frequency select counter become coincident with the data in the time holding register as compared by the time coincidence circuit.

At the moment of coincidence, a signal is sent to the Execution Cycle Control to tell it to send the command message of the first register slot to a command processor previously selected by ground command. At the end of the stored command message to the command processor, the Execution Cycle Control generates an Execute Pulse and routes it to the command processor for execution of the stored command.

At the end of the execute pulse, a X5 counter (command memory execution counter) is stepped to the next count to allow the memory slot select the select the next register pair in order. The above process is repeated until all five commands are executed. At the end of the fifth command, a stop command is automatically generated to turn off the start/stop control.

The maximum time delay capable of being stored in any one slot is 8320 seconds. The resolution of each incremented time delay period is:

- 1 sec between 0 and 32 sec
- 8 sec between 32 and 288 sec
- 96 sec between 288 and 384 sec
- 128 sec between 384 and 8320 sec

Telemetry readout is provided in serial digital form to the DTU by the TLM load cycle control, TLM load control and TLM register. The contents of the time message registers, command message registers, and slot location is fed into the TLM register in that order by the TLM load control and TLM load cycle control. Upon receipt of the word gates from the DTU, the information held in the TLM register is readout in serial digital form via the shift clock from the DTU. At the end of the third word gate, a signal is sent back to the TLM load cycle control to start loading the TLM register with the next storage register contents.

A detailed description of the inter-relationship of each mode of operation (command and time message loading, command execution sequence, and telemetry readout) is given in Appendix 3.9-B.

3.9.1.2.2.3 Memory Operation Unique Characteristics. Due to the intermingling of various operating states and the basic circuit implementation, the memory network has a set of unique characteristics. They are:

a) The serial digital telemetry readout will be all "1's" when the Telemetry Register is in the process of being filled. The conditions under which this mode can occur are:

   - When the word gates and bit rate occur faster than the telemetry register filling rate (see item i).
   - During the loading of the memory time and command storage register via ground commands. The inhibiting of the loading of the telemetry register occurs

immediately after the receipt of a slot select register command. Telemetry loading is re-instated after the loading of the time storage register (see item b) or by a slot select reset command.

- When the first word gate from the DTU comes before the completion of the filling process for the telemetry register.

b) During command and/or time message loading of a register pair, the slot select flip-flop will only reset after the time message or if the "reset slot select register" command (octal number 267) is sent.

c) To re-cycle the command memory execution over and over without stopping (only by the "stop stored sequence" command, octal number 332) load the fifth slot with the "execute stored sequence" command (octal number 254). The loaded command will override the automatic "stop" command and allow the slot select to return to the first location and continue execution.

d) The command and time message registers (contents) will be reset to zero upon a momentary outage of the +5.3V power lines or initial application of power to the CDU.

e) During the stored command sequence execution, the command and time message storage registers are prohibited from being loaded via ground command. The slot select ground commands will get through as noted in item g, but the loading controls are inhibited by the Memory Cycle Control. Telemetry readout is not prohibited and will continue to function.

f) A remote possibility exists that if a Slot Select command is received selecting a particular register pair for data loading, and the Telemetry Control is in the process of reading out the stored data, the contents of the command register or time register would be garbled. This condition occurs because the clock to the select register pair is halted inhibiting the contents from recycling completely. Therefore, if a register pair is selected via ground command then the register pair should be loaded with the desired data. Of course, verification via telemetry (after the Slot Select circuit had been reset) to insure that the register pair was not changed could prevent having to load the previous selected register pair.

g) During execution of the stored sequence process, the register pair slot select flip-flops can be commanded via ground command to select another slot location. It this happens and a different slot location is selected other than the one being executed, then the ground command select slot

will be executed. By again ground commanding the slot select back to the slot being executed before the coincidence circuit has timed out, the correct command will be executed and the correct sequencing will be re-instated. If the slot select ground command happened to be coincident with the execution readout of a time or command memory storage register, the contents will be garbled. Therefore, the "Slot Select Register" command should not be sent during stored sequence execution unless absolutely necessary.

h) If the "Execute Stored Sequence" command is received while the telemetry control circuitry is reading out the first register pair or any register pair, the memory control circuitry will allow the telemetry readout completion before execution of the stored command sequence begins. The contents of the memory registers will not be affected.

i) During the accelerated Subcommutation Engineering format readout by the DTU, and at the fastest bit rate of 2048 BPS, the memory contents readout will be all "1's" for four cycles. The fifth time the DTU word gates readout the memory content the telemetry information will be correct. This condition is caused by the telemetry holding register in the CDU memory circuitry filling up at a 64 BPS rate while the word gate cycle time is 93.75 milliseconds. In other words, the 18 memory content bits takes (18 + 3)/64 = 328.125 milliseconds to fill the holding register (the 3 extra bit times are due to the delays in the telemetry control circuit). If the bit rate is set at 1024 BPS then two cycles of word gates are lost. At 512 BPS the memory content will be read at every word gate cycle time. No problem occurs in the normal non-accelerated Subcommutation Modes.

j) At all telemetry readout bit rates from the DTU, the telemetry holding register will not be fed new data until completion of the total holding register readout signified by the end of the third word gate.

k) During the memory stored sequence execution, if the telemetry control is reading out either the time storage or command storage registers and the execution cycle control wants the data from either register, it waits until the telemetry control completes its readout before allowing the execution cycle control to start its readout.

l) If the execution cycle control is in the process of reading out a particular storage register, the telemetry control cannot start its readout of that register until the execution cycle control completes its readout.

3.9.1.2.3 Sequencer

3.9.1.2.3.1 Sequencer Description. The sequencer provides an automatic time sequenced commanding function. The function is initiated after the separation of the spacecraft from the third stage. The sequencer consists of a binary countdown chain, decoding gates, and various controlling circuits. The timing sequence is driven from a 16 pps clock signal which is derived from the 64 pps clock in the command memory circuits.

3.9.1.2.3.2 Sequencer Operation (see Figure 3.9-5). Upon receipt of a separation signal, the CDU initiates the operation of the sequencer, through the start/reset buffer. The start/reset buffer sends a signal to the on/off control which enables a 16 pps clock signal to be fed into a binary countdown chain. The buffer also sends an enable signal to the enable/inhibit circuit to remove the inhibit clamp from the output decoding gates.

The outputs of the countdown chain are decoded by the decoding gates for the event times. The event time intervals are selectable by means of the time select patching matrix.

The preset events and event times after separation are as follows:

EVENT NUMBER	EVENT	EVENT TIME (SECONDS, + 0.02%)
1	Separation (seq. initiation)	0.0
2	Start Despin and Arm Ordnance Bus	~~64.0~~ 1024
2A	Despin Verification	~~112.0~~ 1072
3	Deploy RTG Booms	~~360.0~~ 1384
4	Deploy Mag. Boom and Actuate Thermal Feed Pin Pullers	~~1024.0~~ 2048
5	Initiate Reorientation	~~1408.0~~ 2432
6	Stop Sequencer	2560.0

To ensure that the RTG and magnetometer booms are not deployed at an improper spinning rate of the spacecraft during the despin period, a despin verification check is made at 112 seconds after separation. The check is made by sampling the despin thruster pressure switch signal at that time. If the indication is that the spacecraft is in a despinning mode, then the deployment sequence is enabled and sequencing continued. If the indication is that the spacecraft is not in the despinning mode, then the clock to run the countdown chain is halted.

In addition, Sequencer events 3, 4 and 5 are interlocked such that unless the signal for deployment of the RTG's occurs, the Magnetometer Boom deployment and Thermal Feed Pin Puller actuation signal is inhibited. Also, the Initiate Reorientation signal is inhibited until the Magnetometer Boom deployment and Thermal Feed Pin Puller actuation signal occurs.

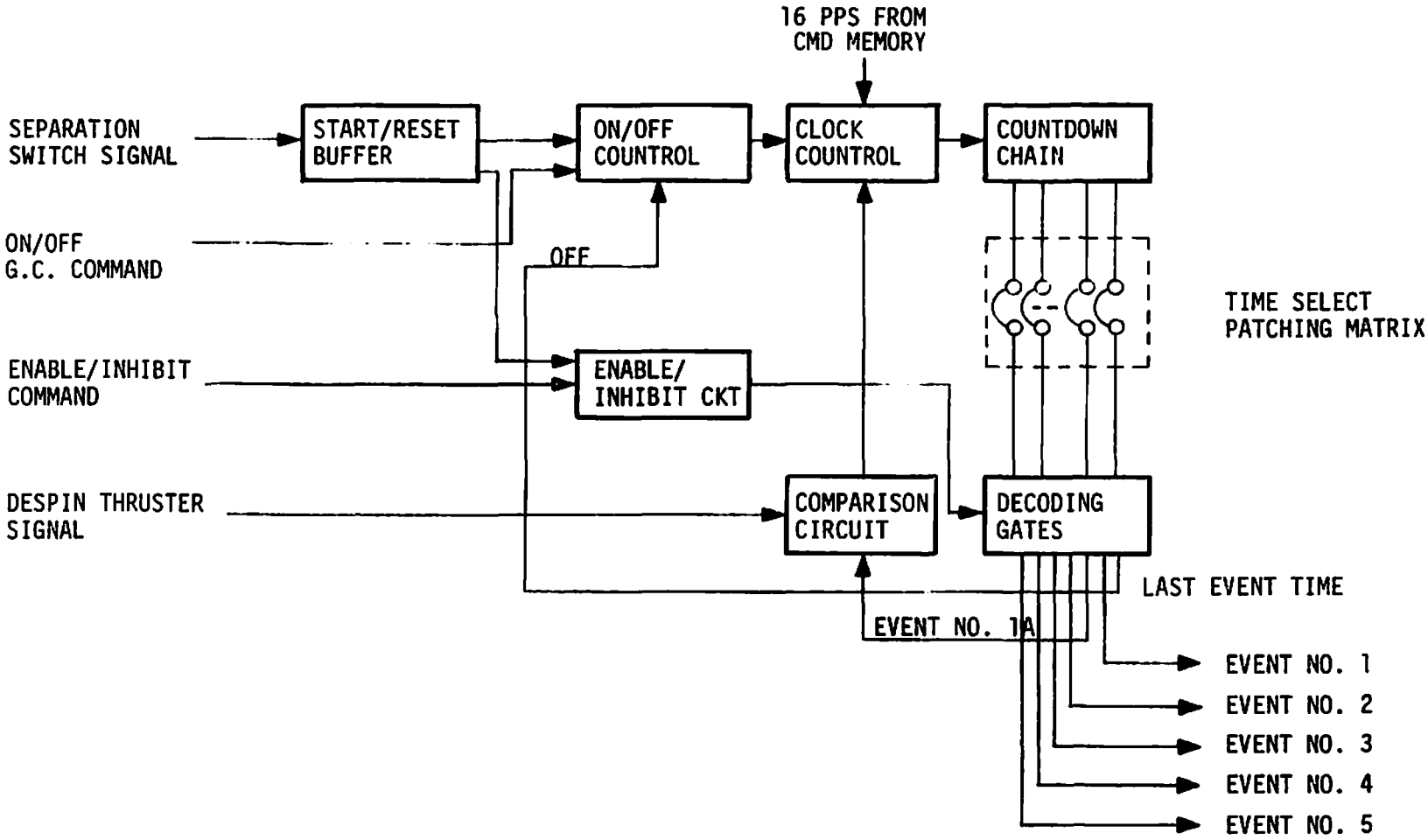


Figure 3.9-5

SEQUENCER BLOCK DIAGRAM

After the last event (event #5) a sequencer stop signal is generated and sent to the On/Off control circuit. The circuits at this time are powered down and remain in a low powered steady state mode.

3.9.1.2.3.3 Operational Options. In the event of a sequencer malfunction, the operation can be terminated by the "sequencer inhibit" ground command. The RTG and magnetometer boom deployment and initiate reorientation functions can be completed by ground commands.

If, for any reason, the sequencer is to be started or restarted; 1) before receipt of the separation signal or 2) after the receipt of the separation signal, the following procedure should be followed:

1) Before Separation Signal
   a. Send the "sequencer enable" command.
   b. Then, send the "start sequencer" command.

2) After Separation
   a. If the sequencer is allowed to continue beyond an operating time of 2560 seconds, send the "start sequencer" command.
   b. If the sequencer operating time is below 2560 seconds:
      1. Send the "CDU select 5VDC Bus A" or "CDU select 5VDC Bus B" command depending on the position of the +5VDC OR switching relay. The command to switch the relay to the opposite bus is sent which also resets the countdown chain to zero condition.
      2. Send the "sequencer enable" command.
      3. Then, send the "start sequencer" command. The sequencer event times will again start from the time of the receipt of the command.

NOTE: At sequencer operating time 112 seconds, if the despin thruster pressure switch is not activated, then the deployment and initiate reorientation commands will not be sent from the sequencer.

Upon recognition of the inhibiting of the Sequencer operation due to an improper spin rate (see Paragraph 3.9.1.2.3.2), the "Sequencer Inhibit" command should be sent. The sending of the command insures that an inadvertent Sequencer start-up does not issue the sequencer command signals. In addition, since the clock is halted and the automatic "Sequencer Off" signal will not be generated, the Power Reset signal can be simulated to power down the sequencer circuits (see Paragraph 3.9.1.2.10.3). If this is accomplished, it must be recognized that the CDU bi-stable functions will be put in their pre-determined state (see Table II of Paragraph 3.9.1.2.10.2).

The following telemetry sequencer status indications are brought out from the circuits:

1) Separation Switch Signal Presence

2) Start/Stop Status

3) Enable/Inhibit Status

4) Despin Thruster Pressure Switch Signal Indication

3.9.1.2.4 Overload Control

3.9.1.2.4.1 Overload Control Description. The overload control performs the function of turning off certain spacecraft loads in the event of an overload condition to the spacecraft main power bus. The loads are turned off in the following sequential order.

1) Scientific Instruments

2) ACS to Standby Power

3) Transmitters 1 and 2

The sensing of an overload condition is performed within the Power Control Unit (PCU). An overload condition is established when the D.C. bus voltage falls to 26.5 ± 0.5 VDC for a length of time of at least 200 ± 100 milliseconds. Upon detection of an overload condition, the PCU sends a signal to the CDU to start the sequential turnoff process. If, during the turning off of any particular load, the D.C. bus voltage returns to its normal voltage condition and the PCU overload signal returns to its normal state, the control circuit will not continue turning off the remaining loads. The loads already turned off will continue to be off until commanded back on via ground command.

3.9.1.2.4.2 Overload Control Operation. Upon receipt of an overload signal from the PCU, the overload control circuitry will be initiated via the overload signal detector, power gate, and reset circuit (see Figure 3.9-6). The power gate and reset circuit applies power to the OSC, counter, and decode select; as well as starting the oscillator and reseting the counter to zero. The OSC output signal is counted down by the counter which has its outputs decoded by the decode select establishing the sequence of the turn off commands. The "off" commands and the delay times for activating the commands are given below:

COMMAND	DELAY TIME
Scientific Instruments (experiments) Off	60 milliseconds
ACS to Standby	180 milliseconds
Transmitters 1 and 2 Off	300 milliseconds
Overload Control Operation Off	420 milliseconds

If at any time the PCU overload signal returns to its normal condition, the sequential turn off operation will terminate by restoring the power gate and reset circuit to its original state, shutting off power to the OSC, counter and decode select circuits.

As noted above, the overload control operation automatically turns itself off at time 420 millisecond after completing the entire load turn-off cycle. The automatic turn off signal is the same as an "O/L Inhibit" command.

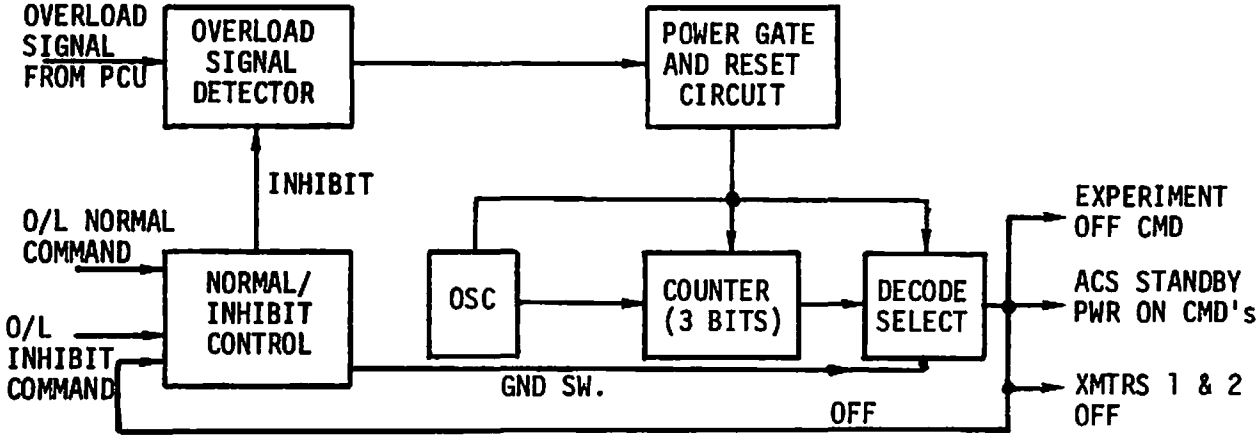


Figure 3.9-6. Overload Control

3.9.1.2.4.3 Operational Options. A means to inhibit the operation of the overload provision is incorporated. By sending the "O/L override" command, the overload signal from the PCU is inhibited from operating the power gate and reset circuit. To reinstate the operation the "O/L normal command" is sent.

3.9.1.2.5 Signal Present Detection

3.9.1.2.5.1 Signal Present Detection Description. The signal present detection provides for the automatic switching of the medium gain and high gain antenna inputs to the two spacecraft receivers. The automatic switching takes place if the receivers have lost lock with the ground station for a 36.4 hour time period. The operation starts in the signal present detection circuit when signal present signals from both receivers disappear. If either receiver is activated before the time delay period then the circuit will reset itself.

3.9.1.2.5.2 Signal Present Detection Operation (see Figure 3.9-7). Disappearance of both signal present signal (RCVR 1 and RCVR 2) inputs is detected by the Detector and Buffer, which in turn enables the Clock Control allowing the 16 pps clock to be fed into the Countdown Chain. Also at this time, the Reset and Power Gate circuit is activated to reset to zero and power up the Countdown Chain. If neither input returns to its normal state the Countdown Chain will continue operating until an elapsed time of 36.4 hours passes, at which time the Execute Control Generates an execute pulse

to the output command select. Prior to the countdown the output command select will have detected and held in memory which receiver was in the active state and will select the command to switch the high gain antenna to the opposite receiver.

If either signal present input returns to its normal state, then the 16 pps clock signal is prohibited from operating the Countdown Chain via the clock control. At that time power is removed from the Clock Control, Detector, and Buffer circuits and the Countdown Chain.

3.9.1.2.5.3 Operational Options. To inhibit the operation of the signal present detection at any time, an override provision is incorporated. By sending "RCVR S/P override" command, power is removed from the countdown chain. To reinstate the operation, an alternate method to reset the Signal Present operation after the "RCVR S/P override" command stops operation is by simulating the Power Reset signal (see Paragraph 3.9.1.2.10.3).

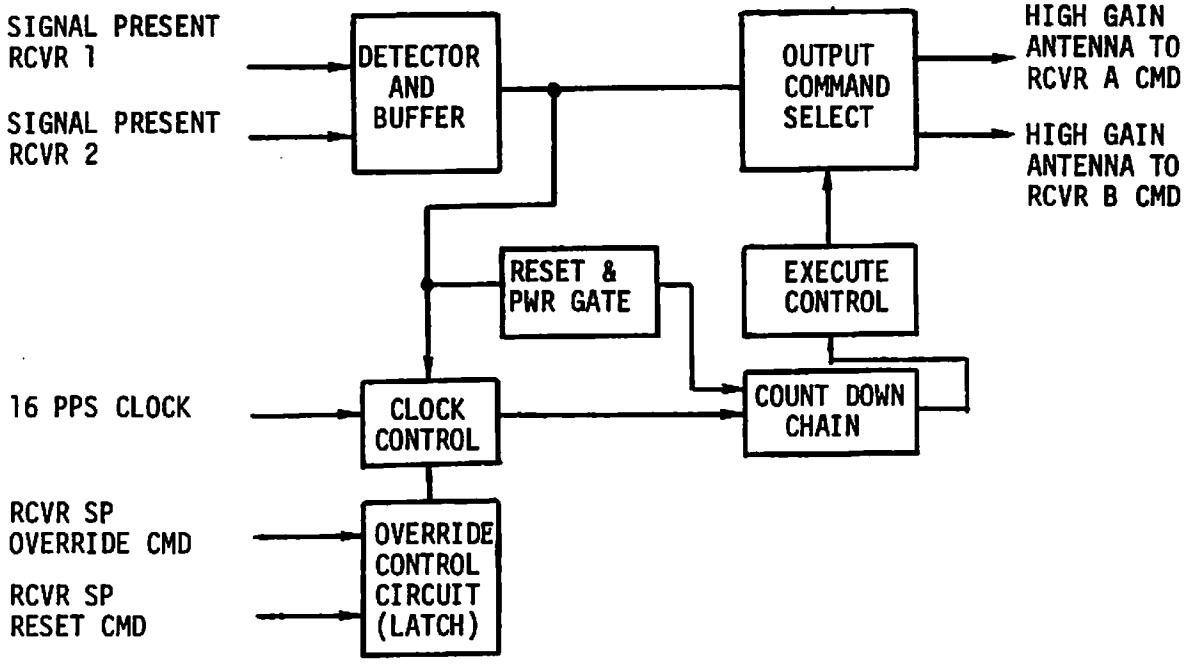


Figure 3.9-7. Signal Present Detection

3.9.1.2.6 Thruster Firing Counting

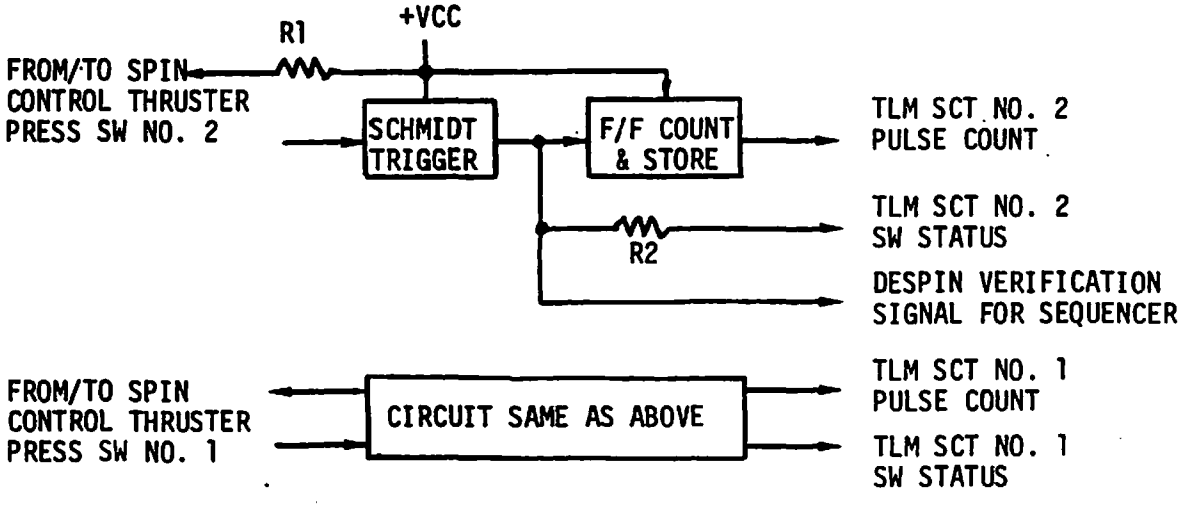
3.9.1.2.6.1 Thruster Counter Description. The function of the thruster counters is to provide indication of the number of firings performed by the four (4) spacecraft Velocity Precession Thrusters (VPT) and two (2) Spin Control Thrusters (SCT). The VPT counters have the capability

of counting up to 64 firings before recycling back to zero and starting over again. The SCT counters have the capability of counting up to two firings. In addition to the counting capability, the counters provide the real time telemetry indication of when the thrusters are firing.

The counting circuitry is activated when a pressure switch in the thruster propellant lines is closed, indicating a firing condition. When the pressure switch again opens, the thruster counter steps by one.

3.9.1.2.6.2 Spin Control Thruster Counter Operation. The spin control thruster counter circuit consists of a Schmidt trigger and a flip-flop count and store circuit (see figure below). When the spin control thruster pressure switch closes, the circuit is closed between Resistor R1 and the input to the Schmidt trigger. The Schmidt trigger is activated, sending a signal to the F/F control and store circuit. The F/F control and store circuit does not react to the signal but rather to the disappearance of the signal indicating the end of the thruster firing. At that time the F/F control and store circuit steps its count to one and holds for telemetry readout and until the next pressure switch closure and opening. The output of the F/F count and store circuit is a binary count of two (0, 1, 0, 1, ... etc.). A real time telemetry indication of the pressure switch status is provided to the DTU through Resistor R2.

The spin control thruster counter circuit for SCT No. 2 also provides a despin verification signal to the sequencer in the CDU (see Sequencer Operation, Paragraph 3.9.1.2.3).



Spin Control Thruster Counter

3.9.1.2.6.3 Velocity Precession Thruster Counter Operation. The Velocity Precession Thruster Counter (VPT) consists of a Schmidt trigger, counter and storage, and TLM holding and serial shift register (see Figure 3.9-8). As in the spin control thruster counter operation, when the velocity precession thruster pressure switch closes, the circuit is closed between Resistor R1 and input to the Schmidt trigger. The Schmidt trigger is activated sending a signal to the counter and storage circuit containing six flip-flops. The counter and storage circuit does not react to the

signal but rather to the disappearance of the signal indicating the end of the thruster firing. At that time, the counter and storage circuit steps up the count by one and holds until the next pressure switch closure and opening. The output of the counter and storage circuit is a binary count from 1 to 64 steps before recycling back to step 1.

The information content in the counter and store circuit is constantly fed into the TLM holding and serial shift register and only is interrupted during a word gate readout to the DTU. During the receipt of a word gate the contents of the TLM holding and serial shift register is serial shifted out via the DTU shift clock.

3.9.1.2.7 Ordnance Firing

3.9.1.2.7.1 Ordnance Firing System Description. The ordnance firing system consists of a capacitor discharge-SCR trigger firing scheme (see Figure 3.9-8). The ordnance devices ignited by the CDU ordnance firing system are:

1) RTG Ordnance

2) Magnetometer Boom Ordnances

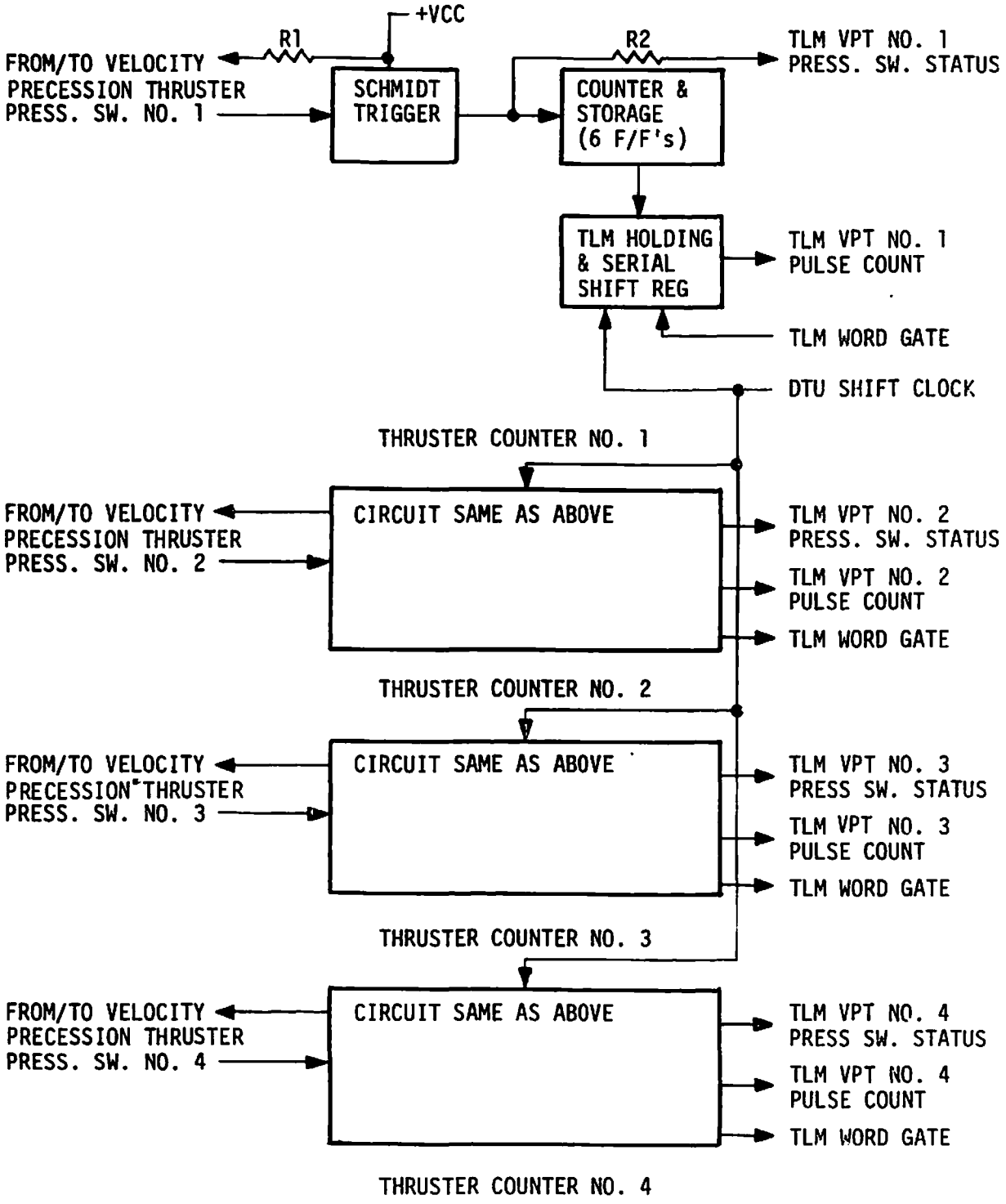
3) Feed Movement Mechanism Pin-puller Ordnances

Redundant capacitor chargings and capacitor storage bank circuits are provided such that the prime ordnance devices are ignited from one circuit and the redundant ordnance devices from the other. The redundant firing circuit is designed to trigger approximately 50 milliseconds after the prime side to ensure the bolts holding the RTGs have ample time to pass the redundant bolt cutter after being cut by the first cutter. Possibilities exist that if the redundant bolt cutter starts cutting the same time as the first, the bolt will hang up.

3.9.1.2.7.2 Ordnance Firing Operation (see Figure 3.9-9). Upon separation of the spacecraft from the spin-up section of the launch vehicle an ARM signal will be received from the sequencer which will arm the ordnance circuit by means of the SAFE/ARM circuit (see Sequencer Operation, Paragraph 3.9.1.3.3). When the SAFE/ARM circuit is in the armed state an AC input signal is allowed to pass into the transformer/rectifier circuit. The full wave rectified output signal is fed into capacitor banks to charge a set of capacitors in each bank. This signal is stored as a "charge" on the capacitors until is is used to ignite the ordnance device squibs when the SCR switches are triggered.

The design of the ordnance circuit has the capability of firing up to six ordnance squibs simultaneously with the energy supplied from one capacitor bank. This is accomplished by carefully selecting the values of capacitance for each capacitor bank, sizing the SCR switch to handle the required load, and providing current equalizing resistors in the output lines (R1 and R2).

The SCR switches are turned on by triggering circuits which have inputs from the CDU sequencer and CDU command processors. The sequencer



VELOCITY PRECESSION THRUSTER COUNTERS

Figure 3.9-8

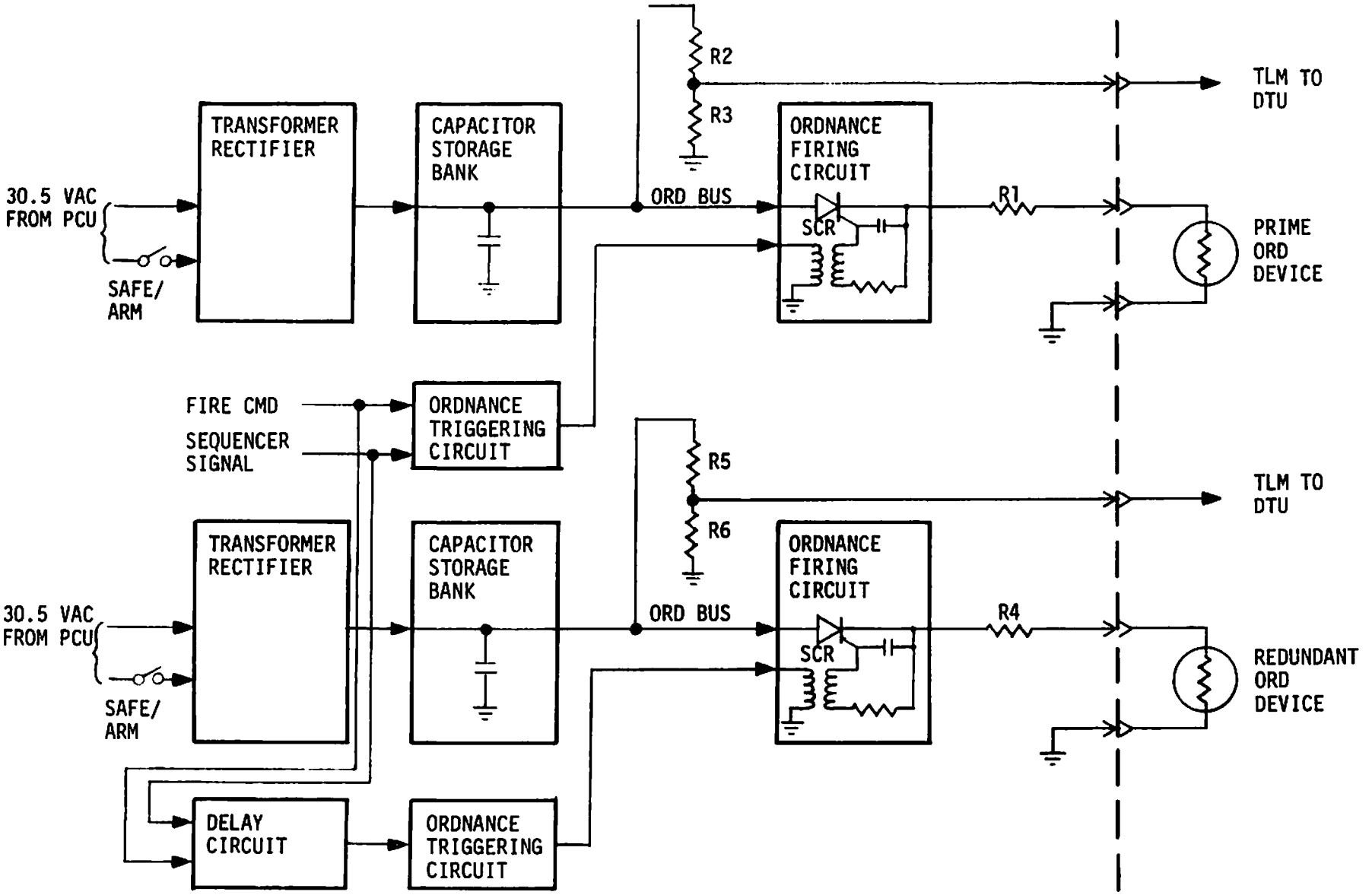


Figure 3.9-9

ORDNANCE FIRING SYSTEM

inputs trigger the SCR switches automatically when each ordnance fire timed event occurs in a ripple fire mode. The command processor inputs provide the ground command backup capability to fire ordnance also in a ripple fire mode. In the ripple fire mode a discrete ground command or sequencer signal is received and the primary set of ordnance is fired first. After a set time delay a secondary set of ordnance is fired automatically.

Resistors R2, R3, R5, and R6 provide a voltage divider circuit so that the voltages on the capacitor storage banks are monitored for telemetry readout to the DTU. The telemetry readout indicates a positive SAFE/ARM status. In addition, the SAFE/ARM switch status is monitored for telemetry.

3.9.1.2.7.3 Operational Options. Redundant ordnance fire commands for deployment of the RTGs and magnetometer boom are provided in the event the sequencer event signal or prime ordnance fire commands fail.

Redundant ordnance fire commands for deployment of the RTGs and Magnetometer boom are provided in the event the sequencer event signal or prime ordnance fire commands fail.

Upon completion of the ordnance fire functions, (RTGs and Magnetometer boom deployed, Feed Movement Mechanism pin pulled), the Ordnance Safe command should be sent. De-activating the Ordnance circuits reduces the power dissipation on the A.C. bus.

3.9.1.2.7.4 Unique Characteristics. Under a degraded condition of the SCR holding current (after the firing of the RTG ordnance devices), if it is noted via telemetry that the capacitor storage banks do not charge back up for firing of the magnetometer boom ordnance devices, the following procedure should be followed:

1) Send the ordnance safe command

2) Wait approximately a minute (60 seconds)

3) Send the ordnance arm command

4) Wait approximately 5 to 10 minutes

5) Send the ordnance fire command

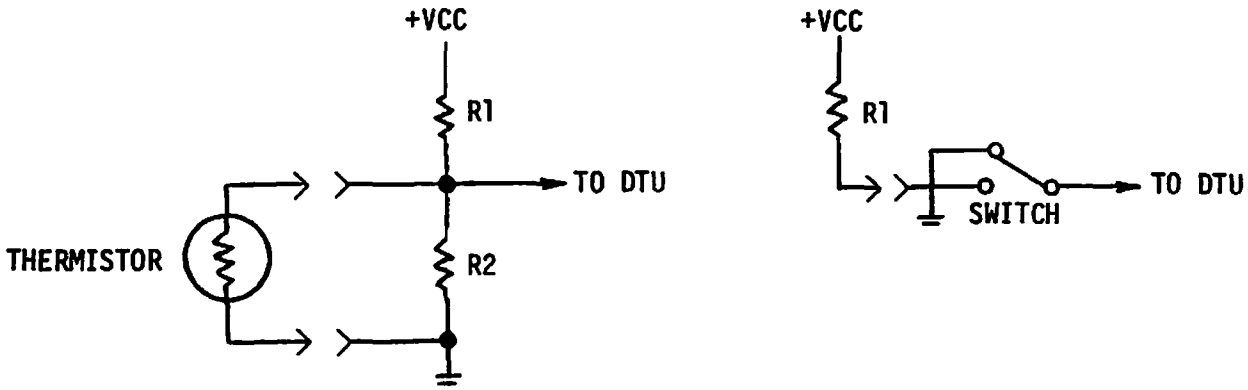
The above procedure allows the remaining charge on the capacitors to discharge through the SCR switches to a point where the degraded SCR(s) can turn off, thus allowing the capacitor to charge up after the ordnance arm command is sent.

3.9.1.2.8 Telemetry Signal Conditioning

3.9.1.2.8.1 Signal Conditioning Description The purpose of the signal conditioning circuits in the CDU is to provide signal compatibility with the DTU input requirements. Each signal conditioning circuit consists of only passive resistor components. The following signals are conditioned:

1) Propellant Tank Temperature
2) Transmitter Driver No. 1 Osc. Temperature
3) Transmitter Driver No. 2 Osc. Temperature
4) Velocity Thruster No. 1 Temperature
5) Velocity Thruster No. 2 Temperature
6) Spin Thruster Temperature
7) Structures Temperature No. 1
8) Structures Temperature No. 2
9) Structures Temperature No. 3
10) Structures Temperature No. 4
11) Structures Temperature No. 5
12) Structures Temperature No. 6
13) Plasma Analyzer Exper. Temperature
14) Charged Particle Exper. Temperature
15) Geiger Tube Telescope Exper. Temperature
16) Cosmic Ray Telescope Exper. Temperature
17) Ultraviolet Photometer Exper. Temperature
18) Thruster Bed Temperature A
19) Thruster Bed Temperature B
20) Thruster Bed Temperature C
21) Thruster Bed Temperature D
22) RTG 1-2 latch Switch Status
23) RTG 3-4 latch Switch Status
24) Magnetometer Boom latch Switch Status
25) Receiver coax Switch Status
26) Transmitter coax Switch Status

3.9.1.2.8.2 Signal Conditioning Operation. Two types of passive signal conditioning are provided by the CDU (see the figure below). The conditioning of thermistors is performed by a three-legged birdge circuit with the thermistor (registor change vs. temperature) in the third leg. The two resistors in the CDU provide linearization of the non-linear resistor change and the voltage for the necessary signal level to meet the DTU input requirements. For event signals, i.e., RTG 1-2 deployment latch switch, a voltage is supplied through a current limiting resistor to the switch which directly interfaces with the DTU.



THERMISTOR SIGNAL CONDITIONING

EVENT SIGNAL CONDITIONING

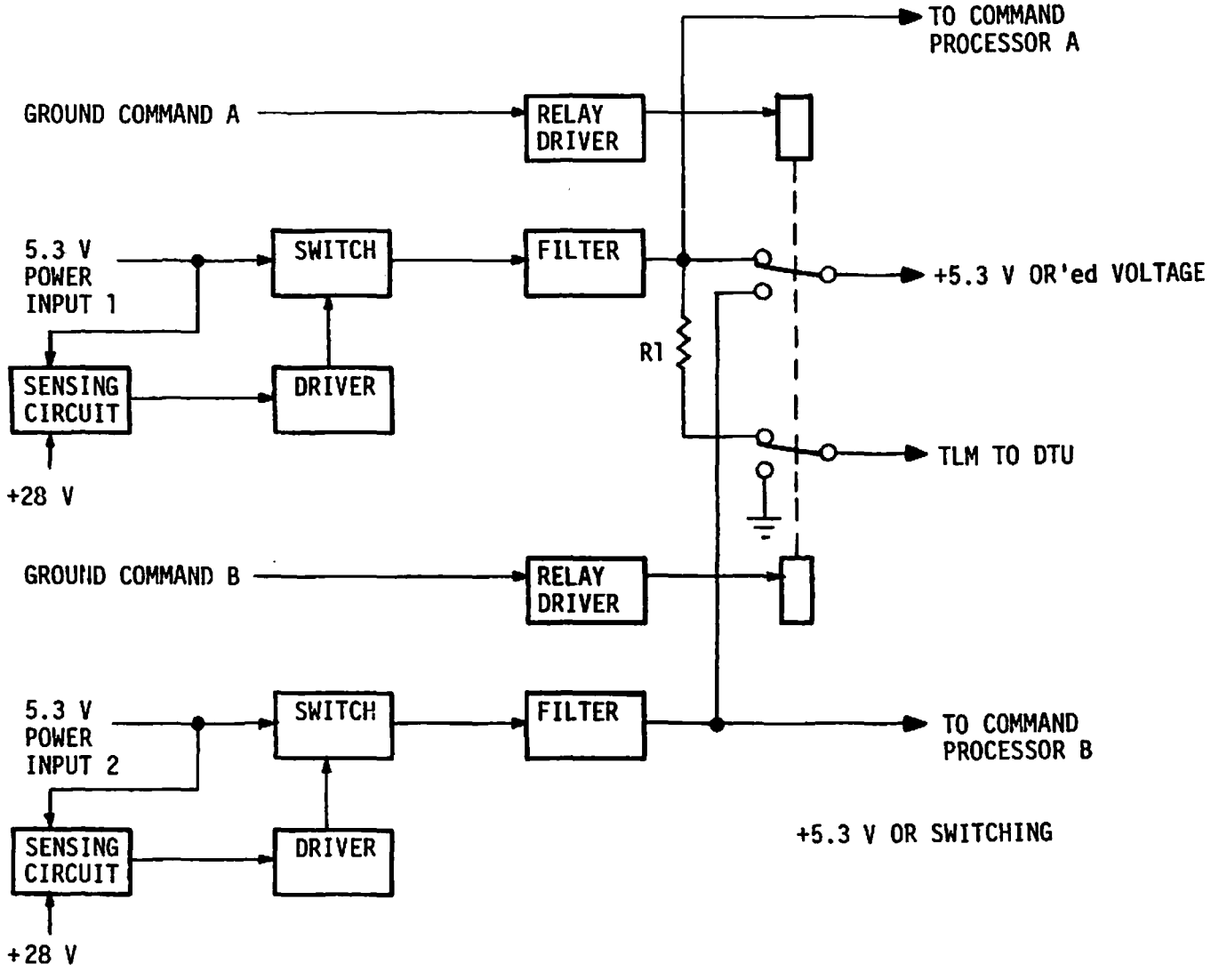
3.9.1.2.9 +5.3V OR Switching and Overvoltage Sensing

3.9.1.2.9.1 +5.3V OR Switching and Overvoltage Sensing Description. The function of the +5.3V OR switching is to provide the CDU with the capability of switching to the redundant +5.3V power supply input in the event of power supply malfunction or CDU circuit shorting of the power line being used. The function of the overvoltage sensing circuit is to detect an overvoltage condition on the +5.3V inputs No. 1 or No. 2 and remove that input from the unit.

The switchable +5.3V OR switching of the power supply inputs is an "OR" scheme to overcome any excessive voltage drop in the power input line.

The reason for the overvoltages sensing circuit is to protect the CDU circuits from excessive voltage on the +5.3V power lines in the event of a power supply malfunction.

3.9.1.2.9.2 +5.3V or Switching and Overvoltage Sensing Operation (see Figure 3.9-10). The +5.3V OR switching circuit consists of a latching relay and relay drivers. Upon determining a failure of the power supply line being used via telemetry, a ground command (A or B) is sent to switch over the relay contacts.



OVERVOLTAGE SENSING

Figure 3.9-10

Command Processors A and B are powered separately from +5.3V power inputs 1 and 2 respectively, so that the command link will not be disabled. Non-redundant functions are powered from the +5.3V OR'ed power supply, i.e., command memory.

The overvoltage sensing circuit detects the +5.3V power input voltage via the sensing circuit. The +28V power input to the circuit provides the bias power to keep the circuit operating during an overvoltage condition. The sensing circuit, upon detecting the overvoltage condition, sends a signal to the driver, turning the switch "OFF." The trip point for the overvoltage detection is set at +6.2 volts.

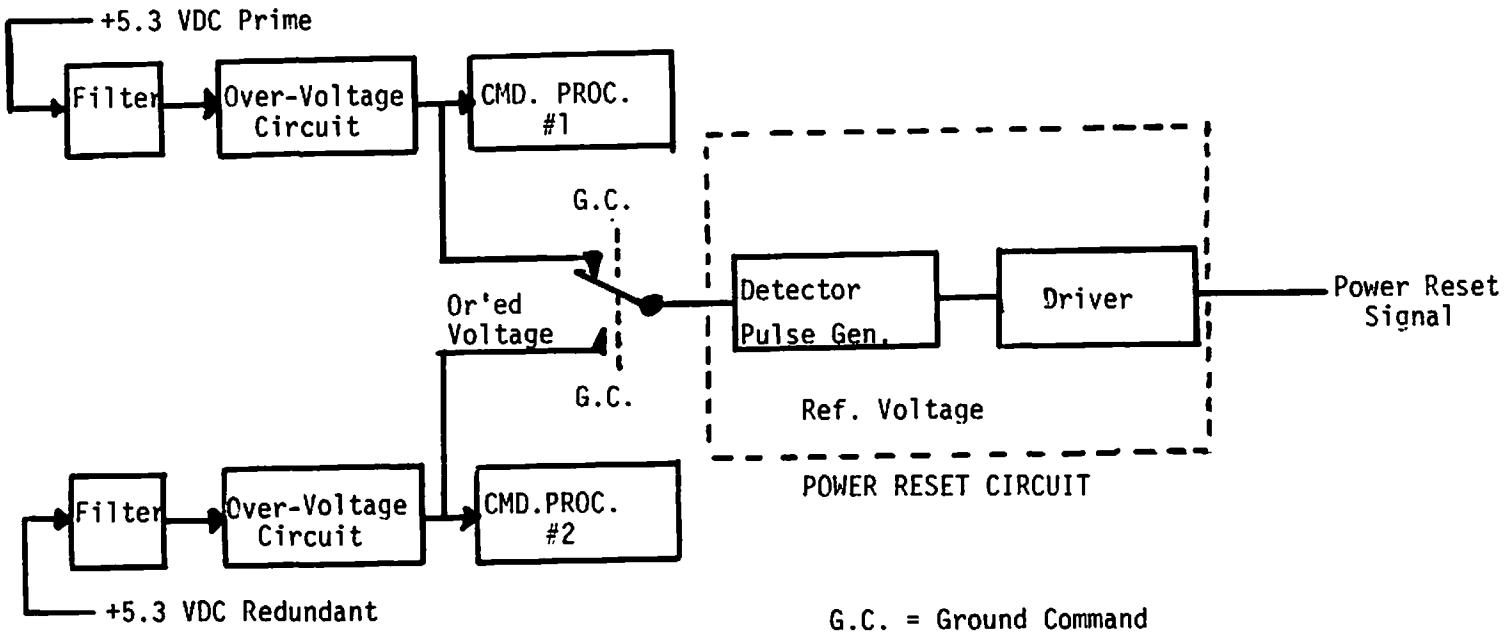
3.9.1.2.10 Power Reset

3.9.1.2.10.1 Power Reset Description. The function of the Power Reset circuit is to set or reset the various bi-stable functions in the CDU to a predetermined condition. This function occurs during the initial application of power (+5.3 VDC power only) to the unit or during a momentary outage of the power input lines.

3.9.1.2.10.2 Power Reset Operation (see Figure 3.9-11). The OR'ed +5.3VDC power input is sensed by the Detector/Pulse Generator. When the +5.3VDC voltages are first applied to the unit, the Detector circuit senses for approximately +4.6 $\pm$ 0.14 VDC. When this voltage point is reached, the Detector circuit triggers the Pulse Generator circuit. The Pulse Generator then sends a power reset signal (pulse) through a Driver circuit to the various bi-stable functions. In the event of a momentary +5.3 VDC power outage, and if the voltage falls below the trigger level of the Detector circuit, the Power Reset circuit sends a Power Reset signal upon return of the +5.3 VDC voltage. Due to the response time of the Detector circuit (via filter capacitors) the momentary +5.3 VDC power outage must exceed a time duration and voltage level drop combination as described in Figure 3.9-12.

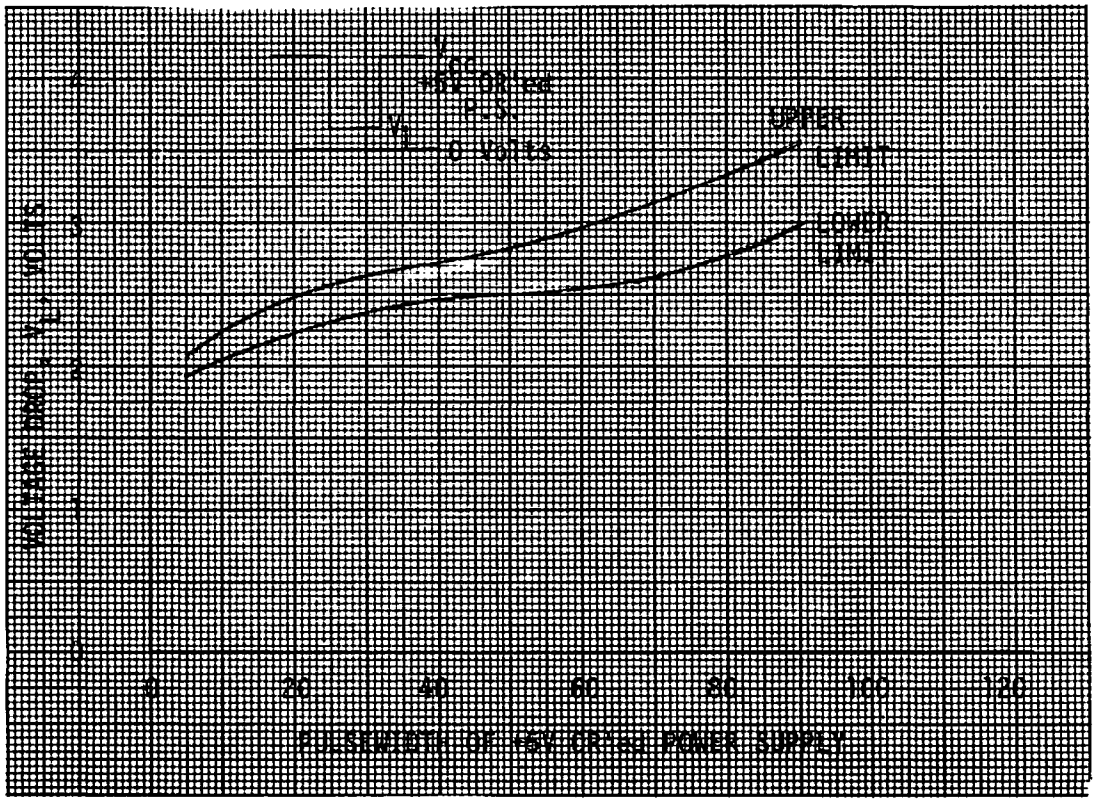
The bi-stable functions set to a pre-determined state are listed in Table II.

3.9.1.2.10.3 Operational Option. The Power Reset signal can be simulated by switching of the OR'ed voltage relay (see Figure #.9-11) via ground command. The two ground commands to switch the relay are: 1) CDU Select 5 VDC Bus A; and 2) CDU Select 5 VDC Bus B. The "CDU Select 5 VDC Bus A" selects the +5.3 VDC prime voltage input, and the "CDU Select 5 VDC Bus B" selects the +5.3 VDC redundant voltage input. The Power Reset signal is simulated when the relay is switched to the opposite +5.3 VDC voltage input from the input being used.



POWER RESET FUNCTION

Figure 3.9-10



+5VDC MOMENTARY OUTAGE LIMITS

Figure 3.9-11

Table II

Function	Pre-determined State
Ordnance Safe/Arm	Safe
Experiments ON/OFF	Off
Command Memory Registers (Command and Time)	Reset to all zero content
Command Memory Register Slot Select	Reset to zero state
Command Memory Telemetry Slot Select Readout	Reset to zero state
Command Memory Execute/Stop Stored Sequence	Stop
Sequencer Start/Stop	Stop
Sequencer Enable/Inhibit	Inhibit
~~Undervoltage Inhibit/Normal~~ Overvoltage Protection	~~Normal~~ ON
Signal Present Override/Reset	Reset
Main Feed Movement Mechanism Normal/Offset (Prime and Redundant)	Normal
Transmitter No. 1 ON/OFF	Off
Transmitter No. 2 ON/OFF	Off
Receiver No. 1 Coho Mode ~~Inhibit/Non-Inhibit~~	Coherent Non-Inhibit
Receiver No. 2 Coho Mode ~~Inhibit/Non-Inhibit~~	Coherent ~~Non-Inhibit~~
Spin Control Thruster Firing Counter No. 1	Reset to zero state
Spin Control Thruster Firing Counter No. 2	Reset to zero state
Velocity-Precess Thruster Firing Counter No. 1	Reset to zero state
Velocity-Precess Thruster Firing Counter No. 2	Reset to zero state
Velocity-Precess Thruster Firing Counter No. 3	Reset to zero state
Velocity-Precess Thruster Firing Counter No. 4	Reset to zero state

3.9.2 Digital Decoder Unit (DDU)

3.9.2.1 DDU Description. The DDU consists of two completely redundant decoders each of which has the capability of demodulating the FSK modulated subcarrier signal from the receiver and verifying the validity of the command. Each decoder is capable of receiving FSK signals from either receiver. The units are connected by a resistance cross strap network to protect against a single failure in either decoder or receiver from totally disabling the processing of commands. See Figure 3.9.2-1.

3.9.2.2 Functional Characteristics Summary. The DDU functional characteristic summary is given in Table 1.

3.9.2.3 Command Format. The bit structure of the command format is shown in Figure 3.9.2-2. The first four bits are the preamble and consist of all zero's. These four bits are required to deactivate the squelch in each decoder and to establish a condition that will recognize the one bit which follows. This one bit is used to establish data bit synchronization.

The decoder address bits follow the sync bit and are used to select one of the two decoders. Two bits are used for the decoder address. The address for decoder A is 01 and the address for decoder B is 10. Only the addressed decoder executes the command.

The routing address and the command message comprise the next two groups of bits in the command format. These bits are the useable data in the format. The three bit routing address is utilized by the CDU to determine the type of command received and to which user the command message is to be routed. The eight bit command message contains the actual information required by the user.

The remaining four bits of the command format are parity check bits. These parity bits are used to check the integrity of the routing address and the command message bits. The code is capable of detecting all possible one and two bit error patterns. Each parity check bit is equal to the modulo-2 summation of selected data bits as shown by the equations in Figure 3.9.2-2. Even parity is used in each case.

3.9.2.4 Demodulation. Upon receipt of an FSK signal modulation by tone frequencies of 128 Hz and 204.8 Hz at a one bit per second rate, each decoder will demodulate the data into a digital form. The 128 Hz tone is assigned to represent a logical zero and the 204.8 Hz tone represents the logical one. Refer to Figure 3.9.2-3.

These tones are first separated into their respective data channels by bandpass filters. The filters convert the wideband input signal into two signals. These signals are fed into and compared in a comparator circuit which decides whether the current bit is a one or a zero based on which output is largest.

Bit synchronization is achieved when the output of the one filter exceeds the output of the zero filter for the first time in the command.

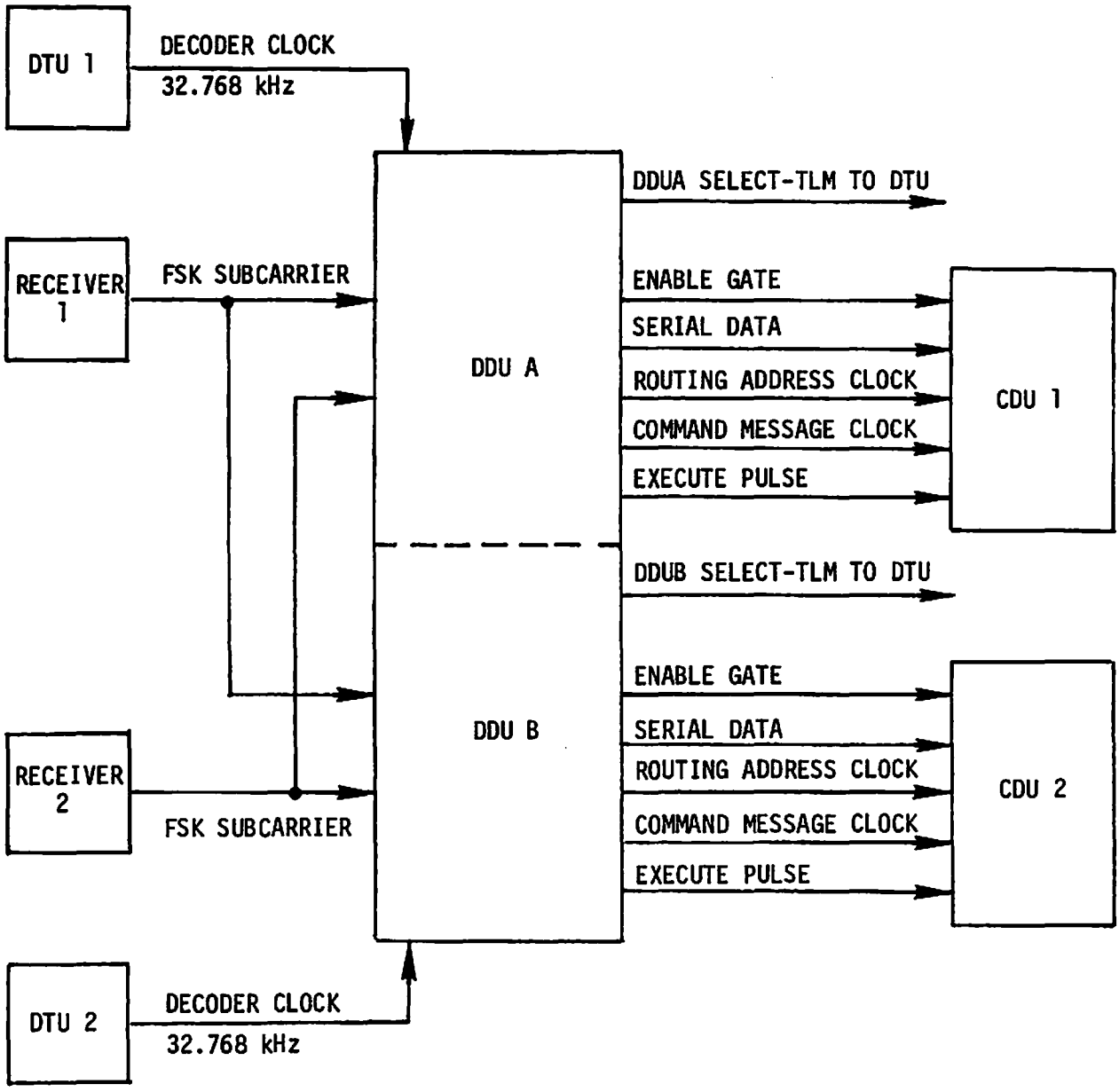


Figure 3.9.2-1. DDU Interface Diagram

Table I. DDU Characteristics

1. Modulation--FSK

2. Type--Active Twin "T" Filter-Demodulator

3. Bit Rate--1 Bit Per Sec

4. Command Structure

   - 4 Bits Preamble (zeros)
   - 1 Bit Sync (one)
   - 2 Bits Decoder Address
   - 3 Bits Routing Address
   - 8 Bits Command Message
   - 4 Bits Parity

5. Output Interface--(Serial)

   - 3 Bits Routing Address
   - 8 Bits Command Message
   - Command Message Clock
   - Routing Address Clock
   - Command Execute

6. Command Authentication--4 Bit Parity
   Block Hamming Code

7. Tone Frequency

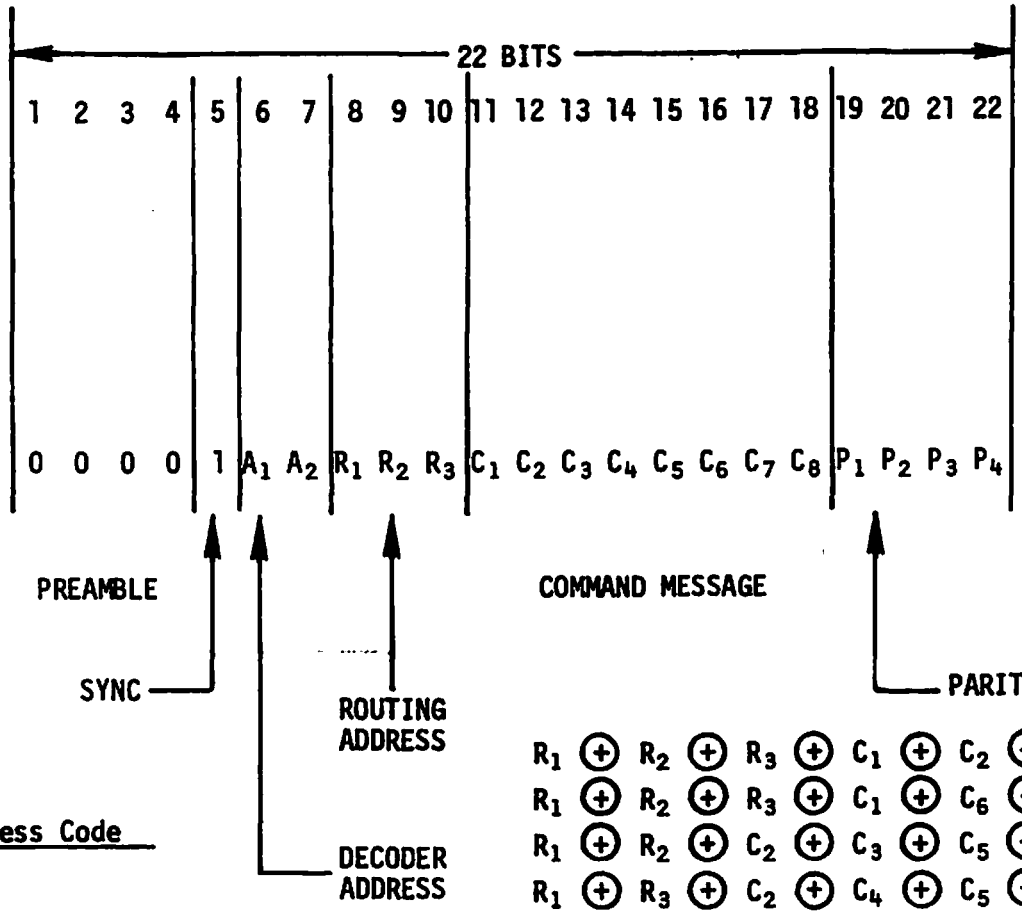
   - 1 = 204.8 Hz
   - 0 = 128.0 Hz

8. Performance

   - Probability of Executing False Command is 1.1×10^{-9} with Input S/N Ratio of 17.3 db (performance requirement).

9. Input Signal

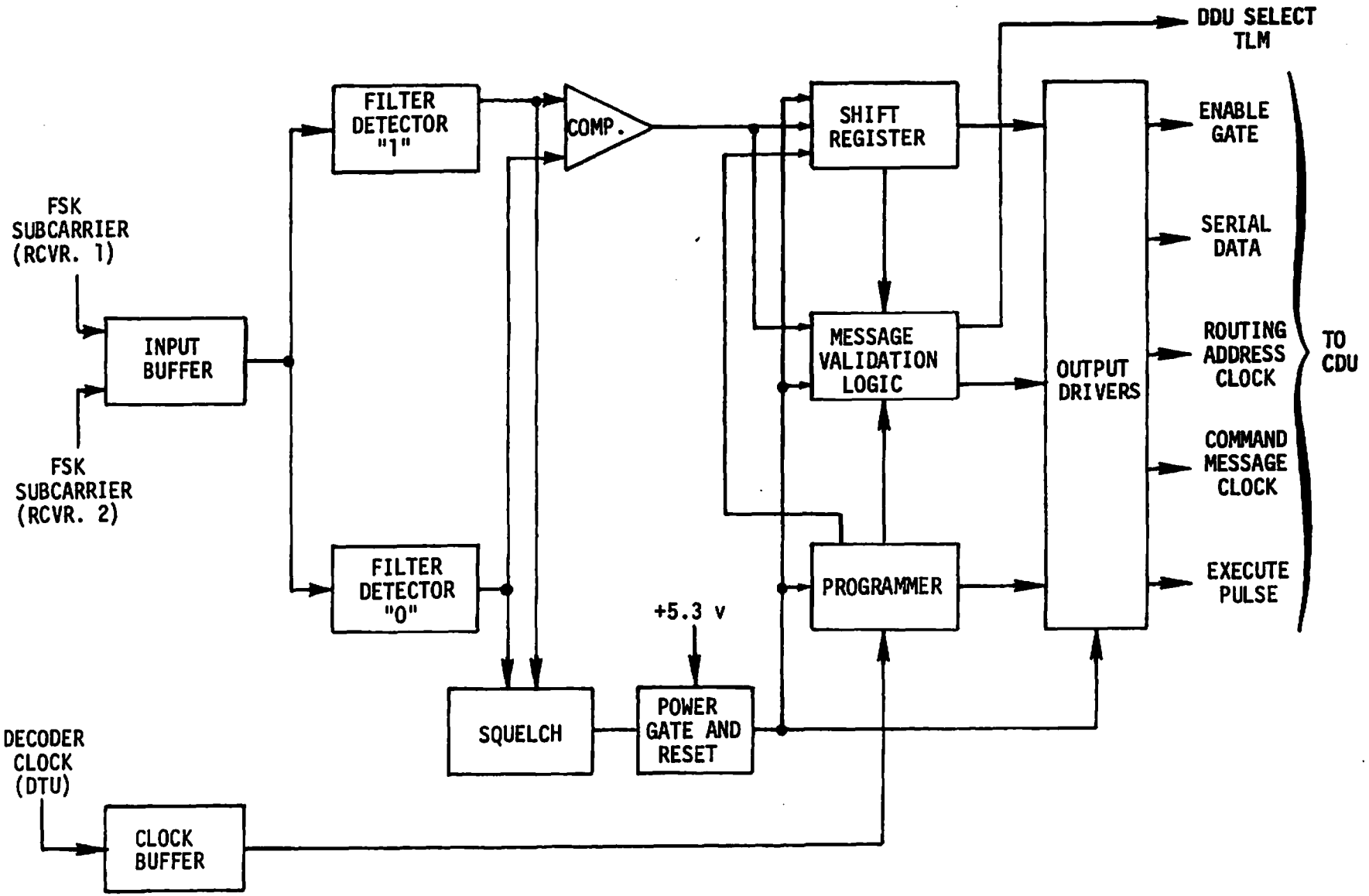
   - Strong S/N 9.4 $\pm$ 1.5 volts P-P (signal only)
   - Threshold S/N 1.90 $\pm$ 0.3 volts P-P (signal only)
     10.6 $\pm$ 1.7 volts P-P (signal + noise)



Routing Address Code

R_1 R_2 R_3	
000	Not Used
001	CDU Command Processor (Discrete Command)
010	CDU Command Memory, Command
011	CDU Command Meory, DTU Serial Command
100	DTU Serial Command Data (8 Bits)
101	ACS Serial Command
110	CDU Command Memory, Time
111	Not Used

Figure 3.9.2-2. Command Format


FSK SUBCARRIER (RCVR. 1)
FSK SUBCARRIER (RCVR. 2)
INPUT BUFFER
FILTER DETECTOR "1"
FILTER DETECTOR "0"
COMP.
SQUELCH
+5.3 v
POWER GATE AND RESET
DECODER CLOCK (DTU)
CLOCK BUFFER
SHIFT REGISTER
MESSAGE VALIDATION LOGIC
PROGRAMMER
OUTPUT DRIVERS
DDU SELECT TLM
ENABLE GATE
SERIAL DATA
ROUTING ADDRESS CLOCK
COMMAND MESSAGE CLOCK
EXECUTE PULSE
TO CDU

Upon acquiring bit synchronization, the remainder of the command word is bit sampled to convert it into a digital word.

3.9.2.5 Squelch. To conserve power, the digital logic portion of each decoder is not turned on until the FSK input signal is present. The detected signals from both channels are summed and filtered to produce a dc signal for deactivating the squelch. When this occurs power is then applied to the logic. See Figure 3.9.2-3. Because of the long time constants associated with the squelch circuit, it is required that at least 5 seconds be allowed between "successive commands," otherwise, accurate resynchronization cannot be achieved. If a shorter time between commands is desired, a continuous string of zeros must be transmitted between the completion of one command and the start of another. Note that "successive commands," as used above, means separate successive commands not contiguous (back to back) commands.

3.9.2.6 Digital Logic. When power is first applied to the digital logic, the logic is reset to an initial state. The programmer is synchronized by the first one in the data, the sync bit, and begins generating the clock and timing signals required for processing the data. See Figure 3.9.2-3.

The decoder address is first checked by each decoder. Both decoders will continue to process the command but only the addressed decoder will generate an enable signal. The selected decoder then generates the following output signals to the CDU in the time sequence shown below:

Enable Gate
Serial Data
Routing Address Clock
Command Message Clock
Execute Pulse

Another output signal, equivalent to the enable gate, is sent to the DTU for telemetry to indicate which decoder is supplying the output signals. If any one or two bits in the 15 bits comprised of routing address, command message, and parity are in error, the parity checks will detect the error and inhibit the generation of the execute signal.

3.9.2.7 Operational Considerations.

A) If one receiver fails in the "ON" state generating worse case noise at its output and the other receiver is within 1 db about threshold with only subcarrier present (no command modulation), a double noise amplitude will be generated at the input to both DDUs. Under these conditions, the DDU squelch can be repeatedly disabled and enabled by noise peaks.

Random disabling and enabling of the squelch may prevent processing of transmitted commands. This is because the programmer is indiscriminately starting up and is active when an actual command is present. This effect can be overcome by simply transmitting at least 22 zeros followed by the

normal command sequence. The string of zeros will reset the programmer and place it in a state to properly process commands.

Random disabling and enabling of the squelch may also cause processing of false commands. In order for this to occur, all of the following conditions must all be present at the same time:

1. One receiver must be failed and generating worse case noise output.
2. The good receiver must be within a few 1 db of threshold.
3. The noise must have a legitimate data and parity sequence for 22 bits.
4. The DDU must remain de-squelched for 22 bit periods.
5. There must be carrier present without tone modulation.

B) It is desirable to transmit an additional zero at the end of single command messages. This provides additional settling time for the DDU-CDU interface circuits. This insures that the enable output to the CDU will go to the zero state prior to commanding DDU power off of its output interface circuits.

C) When transmitting back to back (contiguous) commands, there should be at least two zero bits between the last parity bit of the previous command and the sync bit of the next command.

APPENDIX 3.9-A

DISCRETE COMMAND DECODING METHOD

The decoding method for discrete commands consist of a two level octal decode logic (see Figure A1). The first three bits in the 8 bit shift register (starting from left to right with the least significant bit first) are decoded into eight binary lines creating 8 ABC combinations by the first level decoding gates. The next three bits are similarly decoded into 8 DEF combinations. The last two bits (G & H) are decoded into 4 binary lines creating 4 GH combinations. In addition to the GH bits being decoded into 4 combinations, each is "and'ed" with an execute pulse received from the Digital Decoder Unit (DDU) and extended to approximately 50 milliseconds.

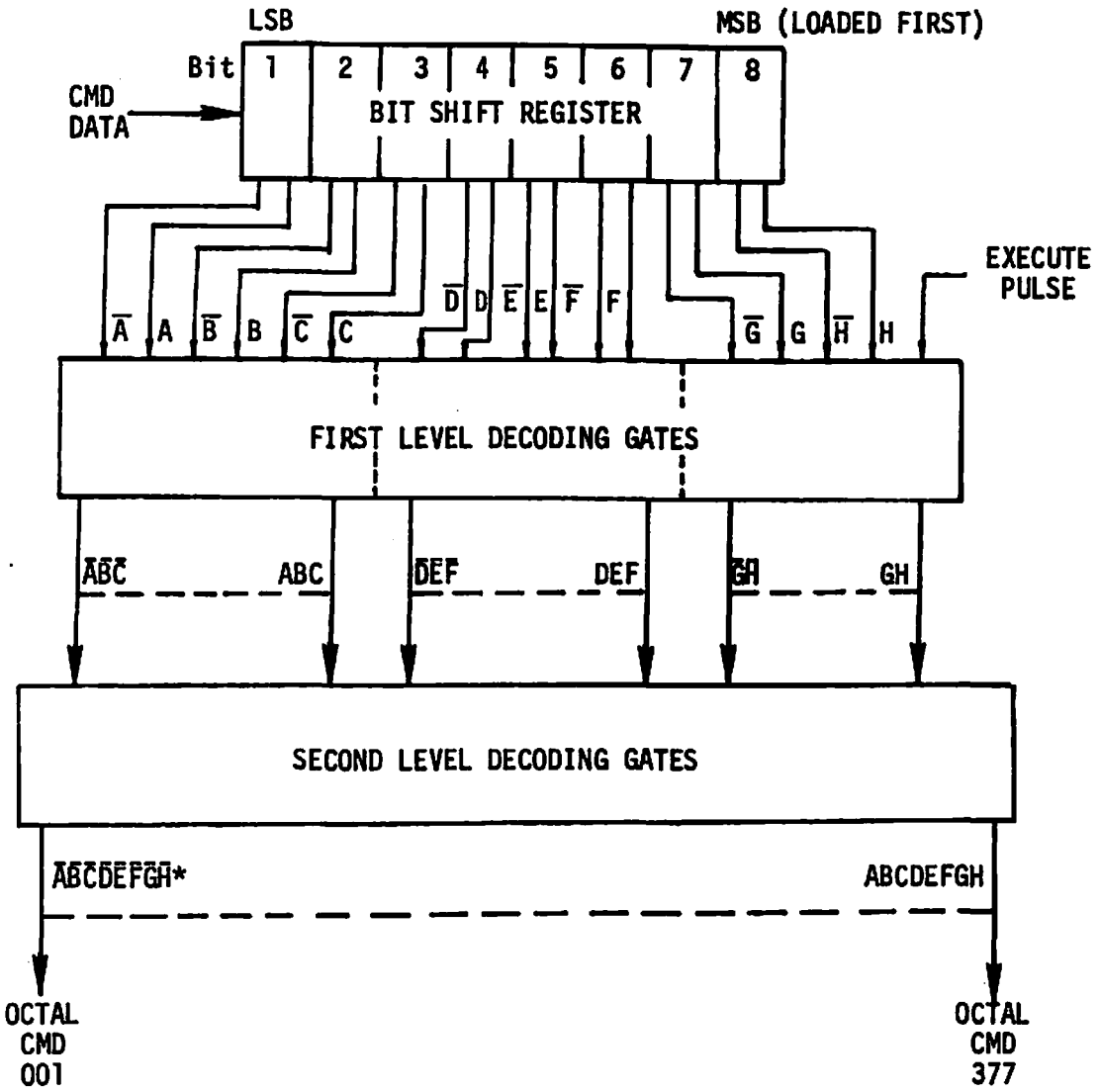
Each second level decoding gate takes one from each of the 3 binary combinations and completes the decoding.

By selectively connecting one combination from each of the 3 groups desired octal number for the discrete command is created (except for octal command 000 which is not decoded).

The conversion from the alphabetic combination to an octal number is as follows:

EXAMPLE

$H\bar{G}$	$\bar{F}\bar{E}D$	$C\bar{B}A$	
10	010	101	
200	20	5	4-2-1 Binary Weight
Octal Cmd Number	225		



* $\overline{A}\overline{B}\overline{C}\overline{D}\overline{E}\overline{F}\overline{G}\overline{H}$ (Octal CMD 000 is not used).

Figure A1.

APPENDIX 3.9-B

COMMAND MEMORY MODES OF OPERATION DESCRIPTION

The command memory consists of many modes of operation that overlap and intermingle. The modes of operation are designed to be synched to each other by a common 64 pps clock signal so that one mode does not interfere with another mode while in the process of reading or loading a register. The in-process mode is allowed to complete its function before the waiting mode begins (i.e., completion of command message loading into the telemetry register when an execution of that command is received).

The various modes of operation are:

1) Load the command and time message registers with data.

2) Timing of the execution and telemetry loading process (memory cycle control).

3) Execute contents of memory (execution cycle control).

4) Telemetry loading of the command and time register contents into the TLM register (TLM load cycle control).

The control process for the various modes of operation are the execution cycle control, memory cycle control and telemetry load cycle control (see Figure B1).

COMMAND AND TIME MESSAGE LOADING

When a command message and/or a time message is to be loaded into the register, a slot select command is sent to the command memory. This condition inhibits the telemetry readout of the register contents by means of the Z signal (see Figure B2 and Table BI), i.e., states D and E are prohibited from occurring in the memory cycle control. During the loading process, the "execute stored sequence" command should not be sent from the ground until verification of the contents of the registers via telemetry. After the time message data is received, the slot select position is automatically reset to a holding zero state. The slot select can also be reset to zero by a "slot select reset" ground command.

MEMORY CYCLE CONTROL (see Figure B2 and Table BI)

The mode operations of items 2 and 3 above must always be processed through item 4 since memory cycle control provides the timing function for serial output of the command and time message registers; therefore, the operation of item 4 will be mostly described as part of items 2 and 3.

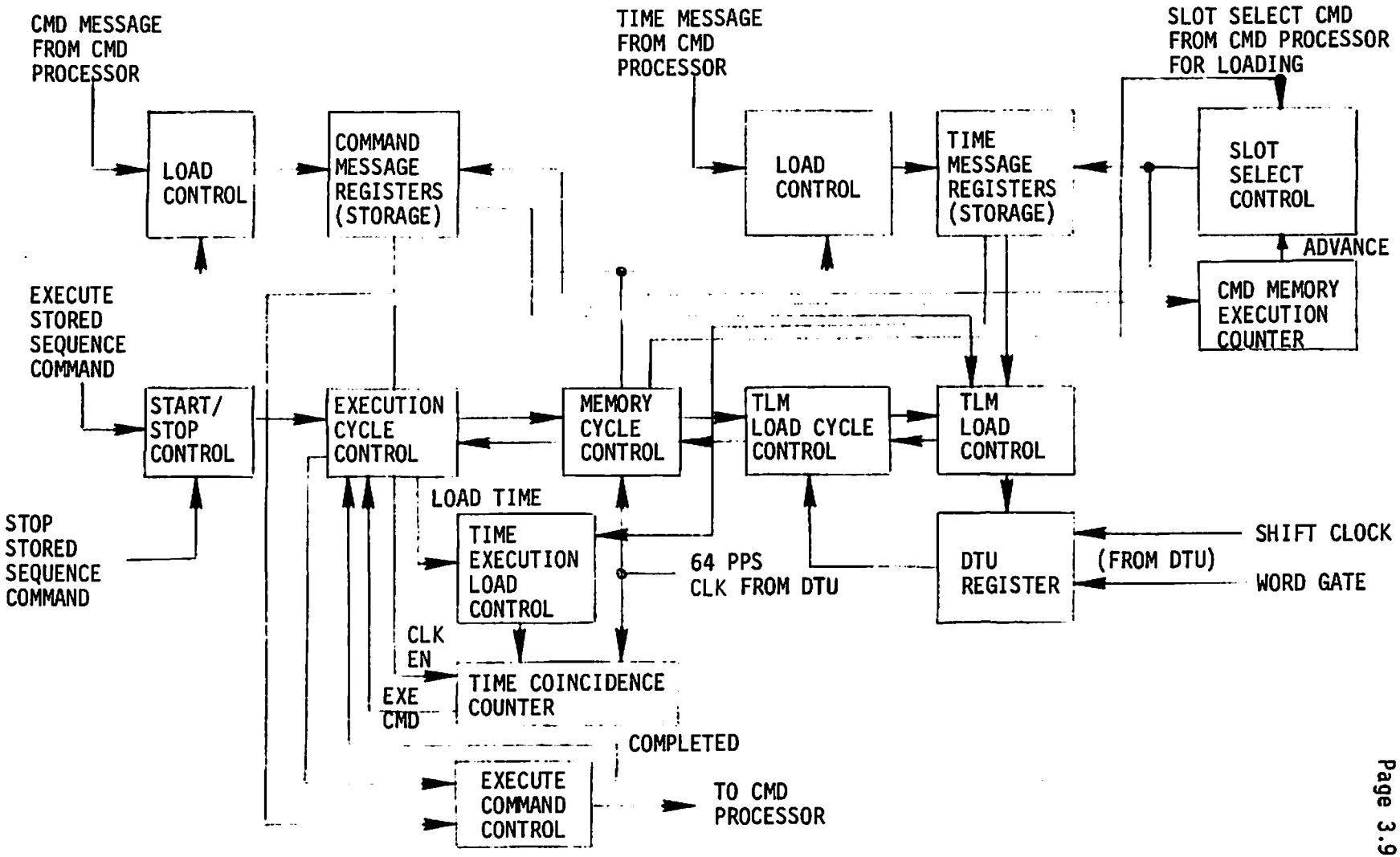


Figure B1. CDU Command Memory Block Diagram

Table BI. Symbol Definition

A = State where after the completion of a seven or eight count of the clock pulses are next memory cycle control state waiting is allowed to occur.

B = State where the time message stored is allowed to be read into the time coincidence counter.

C = State where the command message stored is allowed to be read into the selected command processor after time coincidence has occurred.

D = State where the time message stored is allowed to be read into the TLM register.

E = State where the command message stored is allowed to be read into the TLM register.

H = The holding state until an "execute stored sequence" command is received to start the stored command execution.

J = State where the execute of the stored command starts and/or the slot select control is advanced by one.

K = State where the time message stored is aloaded into the time coincidence counter.

L = Stage indicating the loading of the time message (state K) is in progress.

M = State where the timing out function for the command time delay is initiated in the time coincidence counter.

N = State where the command message is first sent to the selected command processor after the time delay has been timed out.

P = State indicating the sending of the command message (state N) is in progress.

W = Start stored sequence command signal (ground command).

$\overline{W}$ = Stop stored sequence command signal (ground command).

X = Signal indicating the slot select control is in the fifth register pair state during stored command execution.

$\overline{X}$ = Signal indicating the slot select control is in another state than the X state.

Y = Signal indicating that the time coincidence counter has time out and the selected stored command should be executed.

Table BI. Symbol Definition (continued)

Z	=	Signal indicating the command and time message register are in the state of being loaded from ground command and prohibit telemetry readout.
TR	=	Signal indicating that the TLM register is being read out to the DTU.
$\overline{TR}$	=	The complementary condition of TR.
WG	=	Signal indicating the receipt of the first word gate from the DTU.
3CP	=	Signal indicating the end of 3 clock pulse count for reading of the slot select address into the TLM register.
1	=	State initiating the loading of the time message stored into the TLM register.
2	=	State initiating the loading of the command message stored into the TLM register
3	=	State initiating the loading of the slot address for states 1 and 2 into the TLM register.
4	=	State indicating that the TLM register contents is being read out to the DTU.

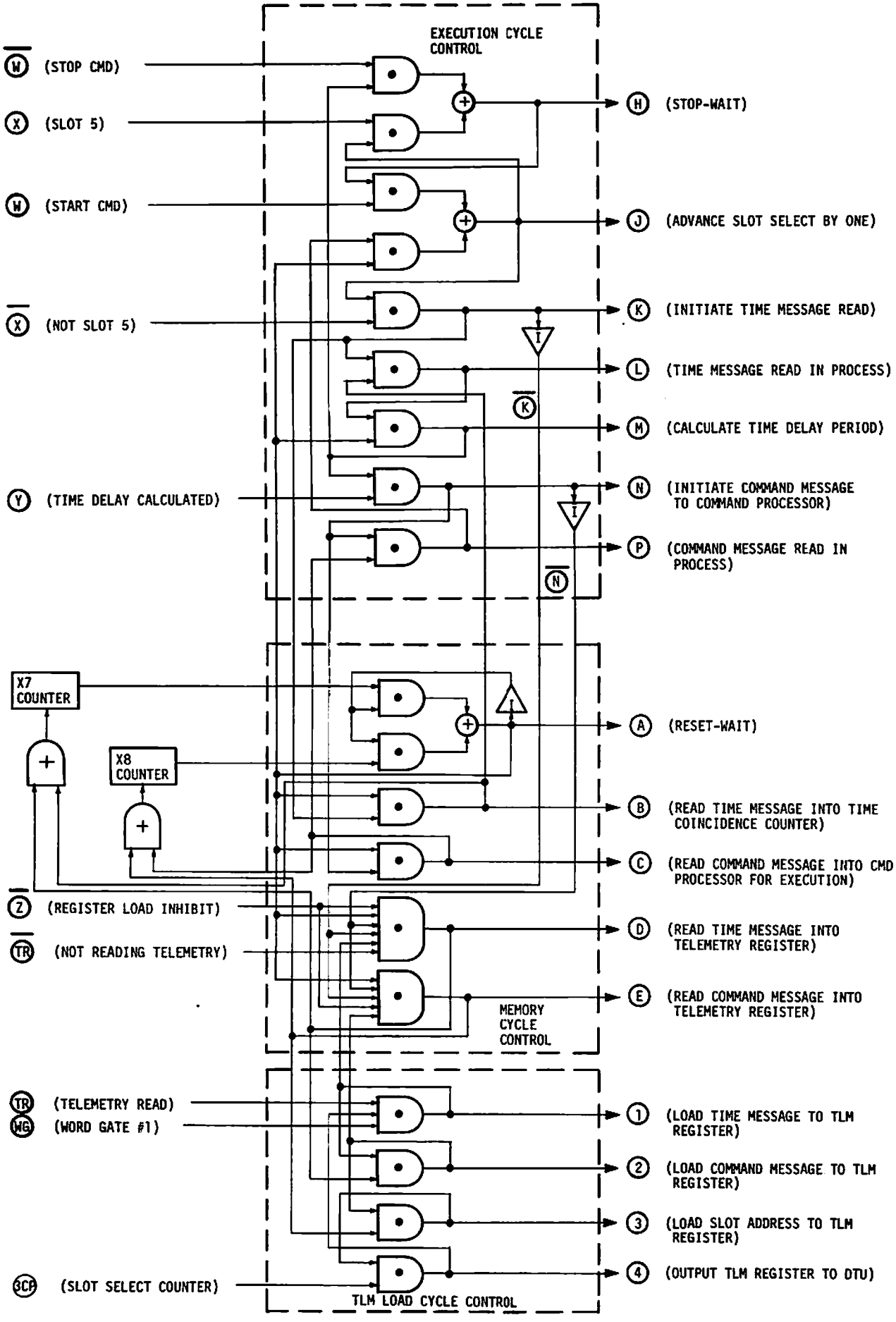


FIGURE B2. MEMORY CYCLE CONTROL

The timing cycling of the memory cycle control is such that after a timing out of 7 or 8 clock pulses, the state A is always activated before the next state is allowed. The timing cycling is always in the D or E state (telemetry readout), only interrupted when execution cycle control is operating.

The timing sequences are:

a) Quiescent (normal) condition

D→7 counts to→A→E→8 counts to→A→
D→7 counts to→A→E→8 counts to→A→Etc.

b) During command memory execution

D→7 counts to→A→E→8 counts to→A→
B→7 counts to→A→D→7 counts to→A→
E→8 counts to→A→C→8 counts to→A→
B→7 counts to→A→D→7 counts to→A→Etc.

(States B and C can appear anytime between D and E or E and D.)

The D and E states are prohibited during command and time message loading by the Z signal. In addition, during TLM register contents outputing to the DTU (see Figure 1) signal TR prohibits state D to occur and since E cannot appear before D, E is also prohibited.

EXECUTION CONTENTS OF MEMORY (see Figure B2 and Table BI) (Refer to Figure BI)

This execution operation is started after the command memory receives an "execute stored sequence" command (W), at which time the execute cycle control jumps into the J state, meaning the slot select control opens register position No. 1. One clock pulse later the operation goes into state K where the time message of slot number 1 is read into the time coincidence counter. On the next clock pulse, state L appears indicating that the time message reading is in progress for 7 clock pulses via state B of the memory cycle control. After the reading period, the memory cycle control jumps into the A (reset) state which allows the execution cycle control to go into the M state. The M state indicates the start and in-process condition of the time coincidence counter. Upon completion of the time delay period indicated by signal Y we initiate reading of the command message into the selected command processor (state N).

The state P is activated one clock pulse after N indicating the in-process condition of the reading, only if, the memory cycle control state C has occurred. Upon completion of the command message reading consisting of eight clock pulses the state J is once more activated to step the slot select control to register position number 2.

The execution of the second through the fifth register position procedes in the same manner as register position number 1. At the end of the fifth slot select position, the execution cycle control will go into

the H state indicating stop and wait by the X signal and the activation of the J state for the sixth position which does not exist.

If the "stop stored sequence" command is sent in the middle of the execution operation, the execution cycle control must reach the M state before halting as indicated by $\overline{W}$ and M at the input of the H condition, so as to not destroy or alter the contents of the time or command message registers.

TELEMETRY LOADING (see Figure B2 and Table BI) (Refer to Figure BI)

The TLM load cycle control allows the TLM register to be loaded with the command message, time message and the selected slot position. When the TLM register is not being read out as indicated by signals TR and WG the TLM load cycle control will start with state 1 allowing the memory cycle control to go to state D and time out 7 clock pulses for loading of the time message data contents into the TLM register.

At the time states 1 and D occur, state 2 is activated allowing state E to begin as soon as state D is completed. State E cannot occur until state A is reached indicating the end of state D.

At the end of state 2 and E state 3 occurs loading the slot address (3 bits long) into the TLM register. The 3 clock pulse indication (3CP) signals the TLM load cycle control that the TLM register is fully loaded and ready to be read out. At this time state 4 goes high and waits for 3 word gates to read out the TLM register contents. At the completion of the readout, the control will return to state 1.

SECTION 3.9 NOTATION

A	Analog
AC	Alternating Current
ACS	Attitude Control System
BPS	Bits Per Second
C1	Command Bit 1
CDU	Command Distribution Unit
CLK	Clock
CKT	Circuit
CMD	Command
CP	Command Processor
D	Digital
DC	Direct Current
DDU	Digital Decoder Unit
DTU	Digital Telemetry Unit (Described in Section 3.5)
FF	Flip-Flop
FSK	Frequency Shift Key
GC	Ground Command
GND	Ground
KHz	KiloHertz, cycles per second
LSB	Least Significant Bit
Ms	Millisecond
MSB	Most Significant Bit
NRZ,NRZ-L	Coding of bilevel-serially transmitted signals, having fixed designations of lower level representing 0 and higher level representing 1.
O/L	Overload
ORD	Ordnance
OSC	Oscillator
PPS	Pulse Per Second
PCU	Power Control Unit (Described in Section 3.7)
R1	Routing Bit 1
RCVR	Receiver
RTG	Radioisotope Thermoelectric Generator (Described in Section 3.8)
SCT	Spin Control Thruster
SCR	Silicon Controlled Rectifier
S/N	Signal to Noise Ratio
SP	Signal Present
SW	Switch
TLM	Telemetry
U/V	Undervoltage
VAC	Volts, Alternating Current
VCC	Power Supply Voltage (5 volts)
VDC	Volts Direct Current
VPT	Velocity Precession Thruster
XMTR	Transmitter

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4. SPACECRAFT OPERATING MODES (NORMAL)

This section addresses several specific spacecraft operating modes. These are associated with operations or series of operations which take place at a specific time or at repeated times during the mission. Each time such an operating mode is employed, it may have a duration of as little as an hour or less, or as much as 8 hours or more.

Propulsive operations of the spacecraft are included in this section, in conjunction with the closed-loop precession maneuvers utilized in pointing the spacecraft toward the earth, with the open-loop precession and steady ΔV maneuvers used in a trajectory correction, and in precession maneuvers (open or closed loop) for attitude changes at those times in the mission when the sun, earth, and spacecraft are aligned. The first post-launch sequence is treated, as is the operating mode for acquiring lock on the reference star.

The normal form of these operating modes is presented in this section. The backup forms, in which contingent provisions are used to counter anomalous situations, are treated in Section 5.

4.1 INITIAL FLIGHT MISSION OPERATIONS

Initial flight mission operations encompass all actions of the spacecraft which are controlled by the on-board automatic sequencer (a portion of the CDU discussed in Section 3.9). These actions commence with the separation of the spacecraft from the launch vehicle third stage (an event which is controlled by the launch vehicle third stage timer), 120 seconds after third stage burnout. They include the closing of the spacecraft separation switch, which starts the sequencer, the use of the spacecraft despin thruster to reduce the spin rate from $60 \pm \frac{2}{3}$ to 22 ± 1.5 rpm, the deployment of the two pairs of RTGs, the deployment of the magnetometer boom, and the initial orientation maneuver. They end with the termination of this open-loop precession maneuver.

The spacecraft events comprising the initial flight mission operations are controlled autonomously by the spacecraft; no commands need be transmitted from ground to spacecraft to implement them. The time intervals between the initiation of successive events are "wired" into the sequencer during its manufacture. The duration of each event (except initial orientation) are predictable functions of conditions which prevail. For example, the duration of despin depends on the actual spin rate at third stage burnout, and the duration of appendage deployments depends primarily on the temperature of the viscous dampers which restrain these deployments. The magnitude and direction of the initial orientation maneuver are entered and stored in the appropriate registers of the CEA before the spacecraft is launched. The duration of this orientation maneuver can be accurately predicted once the magnitude is chosen.

(The magnitude and direction of the initial orientation maneuver will be chosen and instituted by Mission Operations personnel according to the estimated attitude of the launch vehicle at injection and the desired

orientation of the spacecraft after the maneuver. The former is a result of the powered-flight trajectory, and varies--if expressed in celestial coordinates--according to time and day of launch. The latter should be within 10 degrees of the direction from the spacecraft to the earth 2 days after launch, but its actual selection may be influenced by these factors: The necessity of restricting the time for which the spacecraft dwells in a thermally adverse attitude; the fact that the direction of the earth-spacecraft line changes rapidly during initial hours after injection, but soon settles down to a slow variation of less than one degree/day; the fact that the initial orientation maneuver is appropriately conceived as a precession thruster calibration opportunity, discussed in Section 4.1.5; and the use of subsequent fine earth-pointing as an attitude calibration for performing trajectory correction maneuvers.)

4.1.1 Near-Earth Sequence

Nominal timing of the sequencer-controlled events of the initial flight mission operations is given in Table 4-1.

The time intervals set into the sequencer to implement this sequence are given in Table 4-2.

4.1.2 Spacecraft Spin Rate

The expected limits of the spin rate of the spacecraft from spin up (during the launch sequence) through the initial orientation maneuver are shown in Figure 4.1-1.

4.1.3 Sequencer Operations and Associated Events

A simplified block diagram of the spacecraft sequencer timer is shown in Figure 4.1-2. As shown in this block diagram, the control flip-flops FF #1 and FF #2 are initially reset to inhibit and to stop respectively when power is applied to the sequencer. When either separation switch closes, the state of the two flip-flops are changed to Enable (FF #1) and to Start (FF #2). If these flip-flops are not initially set to the desired conditions or fail to change state at separation as indicated by telemetry (words C-132 bit 6, sequence enable/inhibit, and C-332 bit 1, sequencer power on/off) then backup ground commands are provided (sequencer start, SEQ 1 and sequencer inhibit, SEQ 2).

For proper sequencer operation, the Enable/Inhibit flip-flop must be set (CMD SEQ3) prior to or simultaneously with setting of the Start/Stop flip-flop (CMD SEQ1). If the Start/Stop flip-flop is set to the Start condition, while the Enable/Inhibit flip-flop remains in the Inhibit condition, the Start/Stop flip-flop must be reset. This is accomplished by selecting the other +5 volt bus by ground command (CDU3 or CDU4); a momentary voltage drop is detected on the +5V or'd bus by the Power Reset circuit which then generates a Power Reset signal.

Once the separation switches close and the sequencer operation is initiated, the power gate circuit applies +5 volt power to the sequencer counter, reset circuit, counter inhibit flip-flop, gating circuits and the

Table 4-1. Near-Earth Sequence of Events

This table lists the normal sequence of events. Alternate modes are not identified, wherein sequencer initiation of events is inhibited, and ground command backup is employed.

No.	Time After Liftoff	Event	Initiated By	Remarks
1.	15m 55s	1) Separation 2) Initiate Sequencer (t_o)	Launch Vehicle S/C Separation Switch (t_o)	PC-221; separation not later than 120 sec after injection. May be more like 30 sec, pending third stage motor thrust tail-off tests.
2.	16m 59s	1) Initiate Despin 2) Arm Ordnance	Sequencer ($t_o' = t_o + 64s$)	Duration 52 to 127 sec, depending on separation spin rate, in the range of 51 to 83 rpm (123 $\pm$ 0.05 lb thruster).
3.	17m 47s	Thruster Verification: Enables event #5	Sequencer ($t_1 = t_o + 112s$)	If despin thrusters are not firing, subsequent sequencer controlled events are not enabled.
4.	17m 51s min to 19m 06s max	Terminate Despin	Despin Sensor	Despin sensors stop despin at 22 $\pm$ 1.5 rpm (at t_o + 116s min)* (at t_o + 191s max)*
5.	21m 55s	Initiate RTG Deployment	Sequencer ($t_2 = t_o + 360s$)	Initiation approximately 5 min after event #2 to assure adequate capacitor charge time.
6.	21 min to 28m max	S/C Emerges from Eclipse	----	Depends on launch date.
7.	31m 55s	RTG Deployment Completed	----	Duration 10 minutes max. (at t_o + 960s)*
8.	32m 59s	Initiate Magnetometer Boom Deployment	Sequencer ($t_3 = t_o + 1024s$)	Also pulls pin from feed movement mechanism

* Event times not sequencer controlled.

Table 4-1. (continued)

No.	Time After Liftoff	Event	Initiated By	Remarks
9.	37m 59s	Magnetometer Boom Deployment Complete	----	Duration 5 minutes max. (at t_0 + 132s)*
10.	39m 23s	Start Initial Orientation Maneuver	Sequencer ($t_4 = t_0 + 1408s$)	This maneuver precesses the spacecraft from injection attitude to near earth-pointing attitude. Precession magnitude is 165 $\pm$ 15 deg. Precession uses 0.125 sec pulses, 1 pulse pair/rev, 0.3 deg/rev, duration 104 to 125 min at 4.8 rpm. Maneuver direction and duration is controlled by stored program inserted before launch. This maneuver may serve as a calibration of the selected pair of precession thrusters.
11.	143m min to 164 max	Initial Orientation Maneuver Complete	----	End of sequencer-controlled events.
12.	After Event 10	Disarm Ordnance	Ground Command	Places ordnance firing capacitors in discharged state. Can be executed any time after Event 10 when communications are established.
13.	After Event 10 (either before or after event 12)	Shut off sequencer	Sequencer (t_5)	Sets SEQ. Start/Stop in Stop to Inhibit Power to Gate thereby putting sequencer in low power steady state off mode. (Commands "start sequencer, SEQ1", "Sequencer Inhibit, SEQ2" and "Sequencer Enable, SEQ3" are back-up ground commands,)

Table 4-2. Sequencer Controlled Events as a Function of Time After Separation Switch Closure

No.	Sequencer Initiated Events	Event Symbol	Sequencer Event Times Referenced to Sequencer Initiation at Separation (seconds)
1	a. Separation Switch Closure b. Start Sequencer	t_o	0
2	a. Initiate Despin b. Arm Ordnance	t'_o	64
3	Thruster Verification for RTG Deployment Enable	t_1	112
4	Initiate RTG Deployment	t_2	360
5	Initiate Magnetometer Boom Maneuver	t_3	1024
6	Start Initial Orientation Maneuver	t_4	1408
7	Stop Sequencer	t_5	At any time after t_4 not to exceed 2560.0 sec.

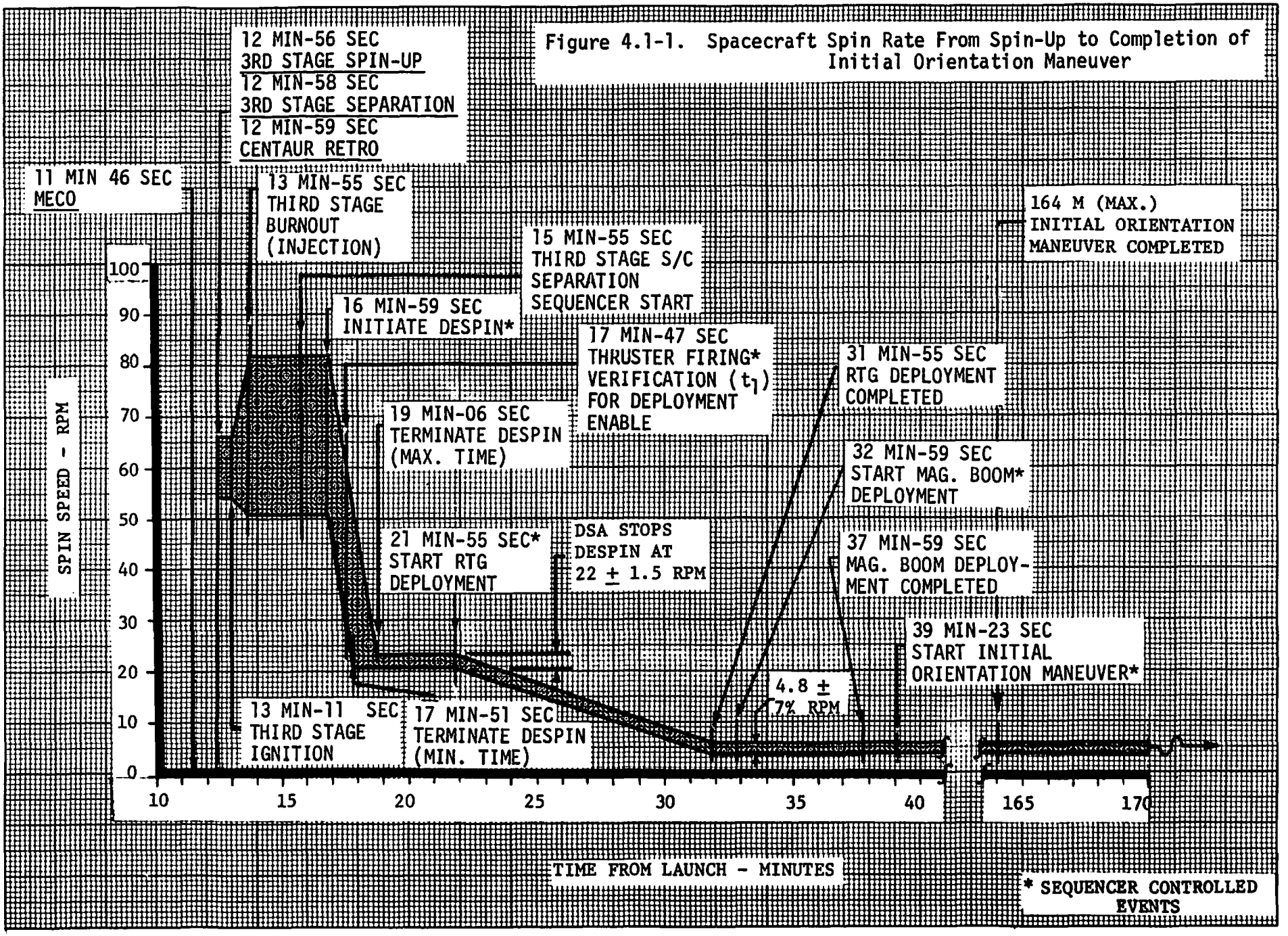


Figure 4.1-1. Spacecraft Spin Rate From Spin-Up to Completion of Initial Orientation Maneuver

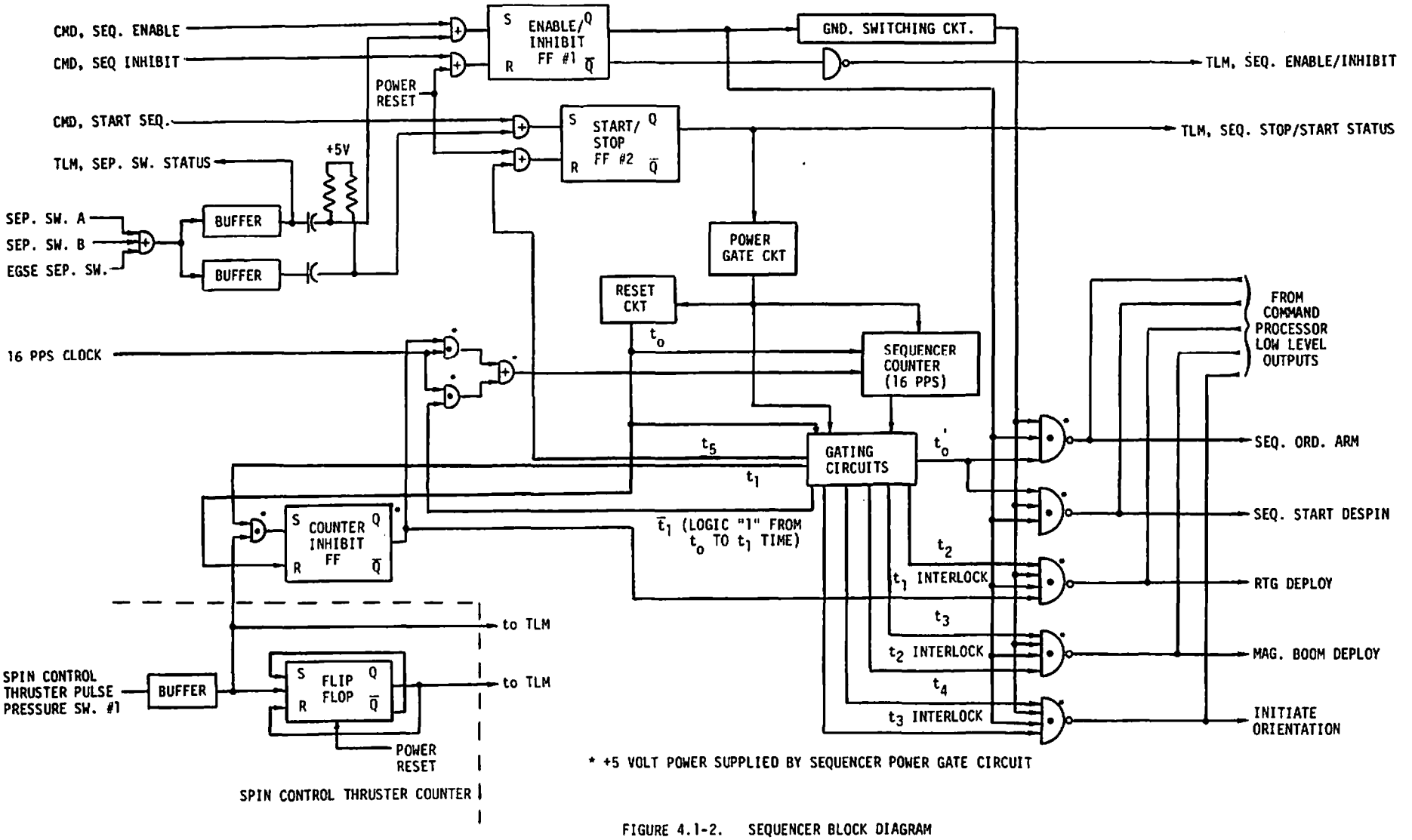


FIGURE 4.1-2. SEQUENCER BLOCK DIAGRAM

associated logic gates. As power is applied, the reset circuit resets the counter inhibit flip-flop and the flip-flops associated with the gating circuits and the sequencer counter. The sequencer counter counts the 16 pps clock and then issues the "commands" through the gating circuits and output NAND gates to initiate the various events.

At 16 minutes 59 seconds after launch (64 seconds after separation and the start of the sequencer operation), the "sequencer ordnance arm" and "sequencer start despin" commands are issued. Despin will require from 52 to 127 seconds, depending on the resultant final spacecraft spin speed (51-83 rpm) after spacecraft separation from the third stage, and the thrust generated by the despin thrusters.

At 17 minutes 47 seconds after launch, 48 seconds after initiating despin, a despin thruster verification event automatically occurs. If the despin thrusters are firing such that pressure switch number 2* detects 31 psi (approximately 0.2 pounds of thrust) or greater, the pressure switch closes setting the gating circuits to permit sequencer initiation of RTG deployment.

After the thruster verification event, the spacecraft will continue to despin to 21 $\pm$ 1.5 rpm where the despin sensor(s) will deactivate despin. Whichever sensor senses the spin speed within its sensitivity range first, that sensor will cause the spin thrusters to stop firing. In any event, the spin speed of the spacecraft must be below approximately 24 rpm prior to RTG deployment otherwise the structure may suffer damage (see Section 3.1.2.1). If RTG deployment occurs when the spacecraft spin speed is above approximately 30 rpm a catastrophic failure of the structure would probably occur. The magnetometer boom can be successfully deployed with spin rates below 6.1 rpm.

All sequencer controller events can also be initiated by ground command. If ordinance firings are performed from the ground, at least 3 minutes between firings must be allowed for the condensers to charge.

Ground station monitoring and verification of the correctness of the spin rate of the spacecraft after despin and before initiating RTG deployment is recommended. This monitoring would provide operational protection against certain failure modes during the third stage/spacecraft separation (which cause premature closure of the separation switch), and during despin due to any anomalies in the despin operation.

At the conclusion of the sequencer initiated events (i.e., after commanding the start of the re-orientation maneuver) the sequencer will automatically turn itself off by changing the state of flip-flop #2 to "stop" which will inhibit the power gate circuit and place the sequencer in a low power standby mode. The ordnance may be disarmed by ground command before sequencer shut off to conserve power if necessary.

* Pressure switch number 1 is used in conjunction with the spin thruster while pressure switch number 2 is for the despin thruster.

4.1.4 Initial Orientation Maneuver

The initial orientation maneuver precesses the spin axis from its orientation after injection to an earth pointing attitude. This maneuver is essentially the same as a midcourse maneuver except that the path length is twice as long, about 178°. The initial orientation maneuver has two goals, attain earth pointing, and calibrate the thruster performance. The calibration is discussed in Section 4.1.5. An example of the geometric configuration for the initial oreintation is shown in Figure 4.1-3.

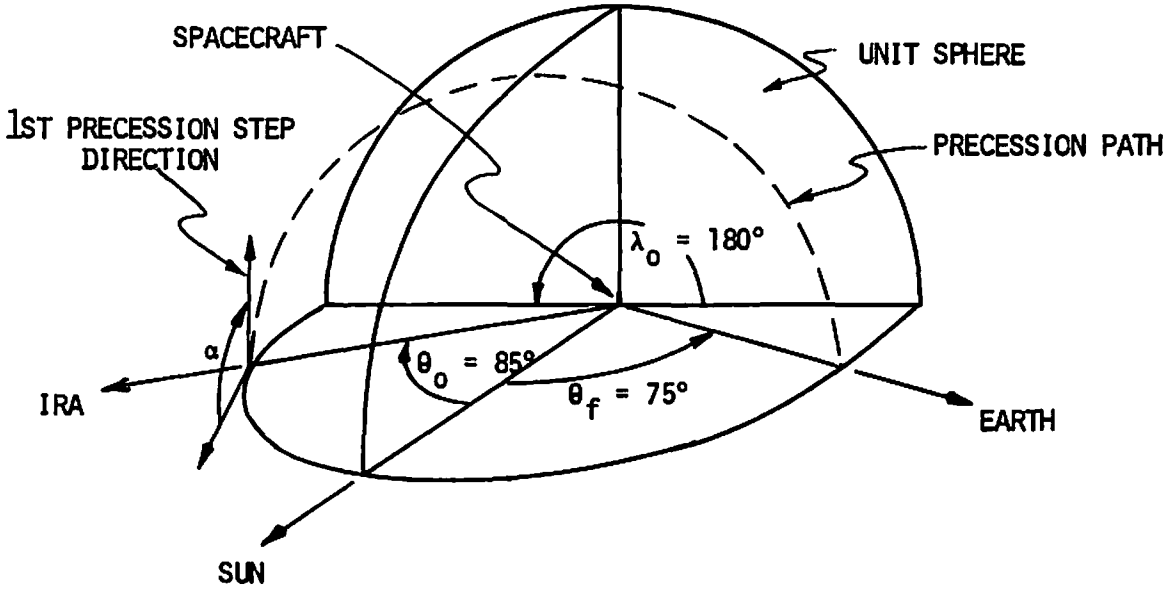


Figure 4.1-3. Initial Orientation Maneuver

The nomenclature and functional requirements for the initial orientation and other open loop precession maneuvers are given in Pioneer Document PC 263. In the example shown, the spin axis location at the start of the maneuver is defined as the initial reference axis (IRA), and is located at a sun look angle of $\theta_0 = 85°$. Precession at a constant angle (rhumb angle, α) to the plane of the sun will result in the precession path shown. Although the path is a rhumb line, it is close to a half cone in this example. The characteristics of the precession path are:

Initial Latitude, $90-\theta_0 = 15°$
Initial Longitude, $\lambda_0 = 180°$

Rhumb Angle, $\alpha = 86.76°$
Number of Precession Steps, $m = 592$
Precession Step Size, $\Delta\psi = 0.3°$

Final Latitude, $90-\theta_f = 25°$
Final Longitude, $\lambda_f = 0°$

The actual values that will occur for the initial and final longitudes and latitudes can range as much as $\pm 15°$ from the nominal values used in this example, and are strong functions of the launch date.

The precession direction is controlled by the Program Storage and Execution (PSE) assembly in the CEA (see Section 3.3.1). The thruster pulses used to precess the vehicle are controlled to occur at a constant angle to the plane of the spin axis and the sun. The basic reference for this angle is the sun sensor (Section 3.3.1.6) which detects the sun when it passes through the +y, z plane of the spacecraft. The sun pulse drives the spin period sector generator (SPSG), the output of which is the roll index pulse reference used in the PSE.

The output of the SPSG can be selected to either coincide with the sun pulse (0°) or occur 180° later. Similarly, one of two thruster pairs may be selected to execute the precession pulses. One pair applies the torque (direction of precession) in the direction of the +x axis, the other pair along the -x axis. The orientation directions of the spacecraft x_b and y_b axes at the instant of sun pulse occurrence are shown in Figure 4.1-4. The z_b axis is perpendicular to the page and completes the orthogonal triad. This figure shows the geometry of the rhumb angle (corresponding to the example of Figure 4.1-3) relative to the roll reference pulse and the orientation of the precession thruster pair. The angle required for the PSE is the angle α_R.

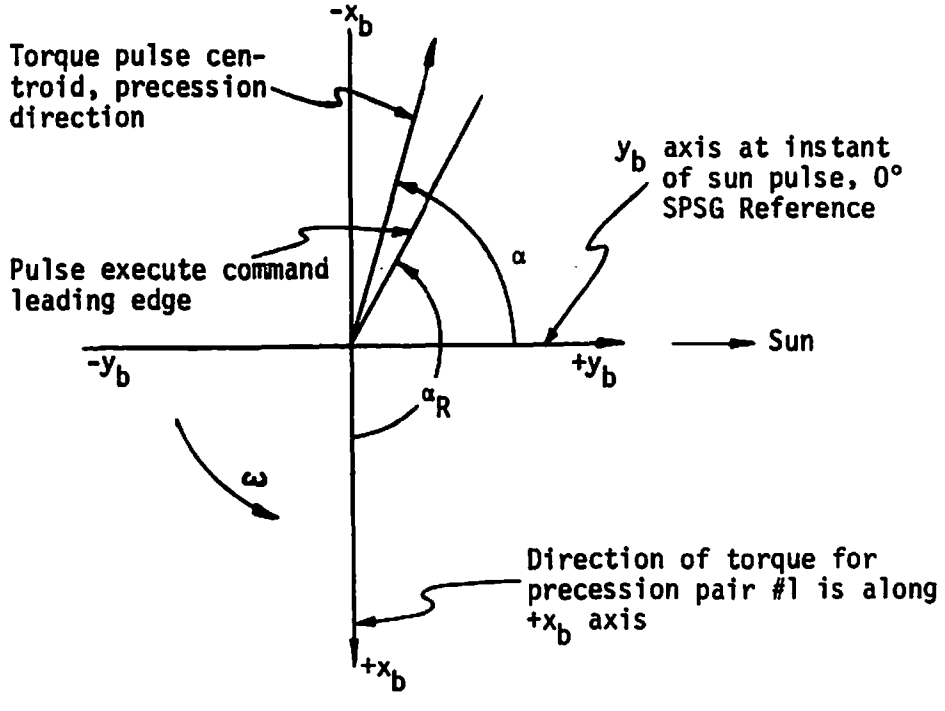


Figure 4.1-4. Rhumb Angle

Selection of the SPSG roll reference (0°, 180°) and the appropriate thruster pair, must be made in determining the value of the angle to be entered in the ACS angle register. This angle is given by

$$\alpha_R = \alpha - \delta_c \pm 90°$$

where α is the angle between the sun plane and the path, δ_c is the correction from the beginning of the thruster command to the impulse centroid, and the $\pm 90°$ correction depends on commands used for 0° - 180° roll reference and the thruster pair used. If the 0° reference is used from the spin period sector generator, (command RIP 1) and the +x torque thruster pair is used (command PRE 1) and if the angle between the execute signal and the pulse centroid is 2.09°, then the angle to be entered into the precession register is

$$\alpha_R = 86.76° - 2.09° + 90° = 174.67°$$

Since the PSE registers for entering α_R are quantized, the actual α that will be executed differs slightly from the α desired. This quantized α is called α_e. In this example, α_e = 86.46° where the ideal α is 86.76° (see Sections 4.1.5 and 3.3.1.3.2). The correction of $\pm 90°$ indicated in the equation for α_R depends on the SPSG reference and on the precession pair selected. The value of this correction is presented in the following table.

SPSG Command--Reference	Thruster Pair	Correction
RIP 1 (0°)	PRE 1 (+x)	+90°
	PRE 2 (-x)	-90° or +270
RIP 2 (180°)	PRE 1	-90° or +270
	PRE 2	+90°

(RIP 3, ACS mode of the SPSG, is normally used for all precession programs.)

The initial orientation maneuver is simply an open loop precession maneuver, and is discussed further in Section 4.4. To generate the various parameters for this maneuver, the "great circle" program discussed in Section 4.4 should be used as described. The time of execution of the maneuver must be taken into consideration since the desired final pointing direction (some point on earth, or some point that the earth will be several days after launch) will be changing its position fairly rapidly.

4.1.5 The Calibration Maneuver

The initial orientation maneuver discussed in Section 4.1.4 is the first and best opportunity to calibrate the precession thrusters. The calibration should include measurements of error in the pulse train total impulse magnitude (Impulse Uncertainty), in misalignment of the impulse, and in spin coupling. Because of the large precession angle during the initial reorientation, the initial and final spin axis attitude does not have to be known with great precision. The thruster calibration is accomplished by computation or measurement of the initial spin axis position, knowledge of the number of precession pulses, and measurement of the final spin axis position.

Before the calibration maneuver, a ground calibration of the thrusters will have been performed to establish the magnitude of the idealized initial reorientation maneuver. This will be used as the basis for comparison with the maneuver that actually occurs. For this comparison the following ground rules would be used:

a) Establish initial and final conditions of the ideal reorientation in terms of longitude (λ) and latitude ($\emptyset$). Trajectory, timing, and ground station availability must be considered.

b) Determine precision of the initial conditions and all errors in performance of reorientation (except thrust uncertainty).

c) Define characteristics and accuracy of the measurement of the final pointing using the medium gain conscan. If the final pointing is within the range of the high gain antenna, then the error in thrust calibration is the same order of magnitude of other system errors, and little practical improvement of impulse uncertainty can be obtained or is necessary.

The initial orientation of the vehicle can be established via injection errors and need only be determined within 2° to obtain the desired calibration accuracy of 4%. The number of pulses used for the maneuver is obtained from the velocity thruster pulse counters, (TLM words C-321, C-322, C-329, and C-330). These counters count up to 64 pulses and since about 600 pulses are expected for the initial orientation, the number of times these counters cycle should be observed. Spin period at the beginning and the end of the maneuver is obtained via TLM words C-405, C-406, and C-407. There are two techniques for measurement of the final spin axis position:

1) Telemetered parameters A sin θ and A cos θ obtained in the Conscan mode described in Section 3.6.2.9.5.

2) Comparison of the downlink received carrier power level with the measured antenna pattern.

The first procedure is simpler and requires minimum data processing, and is preferred.

4.1.5.1 Impulse Uncertainty and Misalignment (Centroid) Calibration. The analysis of the initial orientation/thruster calibration maneuver would include the following:

1) Determine sensitivity of each initial condition on final pointing. The errors in initial conditions (with the errors in execution) must be mapped into the final orientation (Region A, Figure 4.1-5) using items (a), (b), and the open loop precession program described in Section 4.4.

2) The error in measurement of the final position obtained from the A sin θ, A cos θ Conscan telemetry must be determined (Region B, Figure 4.1-5) using results from (c) above.

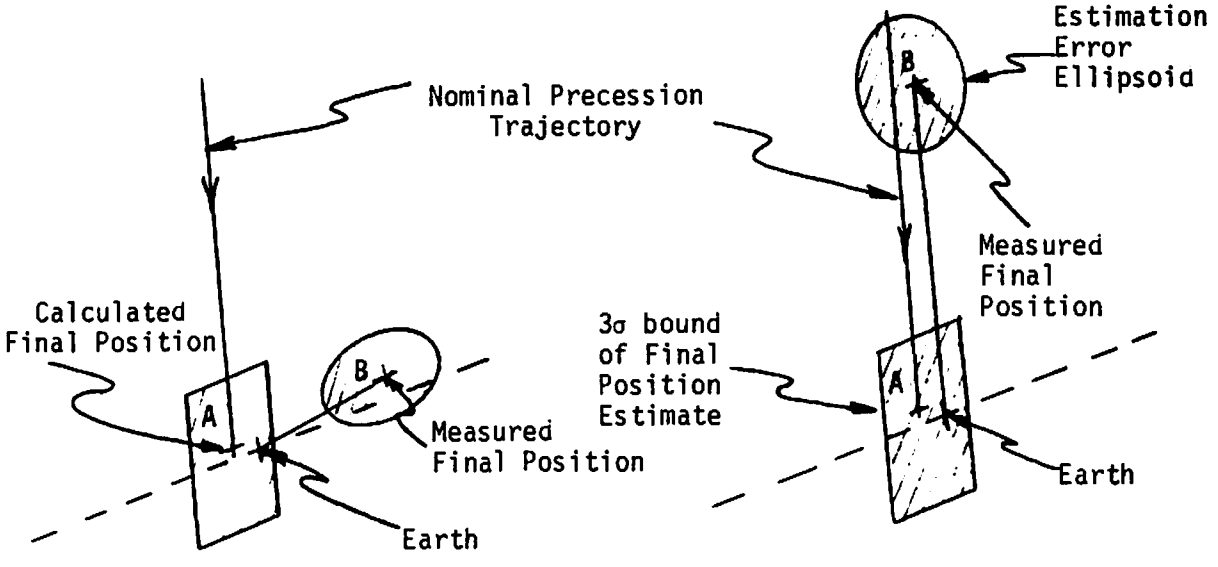


Figure 4.1-5. Correlation of Impulse and Angle Error Calibration Analysis

Based on these results, the error in impulse uncertainty calibration can be found and the confidence in determination of that error can be resolved. For example, Figure 4.1-5 shows an example where little correlation exists between impulse ground calibration error and the data. No improvement in the impulse uncertainty is necessary since the error in the length of the path (impulse uncertainty) is small and within the accuracy limitations of the measurements. The error in angle α does show significance since A and B do not overlap, indicating that the impulse centroid may need a correction for future maneuvers. Figure 4.1-5b is an example for a case where the estimate in pulse train total impulse was low. This result is highly

correlated with impulse error and relatively uncorrelated with errors in α (pulse centroid or misalignment error). In this case, the impulse estimate should be appropriately corrected for future maneuvers.

The character of Region A is shown in Figure 4.1-6. The x_e, y_e coordinate system is normal to the spacecraft-earth line with the y_e axis parallel to the ecliptic plane. For a given initial orientation and sun-earth geometry (same as the example in Section 4.1.4) errors in execution of the direction ($\delta\alpha$) and the magnitude ($\delta\psi$) of the precession path are mapped into a region about the earth. This is the region A shown. The various characteristics are described in the following paragraphs.

- Point 1 is the earth location. With no error in execution, and performance of the maneuver according to the continuous rhumb line equations, the reorientation would stop at this point.

- Since the reorientation is performed by a sequence of great circles, a perfect execution with zero error would give a final orientation at point 2. This error, caused by the difference between the continuous rhumb line equations and the sequence of great circles, is less than 0.1° for this example.

- The angle α is quantized at 0.703125°. If 248 bits of this size are used in the PSE register for maneuver execution (α_R) and the pulse centroid shift is 10 msec at 4.8 rpm, then the angle of execution is

84.375°	quantized (α_R - 90°)
+ 2.088°	centroid shift (1.8° + 0.29°)
86.463° ,	α_e = α executed relative to sun-spin axis plane

  The error between the ideal α and that executed is 86.766 - 86.463 = 0.303°. With this error, the trajectory will arrive at point 3.

- The sensitivity to one bit in α, (0.703125°) is shown by points 4 and 5. These are the final trajectory locations if α_e above is incremented plus or minus by this amount.

- The boundary of the execution error which defines region A is given by the mapping of the 4.8% error in impulse uncertainty in the final orientation region. This gives the upper and lower edges of the region (4.8% of 177.5° = 8.5°) as shown. The lateral boundary is measured from the nominal quantized α_e (point 3) and is the mapping of the trajectory with a 0.5° (3σ) error in the α used in this example.

4.1.5.2 Spin Coupling Error Calibration. The spin speed change for a worst case misalignment of the thrusters of 0.5° is expected to be about 0.00145 rpm per second of thruster "on" time. With 600 pulses for the maneuver, the "on" time would be about 75 sec and the 3σ spin speed change should be 0.108 rpm. The corresponding change in period of 0.275 sec (at a nominal 4.8 rpm) can be accurately measured via the spin period TLM which has a least significant bit of 0.000122 sec.

The spin speed change would reflect in the computations for impulse uncertainty and centroid location described previously, but would be a second order effect. The change in spin rate would produce a corresponding change in the precession step size, but the change in precession angle (α) would be negligible.

The spin coupling measurement is required to determine the maximum magnitude of ΔV that can be performed without spin control. Since a ΔV could last as long as 5600 seconds, the worst case spin coupling would result in an 8 rpm spin change. This cannot be permitted since a complete despin and loss of attitude control could occur.

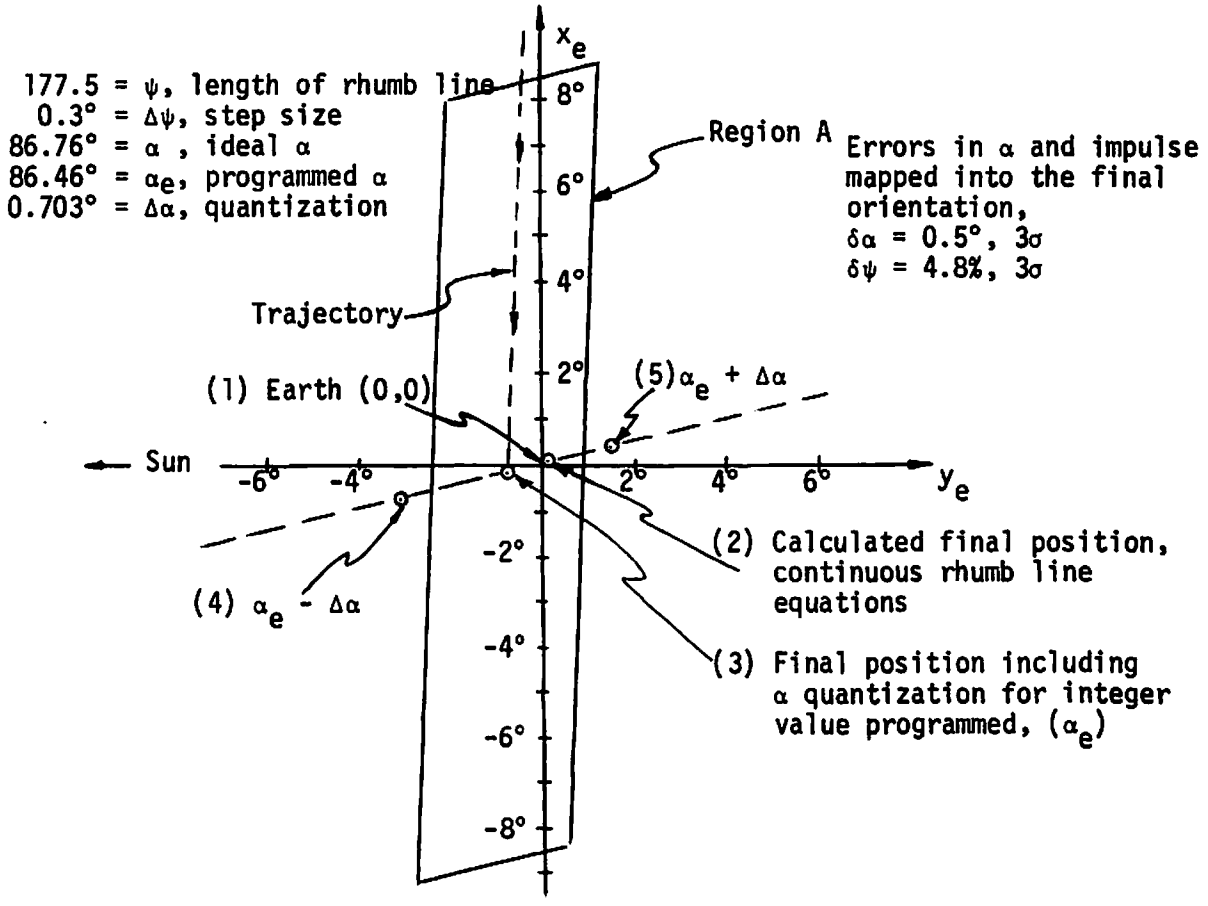


Figure 4.1-6. Projection of Angular Momentum Vector on Plane Normal to Spacecraft-Earth Line: Expected Error for Initial Acquisition Maneuver

4.2 CONSCAN CLOSED LOOP MANEUVERS

4.2.1 Introduction

The Conscan (conical scan) system is a closed loop attitude control system for pointing the Pioneer spacecraft spin axis toward an RF signal from the earth.

The closed loop error signal is generated by the offset directional antennas and the spinning spacecraft, and consists of a modulation of the amplitude of the received RF signal at the spin rate (0.08 Hz, nominally). The error signal contains necessary amplitude and phase information on the pointing angle between the spacecraft spin axis and the earth signal. Processing of this signal takes place to:

1) Provide the necessary timing information for firing precession thrusters at the appropriate instant in the spin cycle to reorient the spacecraft spin axis toward earth.

2) Provide a signal to indicate that the pointing error is below specified minimum threshold for termination of thruster firing.

3) Provide telemetry of the pointing error while the system is in the conscan mode (for pointing errors up to 10°) with an accuracy of better than 5% of full scale reading. (The Conscan Signal Processor must be on, and either the medium-gain or offset high-gain antenna in use; however, thrusters may be off.)

The critical units of the Conscan system have been described in Section 3.6, Communications Subsystem, and Section 3.3, Attitude Control Subsystem (ACS).

4.2.2 System Description

The block diagram of the conscan pointing system is shown in Figure 4.2-1. Either of two antennas can be utilized for Conscan operation. The medium gain antenna is permanently offset from the spin axis by about 9.3 degrees (1 db crossover level) and is used for course conscan. During the nominal mission the medium gain antenna will be used over the pointing angle range from 10 degrees (maximum required) to 1.3 degrees (deadzone corresponding to the threshold setting). Normally, the high gain antenna beam is concentric with the z axis and only will be offset by command by about 1 degree (1 db crossover) when fine conscan is required. With the high gain antenna, conscanning is possible over pointing error angles from 2 to 0.3° (deadzone corresponding to threshold setting). The antenna beam is tilted by offsetting the feed using the feed movement mechanism.

The amplitude modulated antenna output is coherently detected by the spacecraft receiver AGC. The AGC is required to remove conscan signal amplitude dependence on the received carrier power level which varies over a wide range with communication distance. The conscan signal (AGC control voltage) is recovered at the output of the AGC amplifier. The receiver AGC provides over 70 db dynamic range with linearity of $\pm$10 percent.

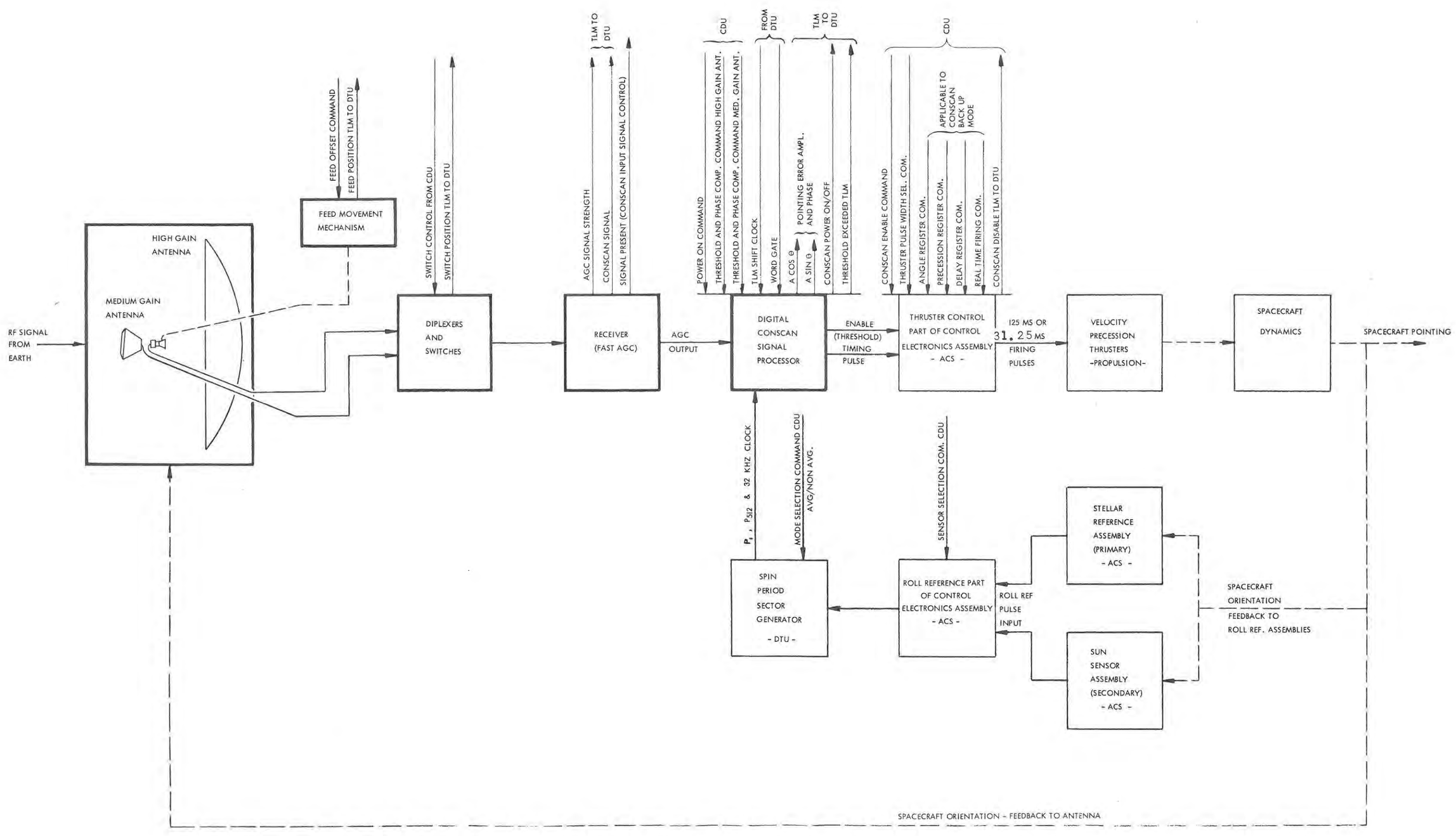


FIGURE 4.2-1. CONSCAN SYSTEM BLOCK DIAGRAM

The main function of the digital conscan signal processor (CSP) is to compute the zero axis crossing time and the amplitude of the conscan signal. The zero axis crossing time is used to time the precession thruster firing. The amplitude of the conscan signal is approximately proportional to the pointing error angle, and is used to terminate conscan when the deadzone has been reached. The CSP also provides telemetry of parameters A sin θ and A cos θ from which the pointing error angle magnitude and phase can be computed (see Section 3.6.2.9.4). This information establishes the spacecraft orientation relative to coordinates based on the spacecraft roll reference (Canopus or sun) and the spacecraft-earth line.

The timing or the firing pulse and the threshold signal from the CSP is sent to the control electronics assembly of the ACS. The two main functions of the control electronics assembly for conscan are to select the pulse length of the velocity precession thruster firing and to terminate precession when the pointing error is within the deadzone. Two step sizes will be used. A 125 millisecond pulse (approximately 0.29° per step) will be used for precession while in the medium gain antenna mode. A 31.25 millisecond firing pulse (about 0.06° step size)* will be used while in the high gain antenna mode. The larger precession step size can also be used with the high gain antenna, however, under normal operating conditions with the recommended smaller steps the deadzone can be more precisely established and the propellant consumption minimized.

The spin rate reference signal for the CSP is supplied by the spin period sector generator (SPSG). The SPSG receives its reference from either the stellar reference assembly or the sun sensor assembly. The SPSG counts an internal clock for one spacecraft revolution and digitally divides this number into a number of sectors for generating evenly spaced timing signals which are harmonically related to the spacecraft spin rate. Three modes of operation selected by ground command can be used; non spin averaging mode, spin averaging mode, and ACS mode. In the non-averaging mode, SPSG provides a spin rate reference (period between two adjacent roll pulses) but there can exist a phase error between roll reference input and output. This mode can be used for conscan operation; however, there can exist a phase error, relative to the roll reference, in the telemetered parameters A sin θ and A cos θ. In the spin averaging mode, the spin rate supplied to the CSP will be the average of the preceding 64 revolutions. It may be necessary to use this mode if conscanning with the spin axis near the sun-line and if the sun sensor is in use. The SPSG in the averaging mode reduces jitter on the roll reference, due primarily to wobble. The ACS mode is the normal mode of operation for the conscan system. As far as conscan is concerned, the ACS mode is the same as the non-spin averaging mode except that the SPSG output is in phase synchronization with the roll reference pulse. In this mode it is possible to get either 511 or 513 pulses from the SPSG instead of the usual 512 if the sun sensor is used while conscanning near the sun. This effect could cause some error in the conscan firing, but the error would be insignificant compared to other error sources.

*Step sizes correspond to full propellant tank; for less than full tank, smaller step sizes will result.

Two spurious wobble frequencies are generated by the precession maneuver which appear in the conscan signal. The frequencies are approximately 1.8 and 0.2 times the spacecraft spin rate. The amplitude of the wobble depends upon the precession step size and on the rate at which they are applied. A rate of pulse firing of one every 3 spacecraft revolutions has been chosen to minimize the high frequency wobble amplitude. Under normal operating conditions, as shown in the next section, wobble interference has a negligible effect on the conscan system performance.

4.2.3 Capability

Closed loop conscan performance is mainly determined by the phase (timing) and amplitude (pointing angle) errors. Primarily, the phase error determines the gas usage ratio, or the percent of wasted fuel, while the amplitude error establishes how reliably conscan can be terminated when the deadzone is reached.

The phase and amplitude error budgets are presented in Tables 4.2-1 and 4.2-2 for the medium gain and the high gain antennas, respectively. Note that most error magnitudes depend on the signal amplitude and noise, and that the AGC nonlinearity errors depend on the communication distance or the received power density. These tables only apply to the outer deadzones and maximum communication distances. For conscan with the medium gain antenna, at low received carrier power levels (large communication distances), errors due to noise predominate. In Figure 3.6-8 these errors are plotted as a function of received carrier power level. Errors not in the above tables and which do not significantly affect conscan performance include antenna second harmonics, phase error due to AGC nonlinearity, power supply variation, roll reference jitter, thrust vector misalignment, DSS ERP variation, antenna thermal effects, and propagation errors.

The effect of the phase errors on the system performance can be partly determined from the simulation results of Figure 4.2-2. In this figure, the gas (fuel) usage ratio versus phase bias is presented. The phase bias is a constant phase error computed separately in Tables 4.2-1 and 4.2-2. The amplitude errors do not directly affect the gas usage ratio; however, the propellant consumption is affected indirectly by the threshold setting accuracy since the phase errors increase with decreasing pointing angle. The gas usage ratio for random errors is not available; however, the conscan simulation trajectory of Figure 4.2-3 indicates that the effect of random errors is not serious. Observe that the case shown can be considered a worst case flight condition since it was obtained for conscan with the medium gain antenna at a range of 5×10^8km and a pointing angle of 2 degrees. Based on bias errors, the gas usage ratio for both antennas is less than 1.1.

Table 4.2-1. Conscan Error Budget for Medium Gain Antenna
(5×10^8 km Distance, 2° Pointing Angle)

Error Source	Nominal Phase Degrees		Nominal Amplitude Percent	
	1 Sigma	3 Sigma or Peak	1 Sigma	3 Sigma or Peak
A. Bias				
1. Antenna Phase Variation	--	- 6.6	--	--
2. Receiver AGC Nonlinearity	--	--	--	+ 10
3. CSP Phase Compensation and Threshold Settability	--	- 0.1	--	+ 0.2
TOTAL BIAS		- 6.7		+ 10.2
B. Random Before Launch				
1. Antenna Range	1.2	3.6	5.8	17.5
2. Mechanical Antenna Alignment	0.44	1.3	0.64	1.9
TOTAL RANDOM BEFORE LAUNCH	1.26	3.83	5.83	17.6
C. Slow Varying (Random)				
1. Third Harmonic	0.2	0.3	0.35	0.5
2. Polarization	1.9	2.6	3.3	4.6
3. CSP Algorithm	1.4	2	5	7
4. Spin Rate Variation	1.6	2.8	0.2	0.3
TOTAL SLOW VARIATION	2.92	4.32	6.87	8.4
D. Fast Varying (Random)				
1. Noise	18	54	34	102
2. Wobble	1.34	1.9	2.4	3.4
TOTAL FAST VARYING	18	54	34	102
TOTAL RANDOM	18.3	54.3	35.1	104

Table 4.2-2. Conscan Error Budget for the High Gain Antenna
(9×10^8 km Distance, 0.5° Pointing Angle)

Error Source	Nominal Phase Degrees		Nominal Amplitude Percent	
	1 Sigma	3 Sigma or Peak	1 Sigma	3 Sigma or Peak
A. Bias				
1. Antenna Phase Variation	--	+ 2.4	--	--
2. Receiver AGC Nonlinearity	--	--	--	+ 4
3. CSP Phase Compensation and Threshold Settability	--	+ 0.05	--	0
TOTAL BIAS		+ 2.4	--	+ 4
B. Random Before Launch				
1. Antenna Range	1	3	1.1	3.4
2. Mechanical Antenna Alignment	4	12	6.8	20
TOTAL RANDOM BEFORE LAUNCH	4.1	12.3	6.9	20.7
C. Slow Varying (Random)				
1. Antenna Third Harmonic	0.35	0.5	0.57	0.8
2. Antenna Polarization	0.6	0.86	1.1	1.5
3. CSP Algorithm	1.4	2	5	7
4. Spin Rate Variation	1.6	2.8	0.2	0.3
TOTAL SLOW VARYING	2.24	3.56	5.15	7.2
D. Fast Varying (Random)				
1. Noise	0.9	2.7	1.8	5.4
2. Wobble	0.85	1.2	1.5	2.1
TOTAL FAST VARYING	1.25	2.96	2.34	5.8
TOTAL RANDOM	4.1	13.1	8.9	22.7

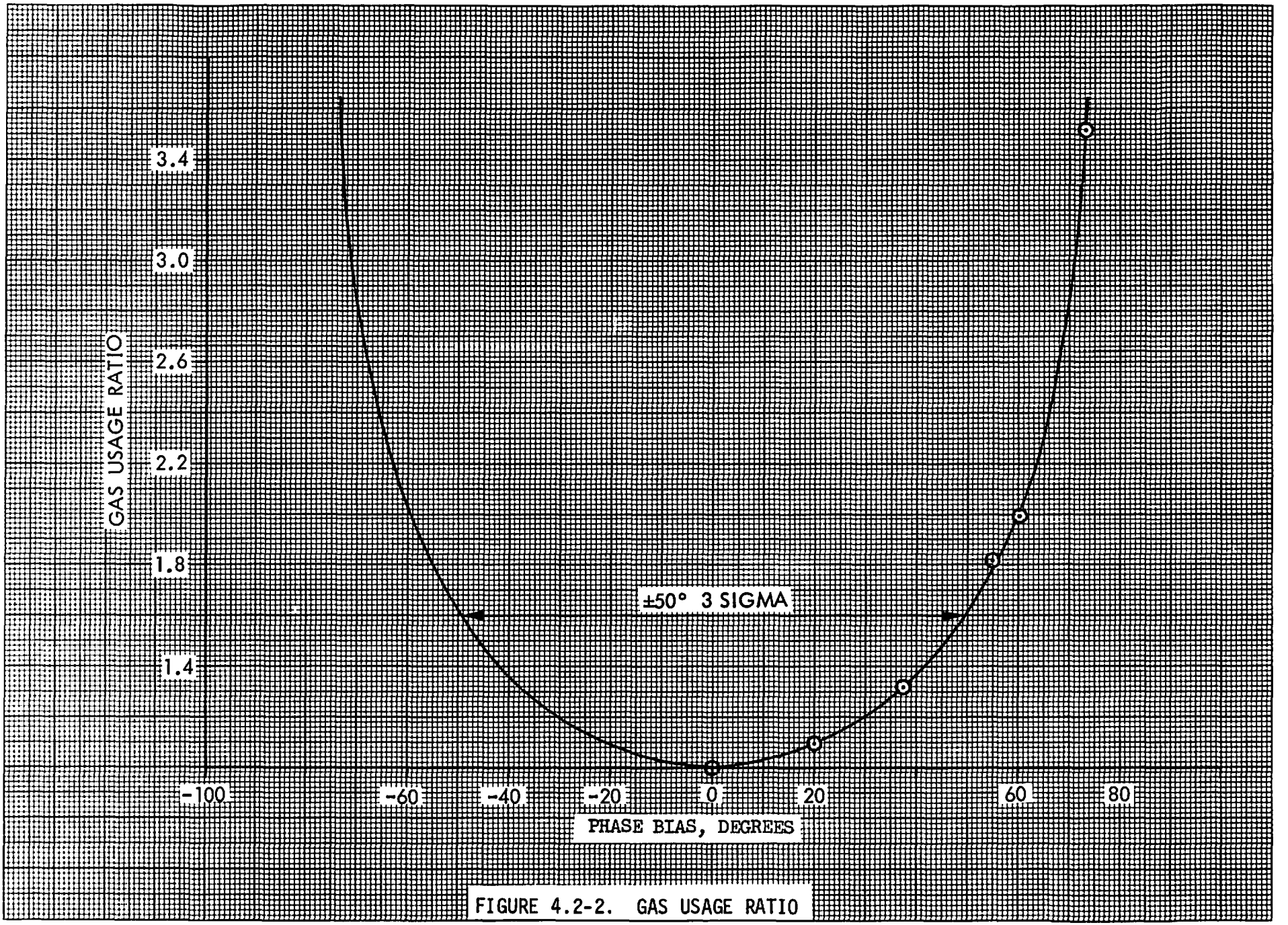


FIGURE 4.2-2. GAS USAGE RATIO

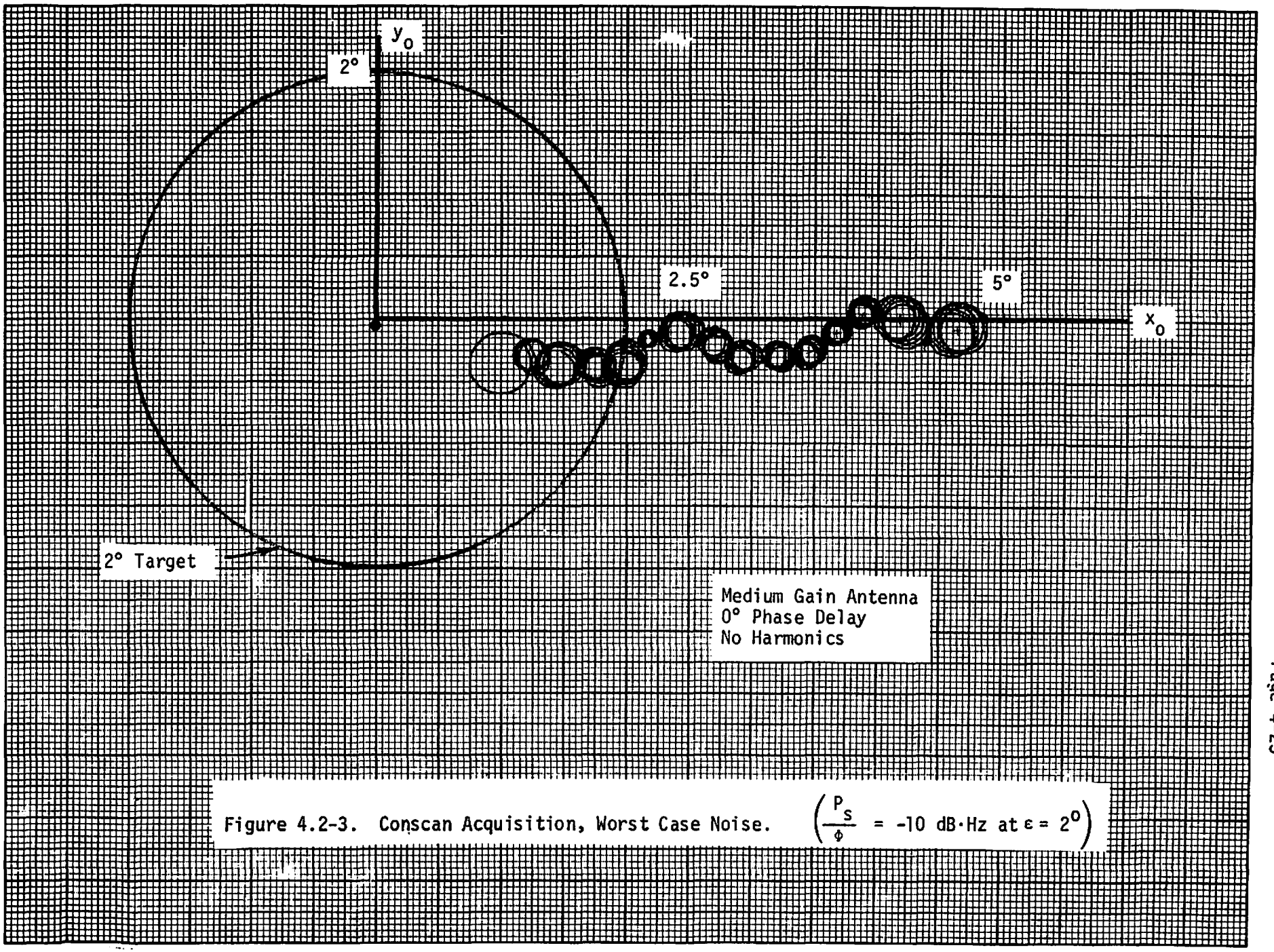


Figure 4.2-3. Conscan Acquisition, Worst Case Noise. $\left(\frac{P_s}{\phi} = -10 \text{ dB·Hz at } \varepsilon = 2^o\right)$

The amplitude errors determine how accurately conscan can be terminated at the threshold. For the medium gain antenna the threshold will be set at a pointing angle of 1.3 degrees and for the high gain antenna it will be set at 0.3 degrees. For the high gain antenna the amplitude errors listed in Table 4.2-2 are small and conscan operation will be terminated with high probability at 0.3 degrees. For the medium gain antenna, at larger communication distances, the amplitude errors will be significant and conscan, in general, will be terminated early as indicated in Figure 4.2-4. Based on this figure it is recommended that with the 85 foot 10 kw DSS, conscan with the medium gain antenna be performed at distances less than 2.5×10^8 km (-137 dbm) with the absolute threshold of 5×10^8 km (-143 dbm). At low received carrier power levels, the modulation should be off since this causes a 2.8 dbm signal reduction. The DSS power limitations for conscan are given in Section 3.6, Figures 3.6-9 and 3.6-10.

Telemetry (parameters A sin θ and A cos θ) error budgets for the high gain and the medium gain antennas are presented in Tables 4.2-3 and 4.2-4, respectively. The telemetry error is dominated by noise and truncation errors. These error budgets are the same as the conscan error budgets except that the AGC nonlinearity is partly calibrated out on the ground and the residual error is due to the AGC slope measurement inaccuracy. Since the post alignment spin axis shift due to propellant consumption can be predicted accurately on the ground, this error has not been included in the mechanical misalignment. Note that the antenna phase variation (conservatively) has not been compensated for since the antenna pattern contributions cannot be separated completely from the range errors.

The error budgets of Tables 4.2-3 and 4.2-4 are valid for a single measurement. By repeated measurements or averaging of N telemetry values, the random, high frequency error can be reduced by $\sqrt{N}$.

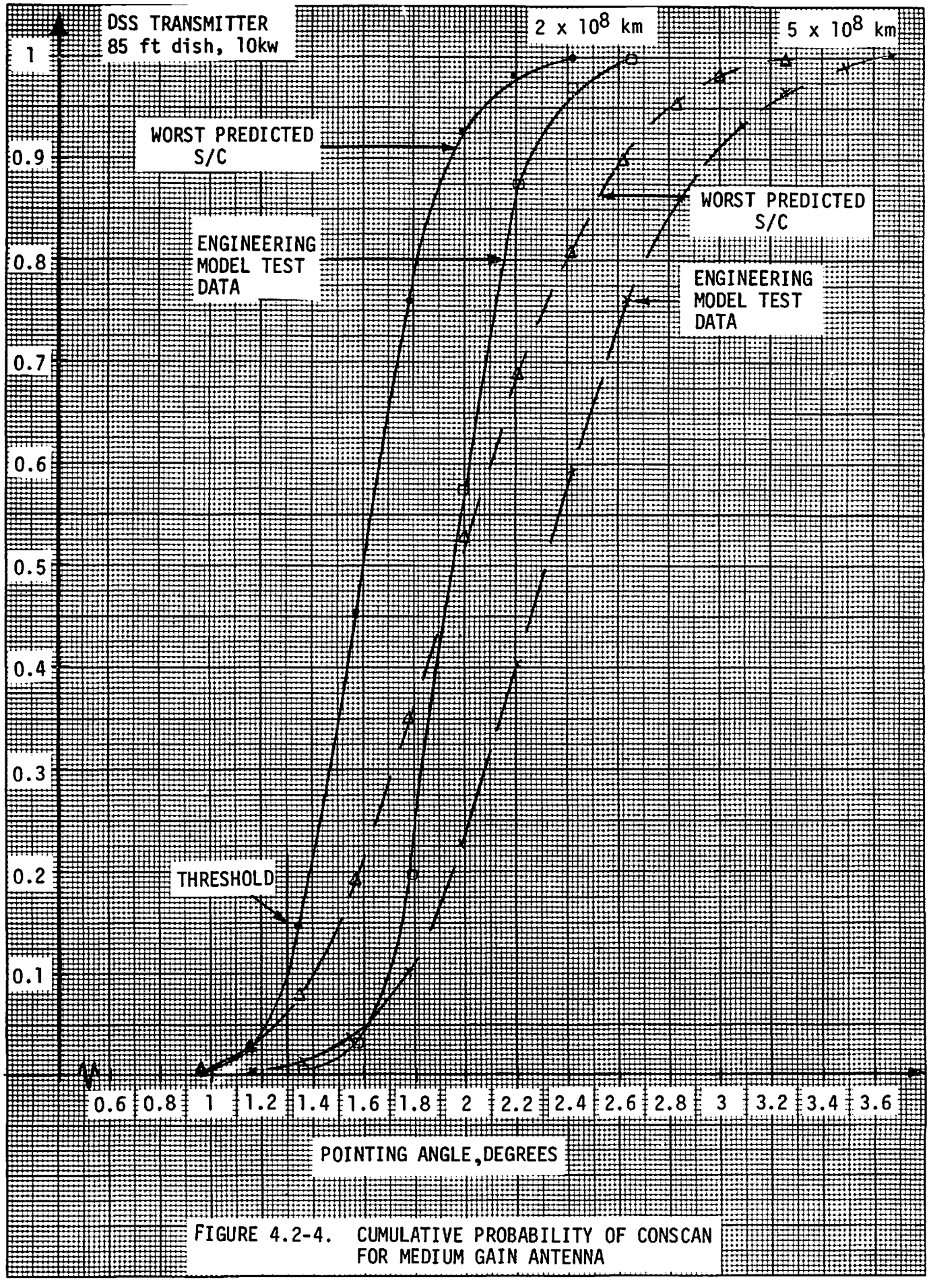


FIGURE 4.2-4. CUMULATIVE PROBABILITY OF CONSCAN FOR MEDIUM GAIN ANTENNA

Table 4.2-3. Telemetry Error Budget for the Medium Gain Antenna
(5×10^8 km Distance, Single Measurement)

ERROR SOURCE	POINTING ANGLE $c = 2^0$				POINTING ANGLE $c = 10^0$			
	PHASE ERRORS DEGREES		AMPLITUDE ERRORS PERCENT		PHASE ERRORS DEGREES		AMPLITUDE ERROR PERCENT	
	1 SIGMA	3 SIGMA OR PEAK	1 SIGMA	3 SIGMA OR PEAK	1 SIGMA	3 SIGMA OR PEAK	1 SIGMA	3 SIGMA OR PEAK
A. Bias								
1. Antenna Phase Variation	--	-6.6	--	--	--	+1.3	--	--
B. Random Before Launch								
1. Antenna Range	1.2	3.6	5.8	17.5	0.2	0.5	1	3
2. Mechanical Antenna Alignment	0.44	1.3	0.64	1.9	0.44	1.3	0.64	1.9
3. Receiver AGC Slope Measurement	--	--	2	6	--	--	2	6
Total Random Before Launch	1.3	3.8	6.2	18.6	0.5	1.4	2.3	7
C. Slow Varying (Random)								
1. Antenna Third Harmonic	0.2	0.3	0.35	0.5	0.2	0.25	0.3	0.4
2. Antenna Polarization	1.9	2.6	3.3	4.6	0.2	0.3	0.35	0.5
3. Spin Rate Variation	1.6	2.8	0.2	0.3	1.6	2.8	0.2	0.3
Total Slow Varying	2.5	3.8	3.3	4.64	1.62	2.83	0.5	0.71
D. Fast Varying (Random)								
1. Noise	18	54	34	102	2.8	8.4	5.5	16.5
2. Wobble	1.34	1.9	2.4	3.4	0.3	0.4	0.25	0.35
3. Truncation	4.6	8	8.3	14.3	1.2	2	1.3	2.2
Total Fast Varying	18.6	54.6	35	103	3.06	8.65	5.65	16.6
TOTAL RANDOM	18.8	55	35.7	105	3.5	9.2	6.1	18

Table 4.2-4. Telemetry Error Budget for the High Gain Antenna
(9×10^8 km Distance, Single Measurement)

ERROR SOURCE	POINTING ANGLE $\epsilon = 0.5^0$				POINTING ANGLE $\epsilon = 2^0$			
	PHASE ERRORS DEGREES		AMPLITUDE ERRORS PERCENT		PHASE ERRORS DEGREES		AMPLITUDE ERRORS PERCENT	
	1 SIGMA	3 SIGMA OR PEAK	1 SIGMA	3 SIGMA OR PEAK	1 SIGMA	3 SIGMA OR PEAK	1 SIGMA	3 SIGMA OR PEAK
A. Bias								
1. Antenna Phase Variation	--	2.4	--	--	--	-0.06	--	--
B. Random Before Launch								
1. Antenna Range	1	3	1.1	3.4	0.2	0.5	0.6	1.8
2. Mechanical Antenna Alignment	3.5	10.6	6.8	20	3.5	10.6	7.3	22
3. Receiver AGC Slope Measurement	--	--	1	3	--	--	2	6
Total Random Before Launch	3.6	11	6.95	20.5	3.5	10.6	7.6	23
C. Slow Varying (Random)								
1. Antenna Third Harmonic	0.35	0.5	0.6	0.8	0.3	0.4	0.5	0.7
2. Antenna Polarization	0.6	0.86	1.1	1.5	1.74	2.5	3	4.3
3. Spin Rate Variation	1.6	2.8	0.2	0.3	1.6	2.8	0.2	0.3
Total Slow Varying	1.7	3	1.3	1.7	2.4	3.8	3.1	4.4
D. Fast Varying (Random)								
1. Noise	0.9	2.7	1.8	5.4	0.45	1.35	0.9	2.6
2. Wobble	0.85	1.2	1.5	2.1	0.2	0.3	0.4	0.6
3. Truncation	1.5	2.5	2.5	4.4	0.3	0.5	0.5	0.9
Total Fast Varying	2	3.9	3.4	7.3	0.58	1.5	1.1	2.8
TOTAL RANDOM	4.5	12	7.9	22	4.3	11.3	8.3	23.6

4.2.4 Operation

A possible conscan operation sequence is presented in Table 4.2-5. Clearly, in some cases the sequence of commands is arbitrary, but in other cases the sequence of events should not be interchanged. In the proposed operation sequence it was assumed that at the start the pointing error angle is larger than 2 degrees and therefore the medium gain antenna is used. It is important to point out that the precession thruster pair 1 must always be used with the CSP 180 degrees phase mode of operation (thruster pair 2 with the 0 degrees CSP phase mode) or the pointing error angle will be increased instead of reduced. To avoid false firing pulses, it is recommended that after turning CSP power on, about 3 minutes be allowed to elapse before closed loop operation is commanded.

Normally, conscan will be automatically terminated as soon as the spin axis reaches the deadzone which is 1.3 degrees for the medium gain antenna. However, to avoid wasting fuel in case of some anomaly, a stop conscan command should be stored in the stored command programmer. This command should be timed to stop conscan some time after the estimated conscan time of completion. The time of completion can be estimated from the telemetered initial pointing angle, the period between precession steps (3 revs) and the precession step size. The precession step size is a function of gas pressure and will vary between 0.32° and 0.21° for the 125 ms pulse. When operating near earth the propogation time is short and the storage of the "stop Conscan" command in the CDU is not critical since a direct command can be transmitted as soon as an anomaly has been detected.

Some tests have indicated that turning the command modulation On or Off results in generating a false firing pulse at the output of CSP. This is caused by the abrupt change in the carrier power level of about 2.8 dB. Therefore, if the command modulation is turned off after the last command is sent, one firing pulse could occur at a wrong time.

During closed loop Conscan maneuvers, the pointing error magnitude and phase should be monitored to verify that the operation is normal. For this purpose telemetered parameters A sin θ and A cos θ together with the calibration curves of Section 3.6 can be used. In addition, if the AGC/Conscan telemetry is being processed on the ground, it can be used to check A sin θ and A cos θ accuracy. Another possibility is to monitor the amplitude modulation on the downlink signal which is approximately proportional to the pointing error angle.

After Conscan via the medium gain antenna has terminated, operation with the high gain antenna should not be initiated without first verifying that the pointing error angle has been reduced to less than 2 degrees. It is possible to conscan on the high gain antenna with pointing error somewhat higher than 2°. However, degraded performance or even instability can occur if the system operates on the antenna pattern sidelobes (see Figure 3.6-12). In addition, if the pointing error is greater than 2°, the Conscan processor begins to limit and incorrect pointing error information is obtained from the telemetered parameters A sin θ and A cos θ.

Table 4.2-5. Conscan Operations Sequence

EVENT	COMMAND	COMMENT
1. Establish uplink communication	-	It is assumed that the receiver connected to the medium gain antenna is selected.
2. Check telemetry for receiver in lock	-	Telemetry channel C-308 SIGA will indicate signal presence.
3. Check that spacecraft received carrier power level is within predicated range.	-	The predicted nominal carrier power is presented in Figure 3.6-7.
4. Turn on command modulation	-	
5. Verify that received carrier power dropped by about 2.8 db	-	Carrier power level drops by about 2.8 db due to modulation loss.
6. Command power on to DSL-A of ACS	SLA9	Start from ACS standby mode is assumed. Turns power on to DSL-A, turns power off to DSL-B
7. Clear DSL-A	ACR1	Reset clock
8. Command precession thruster pair 1 (positive precession)	PRE1	For CSP output phase of 180° (use PRE-2 with Conscan 0° phase).
9. Command medium gain antenna step size of 0.25°	PUL2	Selects 125 ms pulse width to thrusters.
10. Command CSP 180° phase	CNS4	For precession thruster pair 1.
11. Command star reference (or sun reference)	SEN3	Star reference selection to be followed by commands required for Canopus lock (see Section 4.3).

Table 4.2-5. Conscan Operations Sequence (Cont'd)

EVENT	COMMAND	COMMENT
12. Command SPSG ACS mode	RIP9	Can use any other mode of SPSG, TLM degraded.
13. Verify that SRA (or SSA) and SPSG are operating properly	-	
14. Turn CSP power on	CNS9	Wait about 3 minutes (6 revs) before starting closed loop operation.
15. Check CSP operation	-	Telemetered parameters A sin θ and A cos θ can be used to partly verify CSP operation (Section 3.6.2.8.4)
16. Set threshold for medium gain antenna operation	CNS1	
17. Store "stop Conscan" command in SC programmer	CDS1, CNS6, 750	Estimate the time required to complete Conscan and store a command to terminate Conscan some time after the estimated completion time (This is backup)
18. Command closed loop start	CNS5 CDX1	Wait about 3 minutes from event #13. Do not turn off command modulation.
19. During Conscan monitor pointing error angle magnitude and phase	-	Telemetered parameters A sin θ and A cos θ are used (Channels C-313, 314 and 315)
20. Command high gain antenna feed offset	MFA9	Turns power on to the feed movement mechanism A
21. Verify that the feed has been offset	-	Telemetry channel (C-316)-ANTX will indicate feed offsets

Table 4.2-5. Conscan Operations Sequence (Cont'd)

EVENT	COMMAND	COMMENT
22. Command receiver A to high gain antenna	ANT3	Switches receives A from medium gain antenna to high gain antenna.
23. Verify operation and performance	-	
24. Command high gain antenna step size	PUL1	0.03125 sec pulse width for HGA Conscan.
25. Set threshold for High Gain Antenna operation	CNS2	
26. Store "stop Conscan" command in programmer	CDS1,CNS6 1575	
27. Command closed loop start	CNS5 CDX1	
28. Establish that pointing error has been reduced to less than 0.5 degrees	-	Telemetered parameters A sin θ and A cos θ are used.
29. Command feed to on-axis position	MFA0	Turns power off to feed movement mechanism A.
30. Command CSP power off	CNS0	

During the initial portion of the mission, the high gain antenna feed will remain offset (see Figure 3.6-21) because of thermal conditions and the command to offset the feed may not be required. After about 90 days from launch, the feed normally will be commanded to the On axis position and the actuation time will be about 5 minutes. As shown in Figure 3.6-22, at the end of the mission the actuation time will increase to about 14 minutes.

Although the 0.29 degree precession step size can be used with the high gain antenna, it is recommended that the smaller step size of approximately 0.06 degrees be employed. With smaller step size, Conscan will be terminated closer to the 0.3 degree threshold setting and possible instability with very large phase error will be avoided.

4.3 STAR ACQUISITION

4.3.1 Special Tools

A number of quick reference charts and tools are presented in this section for determining special information about various stars. If stars other than Canopus are detected by the SRA, this information will establish the identity of those stars and aid in determination of the position of the ecliptic plane. These tools include a) plot of the angle from the xy spacecraft plane of several stars as a function of flight time (often called the "worm chart"), b) Stereographic Projection Map, c) Probability of detection table, and d) Roll angle star charts. Each of these will be discussed in detail in the following paragraphs.

4.3.1.1 Worm Chart. The worm chart gives information on which stars should normally be within the SRA field of view at any time in the mission. To generate the chart, it was assumed that the spacecraft is always pointing at the earth. This chart is for a nominal 4 March 1972 trajectory. Knowing the right ascension and declination of each star, and having trajectory data to determine the inertial attitude of the spin axis as a function of time, the angle between the spacecraft x-y plane and the star (pitch angle) can be calculated. This information is shown in the "worm" chart, Figure 4.3-1. The SRA field of view is $\pm 19°$ from the x-y plane. During the first 50 days of flight, only Canopus, Achernar and Vega will appear within the field of view. Mira, Beta-Grus and Arcturus will be within 10° of the FOV. Additional charts for other launch dates can be found in Pioneer Document PC 292.

4.3.1.2 Stereographic Projection Map. The stereographic projection map consists of two hemisphere maps and a clear plastic overlay. These hemisphere conformal maps indicate right ascension and declination, with the outer perimeter coinciding with the ecliptic plane. The center of each of these maps is the ecliptic pole. The plastic overlay, Figure 4.3-2, shows roll angles between stars and pitch angles from the spacecraft spin axis which is the straight line between the poles. The one remaining straight line on the overlay is the equator or x-y plane of the spacecraft which defines the center of the locus of the SRA field of view. The lines defining the region $\pm 19°$ from the equator define the boundary of the SRA field of view on any given revolution.

To use the stereographic projection map, the overlay is placed on one of the hemisphere maps (Figures 4.3-3 and 4.3-4) with the centers (ecliptic pole) of each coinciding. A pin is placed in this center such that the overlay can be rotated on the hemisphere. The overlay is rotated so that the spin axis points in any desired direction on the ecliptic plane. On day zero the nominal spin axis right ascention is about 78°. Alignment of the overlay in this direction shows Canopus and Achernar within the FOV with Regulus on the edge of the FOV. The roll angle between Canopus and Achernar can be estimated at 37°. This chart can also be used for small pointing deviations from the ecliptic plane ($<10°$), but accuracy is degraded.

4.3.1.3 Probability of Detection Table. The detectors in the Stellar Reference Assembly have been chosen to selectively detect the spectral characteristic of Canopus. Since the spectral content and intensity of other stars differ from that of Canopus, few stars will be detected with high probability. Table 4.3-1 indicates which stars are expected to be detected and aids in determining the method of star acquisition.

There are always stars other than Canopus with detection probability greater than 0.1 in the FOV. Achernar (probability 0.679) is the most troublesome because it occurs before Canopus by 40° or less. It eventually leaves the FOV, but remains within 9° of the FOV for the duration of the mission.

4.3.1.4 Roll Angle Star Charts. The roll angle star charts (Figures 4.3-5 and 4.3-6) give the expected roll angle between the stars and the sun for any given time in orbit. These charts can be generated by computer software given the trajectory data, star right ascension and declination, and the attitude of the spacecraft (usually earth pointing). This chart is produced using the same program used to produce the Worm Chart. In fact, the Roll Angle Charts are a vertical slice (polar plot) of the Worm Chart for a given day with roll information added. The sun position is shown (3° from earth in the sample) and the SRA field of view is defined by the third line (±19°) from the outside of the sphere. Blanking of stars by use of the angle and time gates can be determined with this diagram.

Also included on the map is the location of the sun and a line parallel to the ecliptic plane labeled "ecliptic." The radial lines indicate the roll angle between the star and the sun pulse or the $+y_b$, z_b plane. The circles indicate the angle between the star and the z_b axis. Each circle represents a 10° increment. The star sensor's FOV is approximately the outer 3 circles, i.e., 71° to 109°. The location of the ecliptic was determined by subtracting 90° from the location of the ecliptic pole.

4.3.1.5 Other Graphical Tools. Other quick reference charts should be produced for full understanding of the desired delay and general operation of the star logic. Perhaps the three most useful of these are: 1) angle between the spin axis and the sun vs flight time (sun look angle), 2) cone or pitch angle from spin axis to the ecliptic vs time of flight, and 3) roll angle from the sun to ecliptic plane vs time of flight. All of these are dependent on the orbital trajectory and would assume earth pointing of the spacecraft throughout the mission. An additional useful value is the angle from Canopus to the ecliptic plane. Since the ecliptic pole is shown on the Worm Chart, this information is readily found there. Figures 4.3-7 and 4.3-8 show these items for a nominal 4 March 1972 launch date. More detailed information for other launch dates can be found in Pioneer Document PC 292. Figure 4.3-9 shows the roll angle from the sun to the ecliptic plane indicating those times of flight when Canopus will be within sector seven. This must be true to use the angle gate for star acquisition. The pitch angle to the ecliptic verifies proper use of the stereographic projection conformal maps. The angle from Canopus to the ecliptic is of course the desired star delay for the ecliptic plane reference for the experiments.

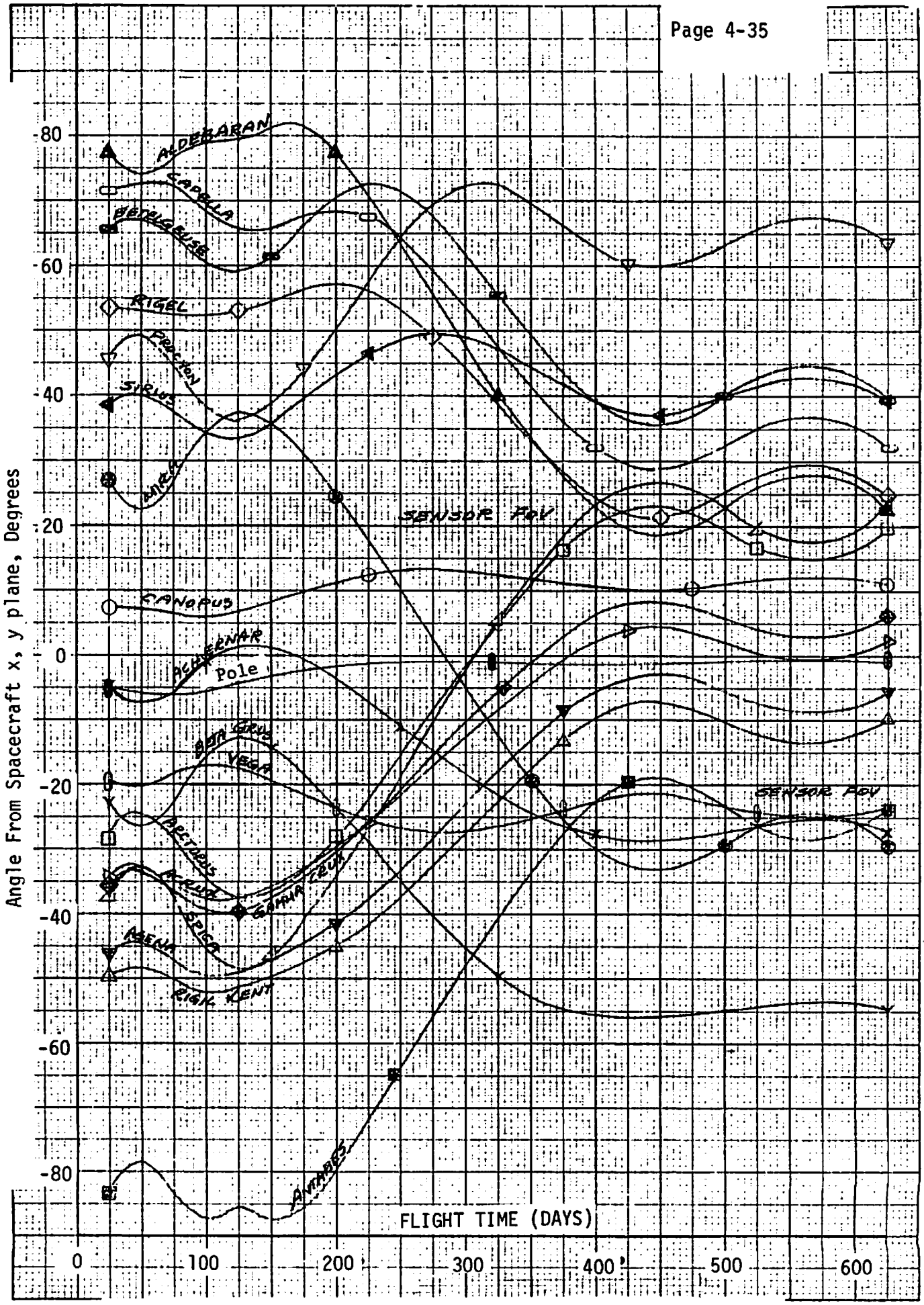


Figure 4.3-1. Angle Between Star Vector and S/C x, y Plane

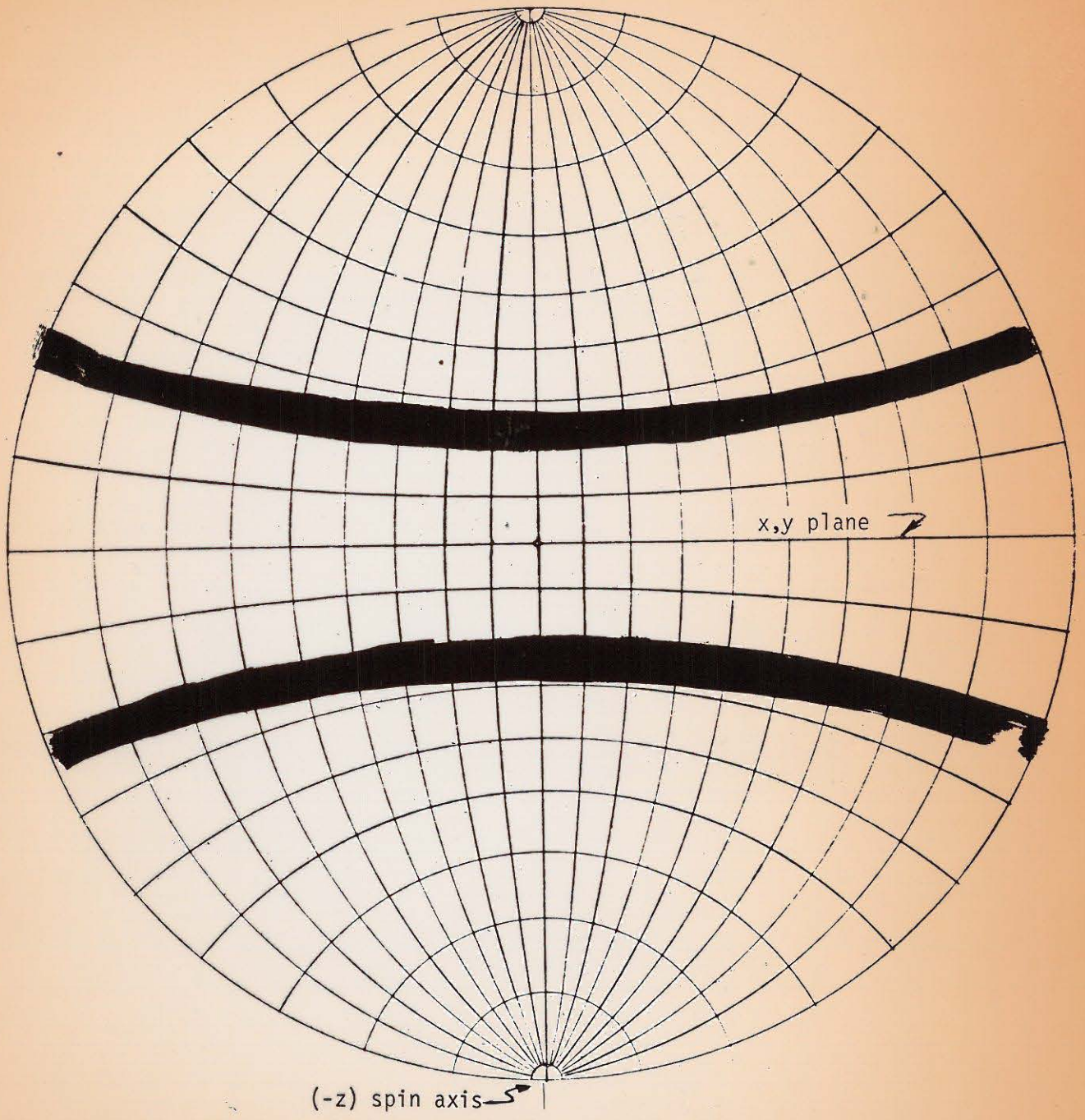


Figure 4.3-2

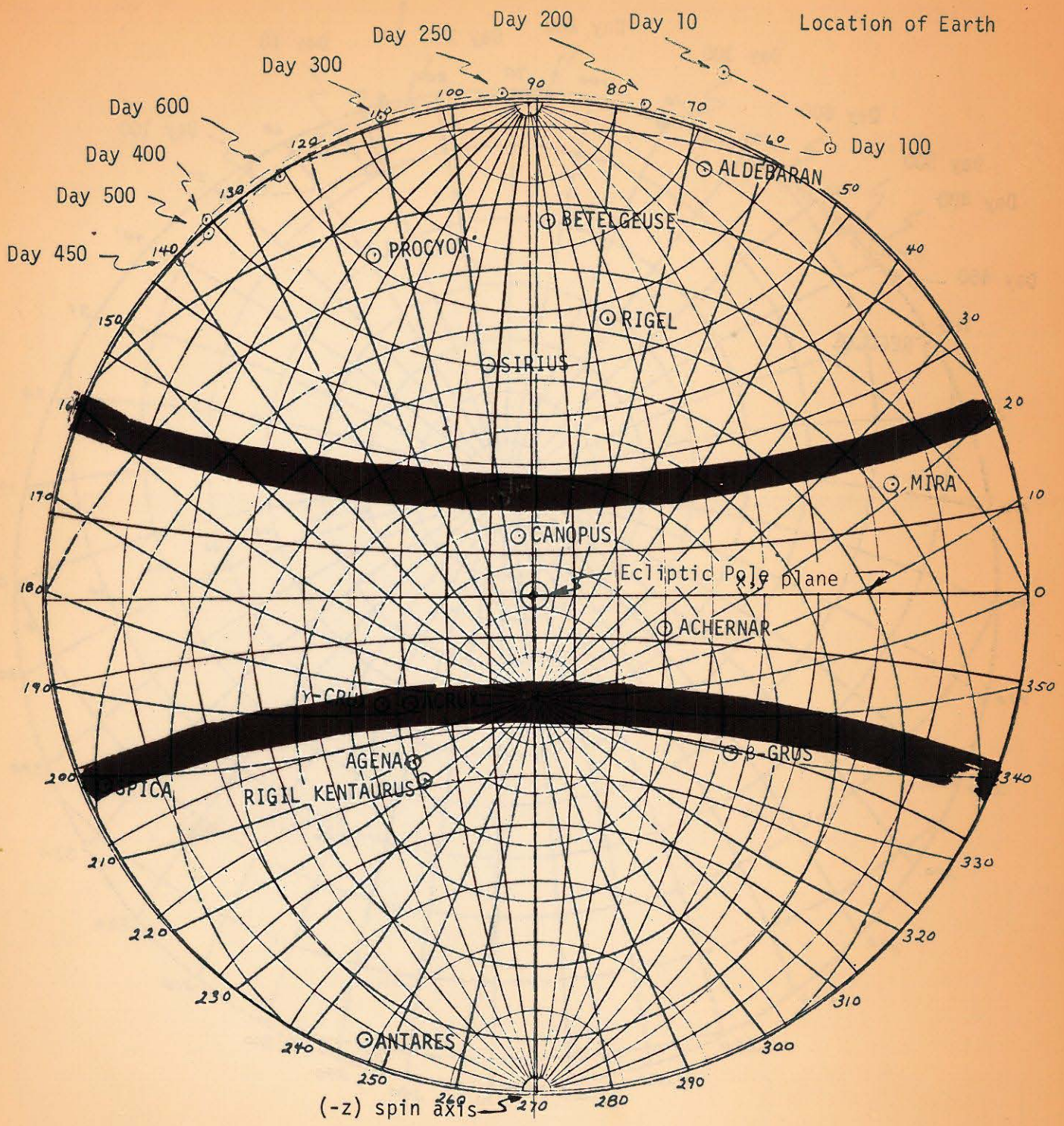


Figure 4.3-2

Figure 4.3-3. Stereographic Projections of Bright Stars in the South Ecliptic Hemisphere

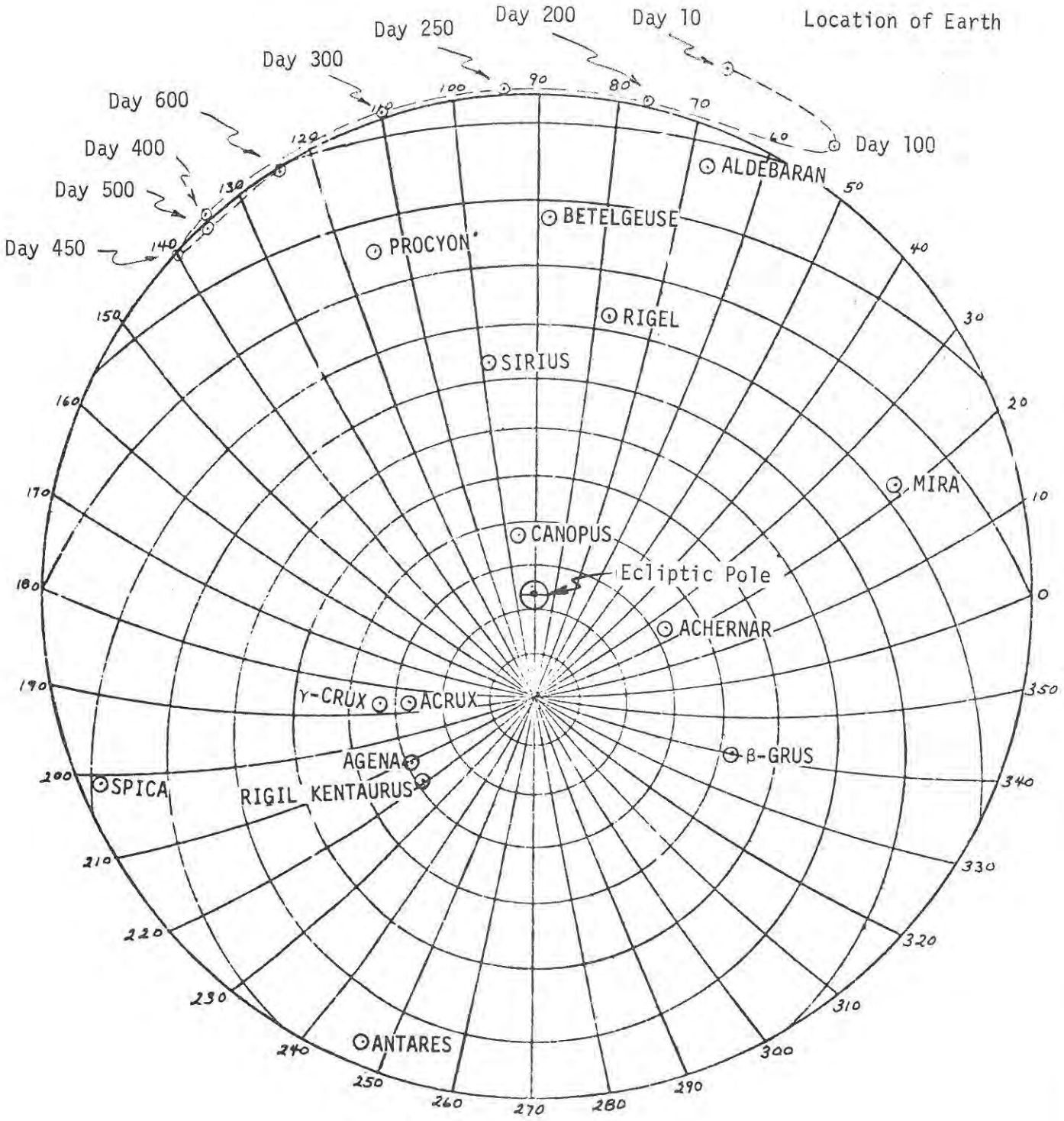


Figure 4.3-3. Stereographic Projections of Bright Stars in the South Ecliptic Hemisphere

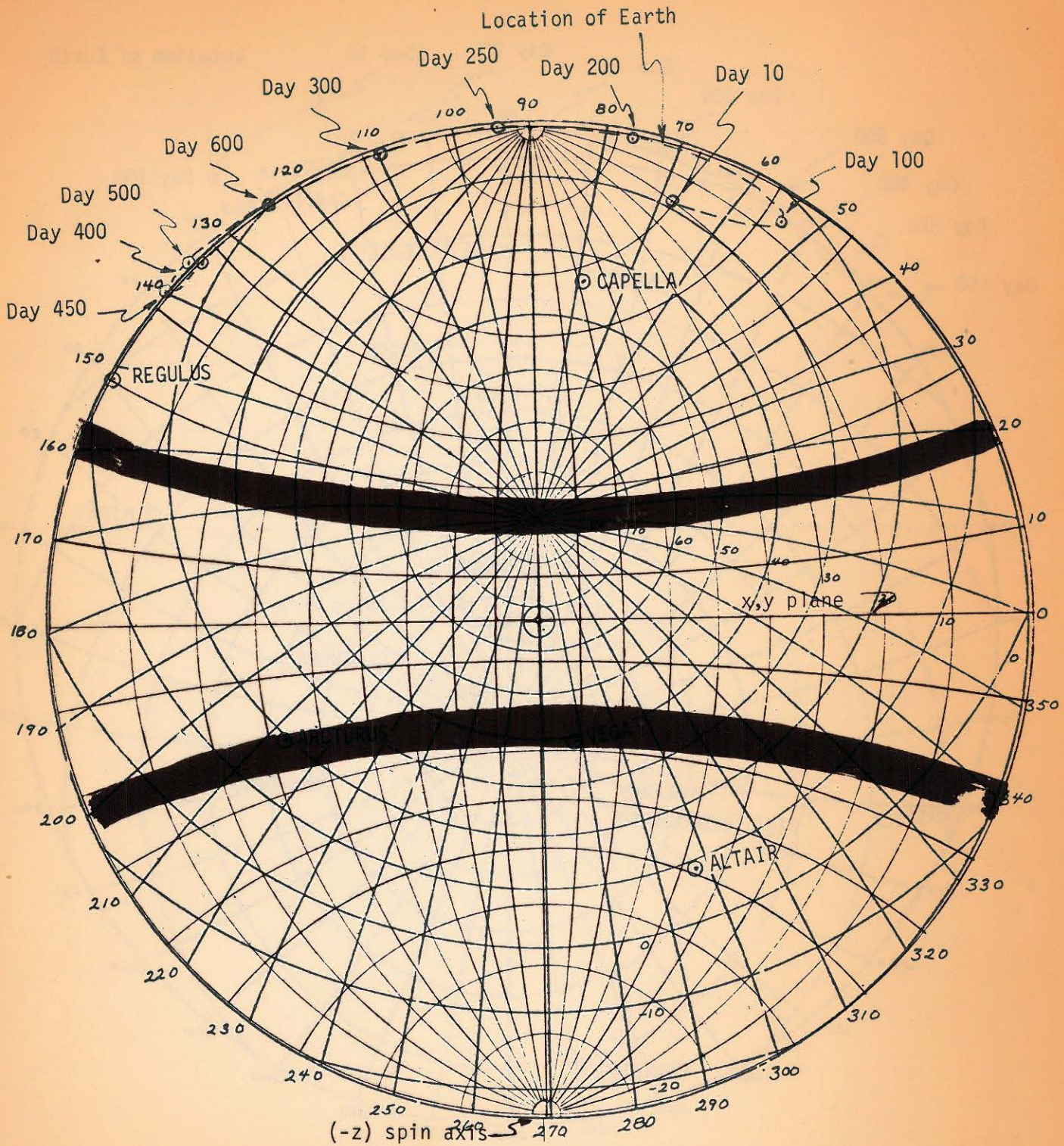


Figure 4.3-2

Figure 4.3-4. Stereographic Projections of Bright Stars in the North Ecliptic Hemisphere

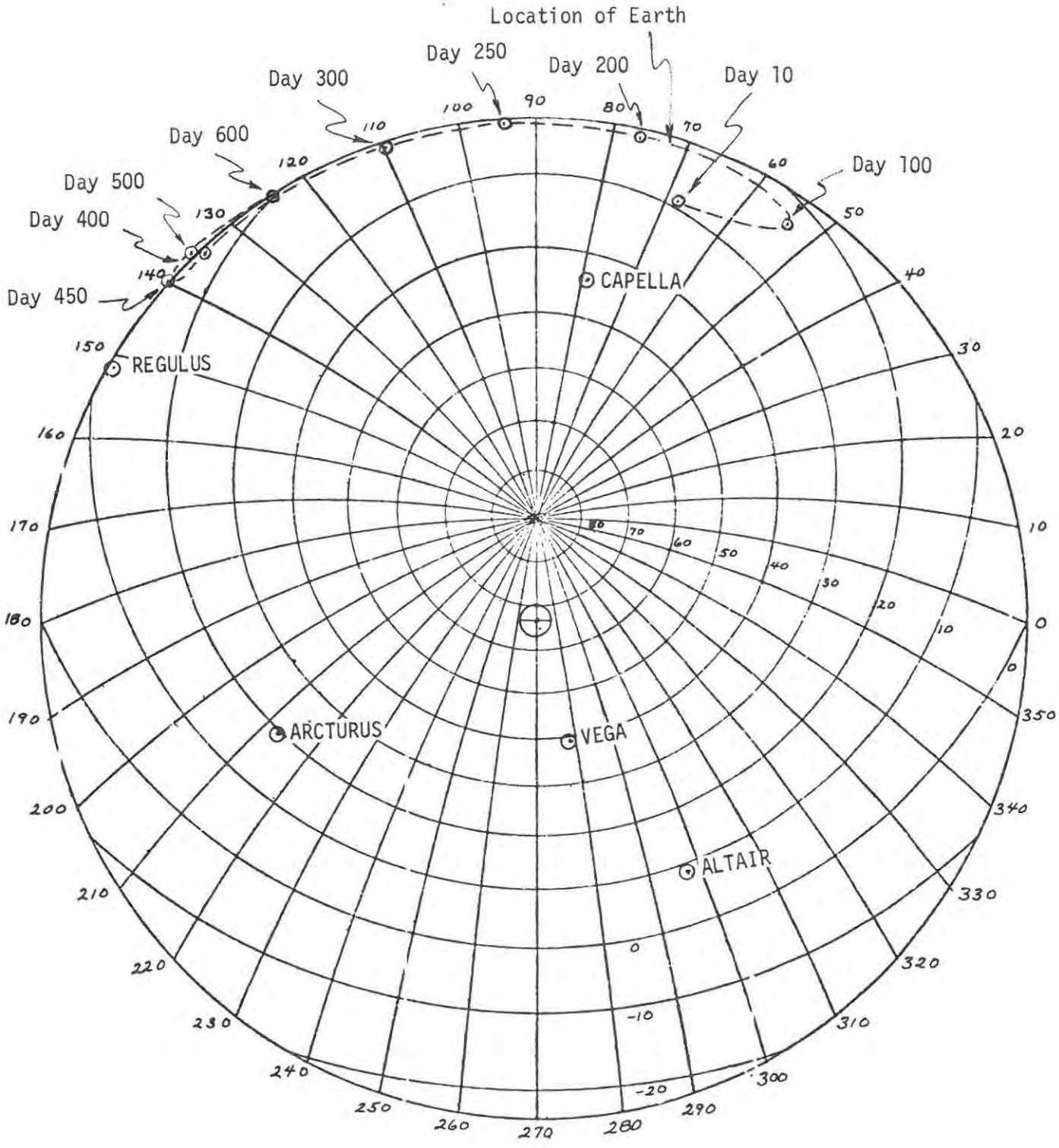


Figure 4.3-4. Stereographic Projections of Bright Stars in the North Ecliptic Hemisphere

Table 4.3-1. Star Location and Brightness

Star			RA	DEC	Probability of Detection	Revs. Between Detection**		
						.9	.95	.99
SIRIUS†	α	CMA	100.9125	- 16.6681	1	0	0	0
CANOPUS	α	CAR	+ 95.7992	- 52.6766	1	0	0	0
BETELGEUSE†	α	ORI	88.3325	+ 7.4028	.42, .07, 1 (variable, period 2000 days)	30	41	63
CAPELLA†	α	AUR	78.5433	+ 45.9656	0.79	1	2	3
AGENA	β	CEN	210.3517	- 60.2100	0.75	1.6	2.1	3.3
ACHERNAR	α	ERI	+ 24.1125	- 57.4092	0.678	2	2.6	4
RIGEL KENTAURUS	α	CEN	214.4240	- 60.7167	0.078(.55)*	2	3	5
VEGA	α	LYR	278.9467	38.7508	0.16	13	17	26
ALDEBARAN	α	TAU	68.4917	16.4425	0.13	16	21	33
ANTARES	α	SCO	246.8246	- 26.3583	0.115	18	24	37
RIGEL	β	ORI	78.2258	- 8.2397	0.101	22	27	43
PROCYON†	α	CMI	114.3808	5.3133	0.0086	253	342	528
γ-CRUX	γ	CRU	187.3032	- 59.9	0.0002	$1.2\ 10^{4}$	$1.5\ 10^{4}$	$2.3\ 10^{4}$
ARCTURUS	α	BOO	213.5275	19.3586	0.0001	$2.3\ 10^{4}$	$3.0\ 10^{4}$	$4.6\ 10^{4}$
MIRA	Θ	CET	+ 20.660	- 8.1613	$0.28\ 10^{-6}$	$8.2\ 10^{6}$	$1.1\ 10^{7}$	$1.6\ 10^{7}$
SPICA	α	VIR	200.8496	- 10.9844	$0.16\ 10^{-8}$	$1.4\ 10^{9}$	$1.9\ 10^{9}$	$2.9\ 10^{9}$
β-GRUS	β	GRUS	340.2490	- 47.0280	$0.16\ 10^{-8}$	$1.4\ 10^{9}$	$1.9\ 10^{9}$	$2.9\ 10^{9}$
ACRUX	α	CRU	186.1729	- 62.9108	$0.8\ 10^{-10}$	$2.3\ 10^{10}$	$3.7\ 10^{10}$	$5.8\ 10^{10}$

** .9, .95, .99 → probability of detection

* measured

† never in FOV during earth pointing of Pioneer F

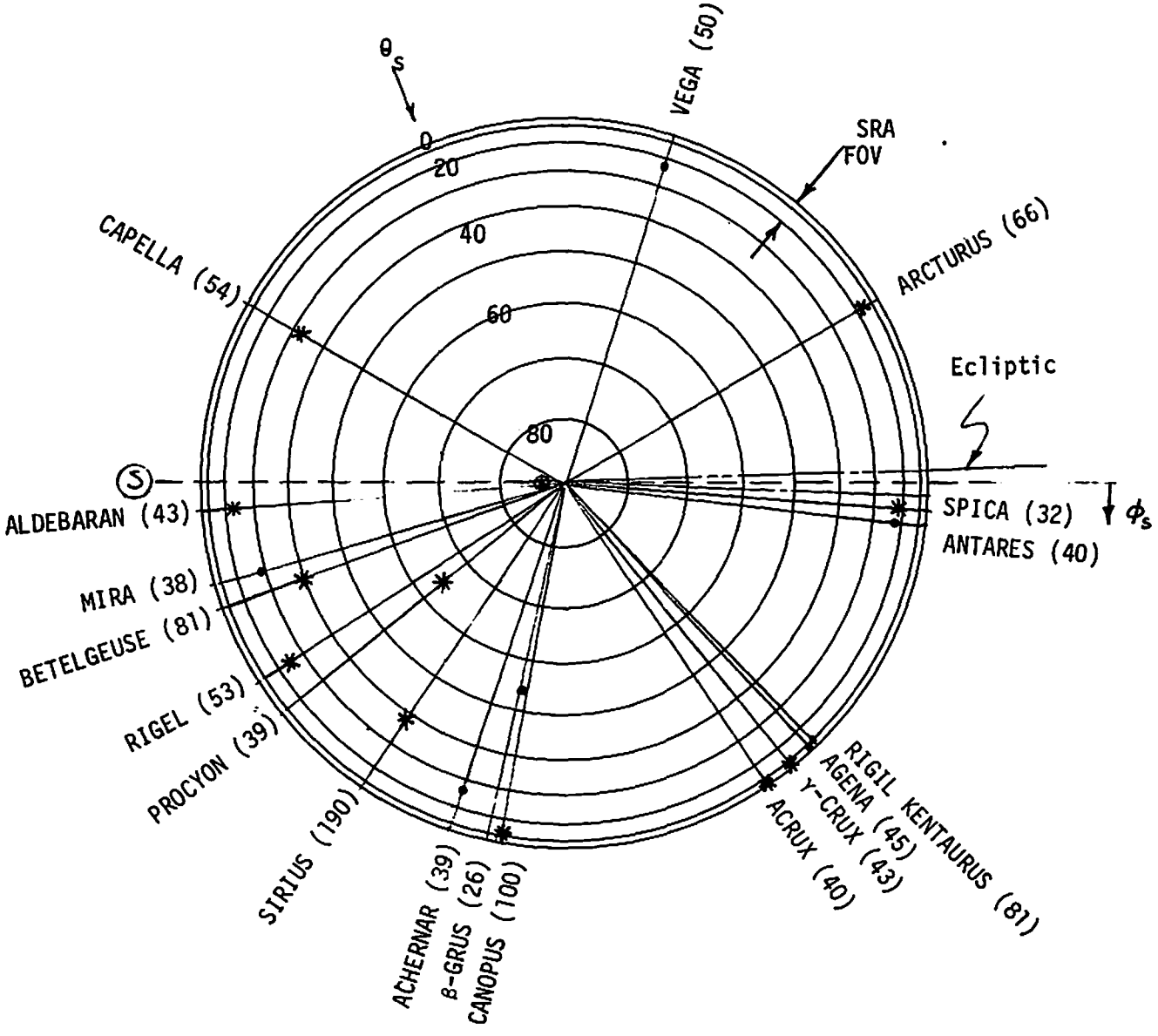


Figure 4.3-5. Roll Angle Chart, Star Location Relative to Spacecraft, Day 500

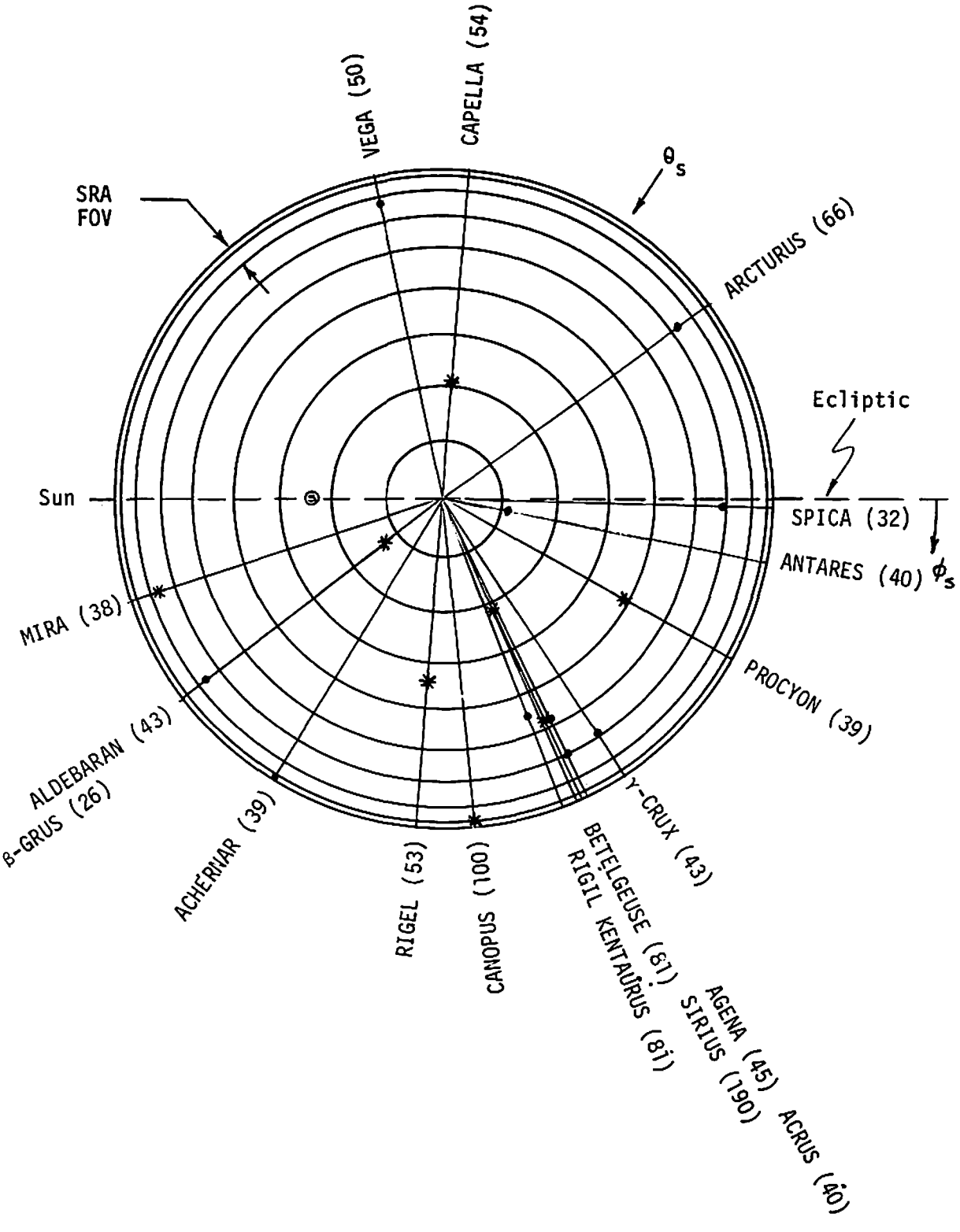


Figure 4.3-6. Roll Angle Chart, Star Location Relative to Spacecraft, Day 200

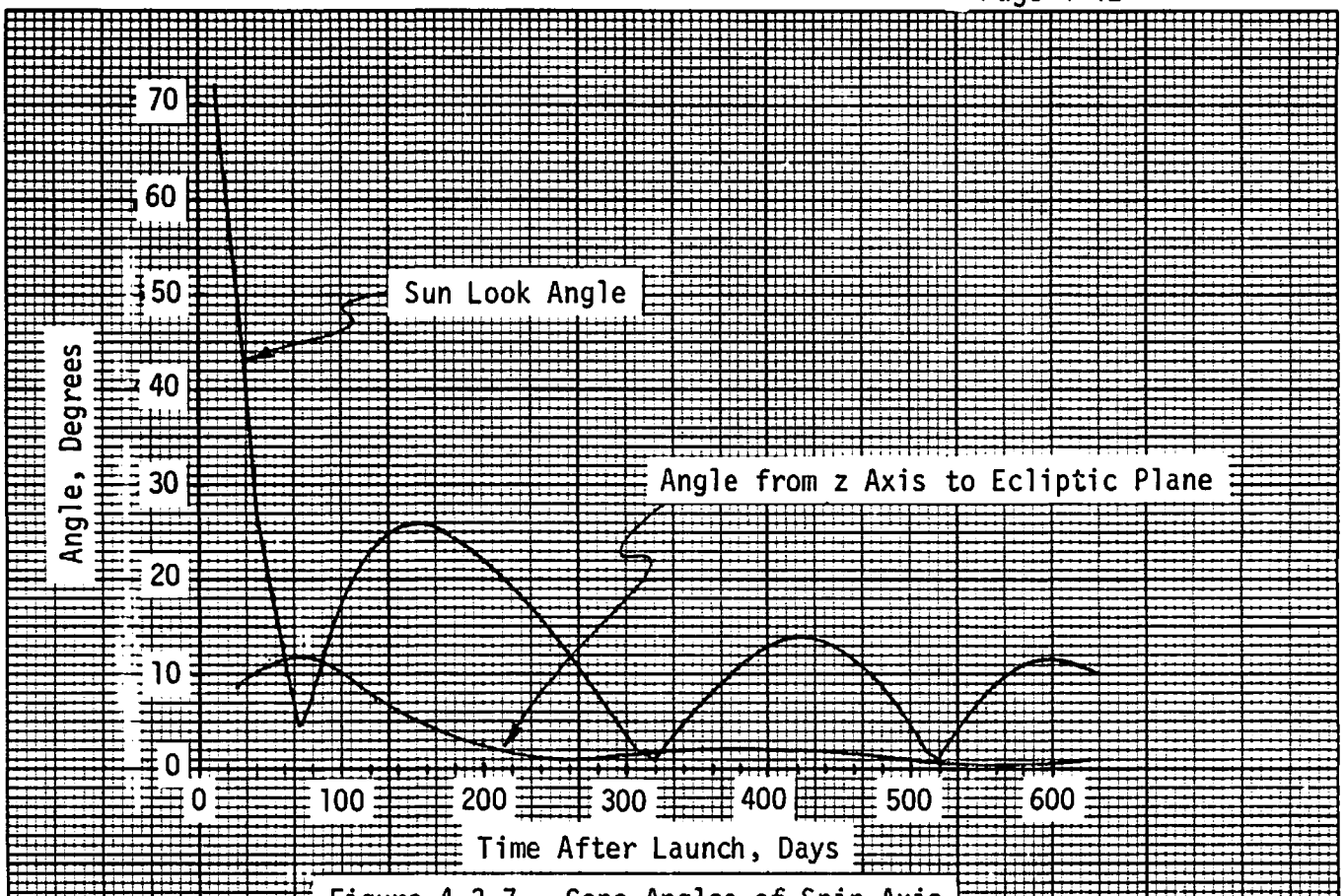


Figure 4.3-7. Cone Angles of Spin Axis

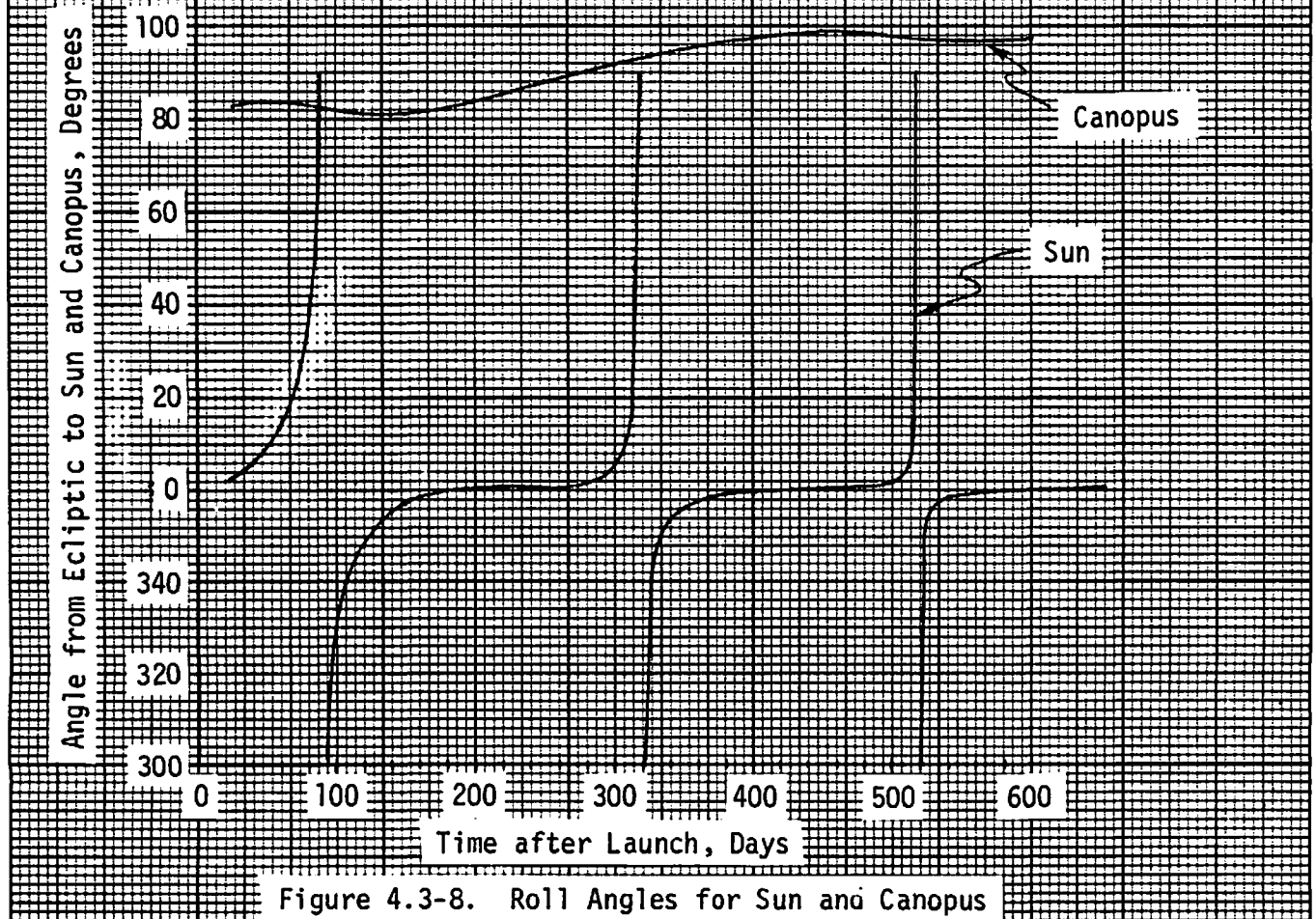


Figure 4.3-8. Roll Angles for Sun and Canopus

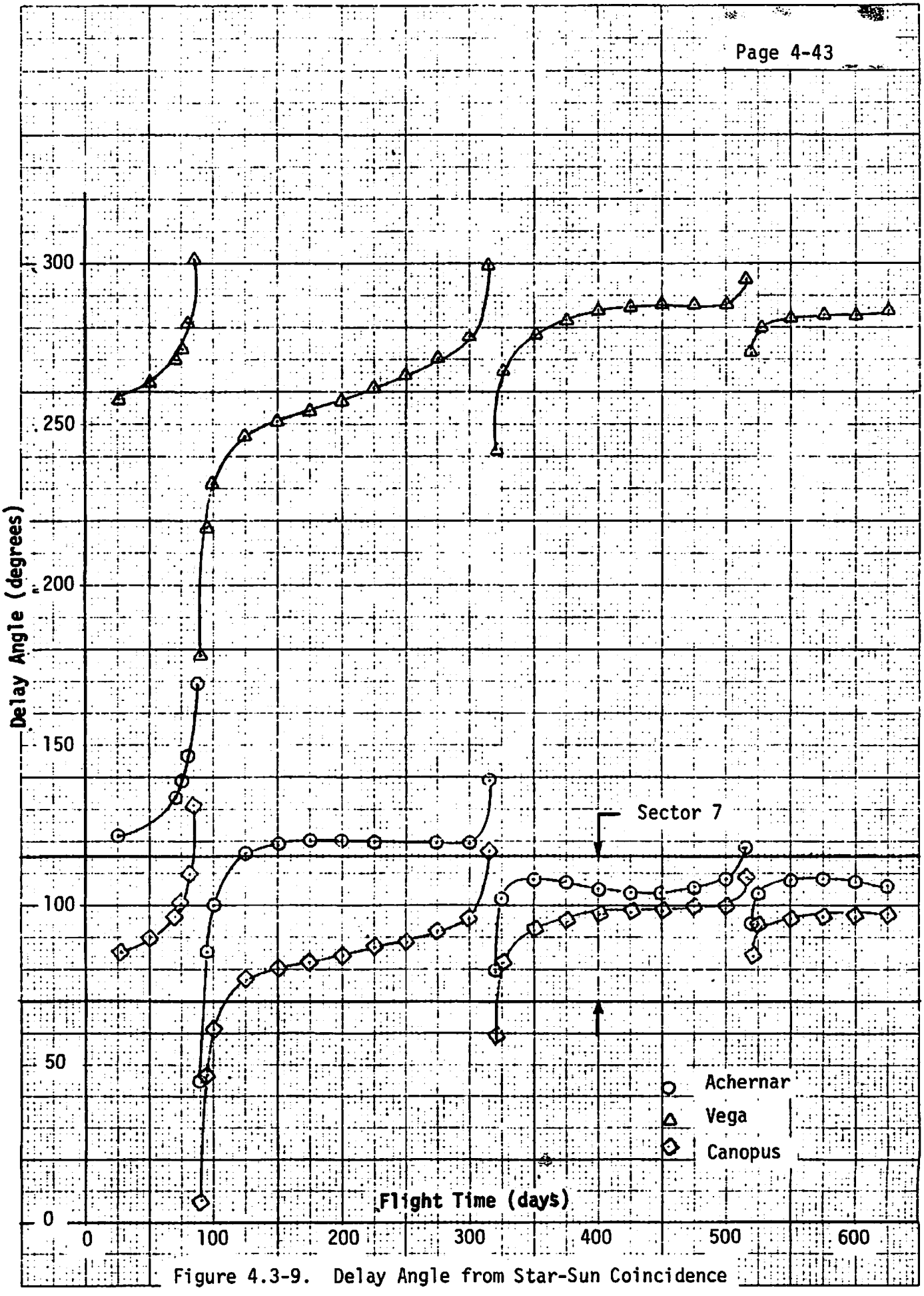


Figure 4.3-9. Delay Angle from Star-Sun Coincidence

4.3.1.6 Real Time Star Telemetry. The star count logic can be used to observe the star pulses from the SRA as they occur in real time during a revolution of the spacecraft. During each revolution, the star counter counts each star by incrementing a binary counter. This counter can be interrogated at a high rate so that the times that the counter is incremented can be observed. Using this feature and knowing the spin rate, these times can be converted to angles and all stars can be identified by comparing the angles between stars and between the stars and the sun.

The real time telemetry for a complete scan of the SRA output is obtained by disabling the star angle gate and the star time gate (Command SAS2) such that the star count logic receives the entire 360° star scan output of SRA. Since the star counter counts stars at the output of the time or angle gates, only those stars in the gate will be counted if a gate is enabled. The sun sensor is preferred as the roll reference to the SPSG when using the star counter since the 315° roll pulse resets the star count logic each revolution. If the SRA is used as a reference, more than one pulse per revolution could reach the SPSG since the gates would be disabled. Only the star delay would block pulses to the SPSG. The star counter will recycle, and can count more than 7 stars if the reset is observed. If the count goes from 7 to zero at some point other than 315°, a star has been counted.

The Roll Angle Star Charts are the tool for comparison for identification of stars. The angles measured can easily be determined to an accuracy dependent on the bit rate. The following table indicates the time quantization associated with various bit rates and telemetry formats.

Telemetry Bit Rate	C-4	Format C	A or B	D
16	12 sec	48 sec	25.6 min	51.2 min
32	6	24	12.8	25.6
64	3	12	6.4	12.8
128	1.5	6	3.2	6.4
256	0.75	3	1.6	3.2
512	0.375	1.5	0.8	1.6
1024	0.1875	0.75	0.4	0.8
2048	0.09375	0.375	0.2	0.4

It can be seen that in Formats A,B or D, little useful information can be obtained from the star counter with its nominal reset period of 12.5 sec (4.8 rpm spin rate). In the C-4 format at the 2048 bit rate, the quantization is 0.09375 seconds or 2.7 degrees. Since to a high degree of probability the telemetry bit rate and the spin rate will asynchronous, observation of the real time star telemetry for a number of revolutions will permit considerable improvement in measurement of the angles between stars. At times during the mission, high telemetry rate may not be possible to directly obtain real time telemetry of the star counter. The communication distance for example, may require a low bit rate, say 16 bps. In this event, the Data Storage Unit may be used to store several revolutions of data at the high (2048) bit rate. This data could then be sent to ground at the 16 bps required by communications.

4.3.1.7 Star Angle Gate. The star angle gate is not in itself a tool, but an understanding of the logic behind its design is a tool; and is presented here as such. In the following figure, the plane of the page is the x-y plane of the spacecraft. The sun pulse will occur at some point during the rotation as shown. This pulse indicates the start of octant generation.

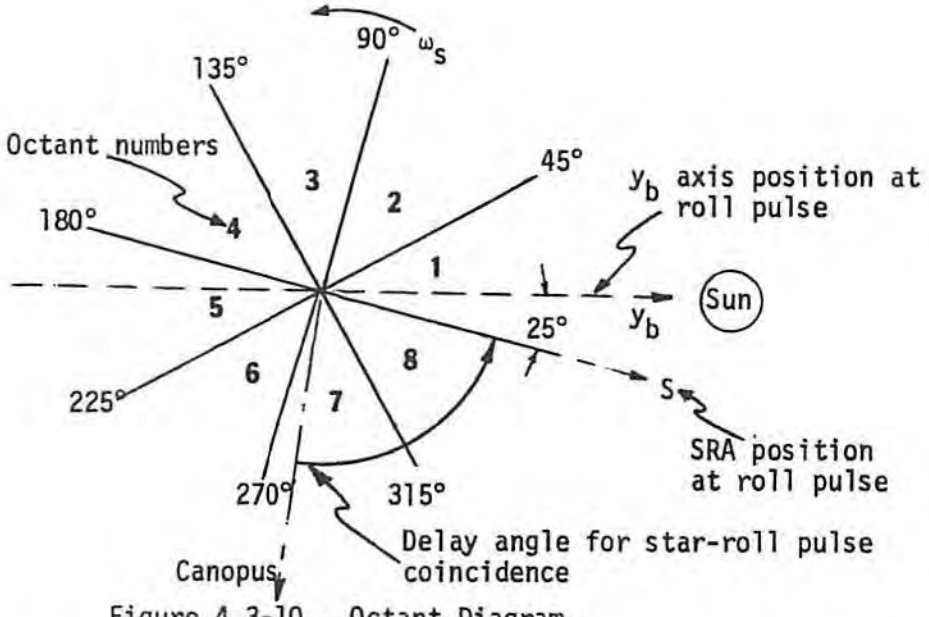


Figure 4.3-10. Octant Diagram

The octant dividing lines are shown as the position of the star sensor at the instant that the given octant begins. The roll pulse will correspond with the sun pulse when the 0° roll reference selected in the DTU (Command RIP1) and the sun is on the left side of the spacecraft as it looks at earth (position shown for the sun is defined as sun left). Canopus will normally be detected in octant 7 since it is near the south ecliptic pole. It will not occur in octant 7 during some phases of the mission since the sun pulse will not always be near the ecliptic (e.g., during syzygy). The relative time of pulse occurrence is shown in the following diagram.

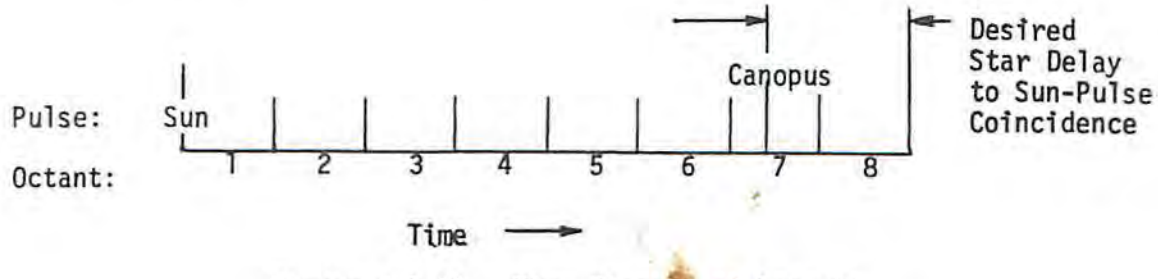


Figure 4.3-11. Time Diagram of Octants

Octant 7 is the star angle gate and octants 1 and 8 are used in the star location logic. The star is delayed until the delayed pulse just changes from octant 8 to octant 1. It then coincides with the sun pulse. The angle for the delay is that between Canopus and the S axis at the instant the roll pulse occurs as shown in the octant diagram (Figure 4.3-11). It is possible, that the delayed star pulse could be indicated in both octants 1 and 8 simultaneously since it has a finite width.

The star count logic is reset on the 315° point just at the end of octant seven. If the star counter is used with the angle gated stars, the star count may not be held long enough for telemetry interrogration. Star count may be incorrect in this case. At the highest telemetry interrogation rate (0.09375 sec period) stars within 2.7° of the 315° counter reset mark will not be see unless the counter is observed for several revolutions (using the asynchronous character of spin and telemetry interrogation). The discussion of this behavior of the star count logic is somewhat academic. If the angle gated star is used as the roll reference, then if any number of stars other than one (zero, two or more) occur in the gate, then the roll reference (SPSG) lock on the star may be lost and reacquisition may be required.

4.3.2 Acquisition Procedures

In the following paragraphs, it is assumed that software will provide the information necessary to properly perform the acquisition at whatever level of accuracy required. Such data includes the delays for the star pulse to obtain star coincidence, ecliptic plane indication and the roll and pitch angles of the stars relative to the spacecraft axes. The software to provide this information must be supplied data on the S/C attitude and location; the location of stars, planets, and the ecliptic pole; and orbital data. The software would provide the necessary transformations and computations to show roll and pitch angles and indicate objects that are within the FOV of the SRA.

Further, it is assumed that the reader is familar with the operation of each of the logic elements which comprise the star logic (Section 3.3.1.5) and the various tools available (Section 4.3.1).

4.3.2.1 Star Acquisition with Sun Reference. Prior to acquisition, it is desirable to identify stars in the FOV. With the sun available as a reference, the star count logic can be used for real time star telemetry to determine not only the number of stars but also the angles between them (see Section 3.4.1.6). This information is not required but greatly simplifies the acquisition and improves confidence in achievement of acquisition.

The most desirable condition for acquisition of Canopus occurs when the sun sensor is producing pulses once each revolution and the spin period sector generator is operating. If there are no stars other than Canopus, the ground operator need merely set the star delay required for that day and check the star location using the location logic. Either the time or angle gate should be set to provide protection from stray pulses or stars that appear occasionally (see Table 4.3-1).

The delay angle is telemetered and four ground commands are used for setting the delay register (SAS 3, SAS 4, SAS 5, and SAS 6). In order to obtain sun-star coincidence, the delay is incremented or decremented until the delayed star pulse just moves from octant 1 to octant 8 or vice versa. The "star sector" telemetry indicates whether the star pulse is in coincidence with the 0° roll pulse, not necessarily the sun pulse. During those days when the sun appears to the right of earth (south ecliptic pole is "down") for an observer on the spacecraft, the roll reference pulse is selected as 180° from the true sun pulse. If the star pulse is delayed to the roll reference pulse and the SPSG reference is switched to the star pulse (Command SEN 3) the SPSG will put out an erroneous pulse rate for one revolution. Figure 4.3-12 shows the problem more clearly.

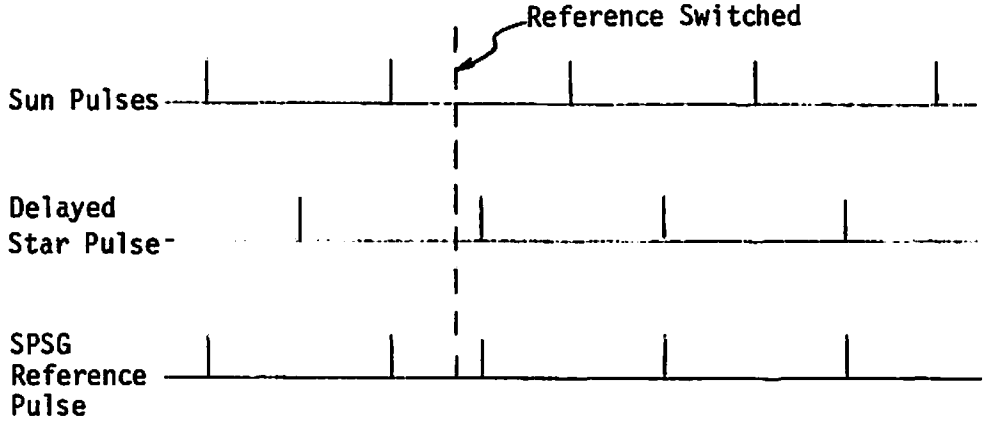


Figure 4.3-12. SPSG Input Switching

A change in the SPSG period in telemetry words C-405, C-406, and C-407 would be observed if this occurs. These telemetry bits indicate the spin period in seconds, with a quantization of about 0.00012207 seconds, and the most significant bit equal to 16 seconds. If octant 7 from the SPSG gates the star pulses during this switch from sun to star pulses, the star pulses will be lost. The solution to the problem is to add an additional 180° to the star delay so that the sun and star pulses coincide. To add this 180°, the spin speed must be accurately known. If there are no other stars in the gated FOV the operator may now switch the SPSG input to star pulses. If there are 2 or more pulses the first will be delayed by the delay logic and blank the second if it occurs within the delay time. Care must be taken in the use of the star coincidence sectors 4 and 5. Switching of the SPSG reference (0° or 180°, commands RIP 1 and RIP 2) can also cause the delayed star to occur between sectors 4 and 5.

4.3.2.2 Acquisition With Two or More Stars. The operator has two options for blanking out unwanted stars: The star time gate and the star angle gate. The angle gate allows the star pulses to pass during octant 7 and has the advantage of being independent of spin speed. Since the star sensor location on the spacecraft is 25° in roll behind the sun sensor, the actual angle within which star pulses will be passed is 70° (45° + 25°) to 115° (90° + 25°) before the sun pulse.

Octant 7 is indicated on Figure 4.3-9 showing that Achernar (troublesome because of 0.68 probability of detection) is within octant 7 from day 300 to Jupiter encounter, however it is not in the field of view after day 315. During the period between days 300 to 315 when Achernar is both in the field of view and in sector 7, the time gate should be used. If accurate earth pointing is maintained (spin axis not more than 5° from ecliptic plane) and the time gate opens within 15° of Canopus, then Achernar will be successfully blanked for the entire mission. A 12.25 second time gate blanking will give a maximum window opening of 14.4° at 4.8 rpm (see Section 3.3.1.5.5). The star time gate cannot be set closer to Canopus than .131 second in time (.125 sec hold + quantization & jitter) to maintain Canopus lock.

If Canopus is acquired and the SPSG is switched to star reference, the presence of Achernar may cause erroneous operation of the SPSG. Earlier than day 300 Achernar is only 3° to 5° out of octant 7. After day 315, Achernar is no more than about 9° outside of the FOV and is therefore a potential problem during the entire flight if pointing is not maintained within 5° of the ecliptic plane.

The time gate has a major advantage over the angle gate in that it takes full advantage of the probabilities of detection of various stars. Assume for example, the time gate is locked onto a given star. Should that star not be detected on a revolution, the gate will automatically lock onto the next star. Lock will be maintained only on stars of very high probability of detection. Since there are only 3 of these and since only Canopus is within the FOV of Pioneer F, the operator need only set the time gate and wait for Canopus acquisition to occur. This will be within 43 revolutions with probability 0.99, if the probabilities in Table 4.3-1 are correct. On day 450 the stereographic projection map shows an angle of about 5° in roll between Achernar and Canopus. The time gate cannot be used to reject Achernar's pulse since the time gate must open 10.9° before Canopus. In this instance it would be necessary to rely on accuracy of spacecraft pointing to keep Achernar out of the FOV.

Star maps show that all stars except Achernar that appear in the field of view between the y_b axis and Canopus would be rejected by the angle gate. These include Arcturus, Vega, Mira, Aldebaran and Rigel.

Figure 4.3-2 shows that from day 250 to 350 a group of stars come into the FOV including Acrux, Rigel Kentaurus, Agena and γ-Crux. When sun pulses are available as an aid to acquisition of Canopus these stars should present no problem. After the delay angle for Canopus has been set, the delay counter will still be counting at the time these stars appear in the FOV. The delay register, in effect, rejects any stars that occur during the time it counts down since new pulses do not reset and restart the count. A similar phenomenon occurs when Achernar appears in the gated field of view. It will produce a pulse before Canopus and therefore will start the star delay counter and blank out the Canopus pulse. The time gate and accurate pointing must be used to exclude Achernar from the FOV. For Pioneer G it is possible that Sirius can appear in the FOV before Canopus. However, it never appears closer to Canopus than 33° and can therefore be rejected by the time gate.

4.3.2.3 Acquisition Without Sun Reference. If sun reference is lost, then most likely the SPSG period will be incorrect. In this event the octant generator, the star location logic, the star angle gate, and the star count logic will not be useful until the SPSG period has been restored. The fastest and easiest Canopus acquisition procedure is to set the time gate at a value of least 1/8 sec less than the spin period, and wait 9 minutes for acquisition to occur. The existance of Canopus lock is then checked by star coincidence of the delayed star with other stars (or planets). The star advance command (SAT 1) is then used if lock has been achieved on some object other than Canopus. Telemetry should be checked for star level and spin period. Changes in SPSG period that occur after the command to set the time gate can be correlated with angles between stars detected and in the probabilities of occurrence if the ACS mode is used for the SPSG (Command RIP 5).

When operating on the time gate, a lost star pulse will cause lock to shift to the next star. This will be indicated by a shift in period in the SPSG telemetry for the revolution in which the loss occurs. The shift in reference can be checked by setting the star delay and watching for star coincidence with another known object in the SRA field of view. Stars of prime usefulness for this are:

Early in Mission		Late in Mission	
Star	Probability of Detection	Star	Probability of Detection
Canopus	1	Canopus	1
Achernar	0.68	Agena	0.75
Vega	0.16	Rigel Kentaurus	0.55

4.3.2.4 Summary of Star Acquisition Procedure. The procedures presented here are based on the preceding discussions. No attempt has been made to synthesize a rigorous procedure which takes care of all contingencies. The procedure is proffered as a guide to acquisition.

The following table contains procedures for various known conditions.

Star Acquisition Procedure

	Condition	Procedure
1.	SSA producing pulses One star in 360° FOV Sun left* geometry	Set star delay for Canopus-sun coincidence Set time gate to block spurious pulses and noise. Switch SPSG to star pulses

* Sun right, sun left defined in Section 4.3.1.7.

	Condition	Procedure
2.	SSA producing pulses One star in 360° FOV Sun right* geometry	Star should be Canopus Determine spin speed Set star delay for star-roll reference coincidence Add 180° for star-sun coincidence Set time gate to block spurious pulses and noise
3a.	SSA producing pulses Multiple stars in 360° FOV	Send SAS1 (Sector 7) command Determine from TLM how many stars in gated FOV
3b.	One star in sector 7	Set delay for star-roll reference coincidence Star should be Canopus If left switch SPSG to star If sun right add 180° and then switch
3c.	Multiple stars in sector 7	Set time gate for less than one revolution period Send time gate enable command and wait 9 minutes Set delay for star-roll reference coincidence. If delay corresponds to Canopus, switch SPSG to star; if not, check map to determine star's identity, send advance command, try again.
4.	No sun pulses	Set time gate to less than one spin period Send time gate enable command Wait about 9 minutes Increment star delay until TLM indicates coincidence with another star Check star map against delay to determine identity of star If delay is 360° assume star is Canopus If star locked onto is not Canopus send advance command, wait 9 minutes and check delays again If same star is locked onto as before and after sending advance command, set time gate to more than one spin period and then back again. Then try again for coincidence until Canopus is acquired.

4.3.2.5 Star Logic Description Summary. The logic elements used for star acquisition and located in the sensor and power control subassembly are presented for reference. A complete discussion of all logic elements is given in Section 3.3.1.5.

1) On receipt of command SAS1 the star angle gating enables the star pulses to be passed only during sector 7, i.e., 270° to 315° from the sun pulse. On command SAS2 the gate is disabled and the star pulses are passed continuously. The sector 7 signal is produced by division of each revolution into 8 equal segments (sectors) via the SPSG.

2) Star delay produces time delay of the star pulses. The delay can be set by ground commands SAS3, SAS4, SAS5, and SAS6.

3) The star time gate is a real time gate which is enabled on a star pulse and prevents all other star pulses from passing during a fixed time. The time gate is enabled by command SAT1 and disabled by commands SAS1 and SAS2.

4) The star advance capability works in conjunction with the time gate. It advances the time gate so that it begins blanking on the next star following the one that was previously enabling the gate (command SAT1).

5) Star coincidence is used in conjunction with the star delay. It sends a discrete when a delayed star is coincident with an undelayed star. The purpose is to provide a measure of the angle between two stars.

6) The star counter provides a count of the gated stars to telemetry.

7) Star level provides a discrete when the star is greater than 180% of the level of Canopus.

8) Sensor select allows ground command to select the pulse source to the SPSG, i.e., Sun A, Sun B, or star.

9) Star location logic is used in conjunction with the star delay. It indicates whether the delayed star is in Sector 8 or Sector 1.

10) The sector generator divides a given revolution into 8 equal parts and generates signals which coincide with Sectors 1, 7, 8. The sectors are counted in time from the sun pulse.

4.4 OPEN LOOP PRECESSION

4.4.1 Precession Path Description

Open-loop precession of the Pioneer spacecraft is required for ΔV midcourse corrections and other reorientation maneuvers. The precession is effected by firing the velocity-precession thrusters (VPTs) once every spacecraft revolution at a constant pre-selected angle from the sun-spin axis plane. During this maneuver the vehicle's angular momentum vector inscribes a path on an imaginary sphere surrounding the spacecraft which is called the precession path. For each precession step the precession path follows a great circle. However, the total precession path, which consists of a sequence of great circle segments, approximates a loxodromic curve. A segment of the loxodrome is known as a rhumb line. The ideal rhumb line path is at a constant angle to the local meridian line at every point on it. The geometry for the precession path is shown in Figure 4.4-1. The notation is more completely defined in Pioneer Document PC 263.

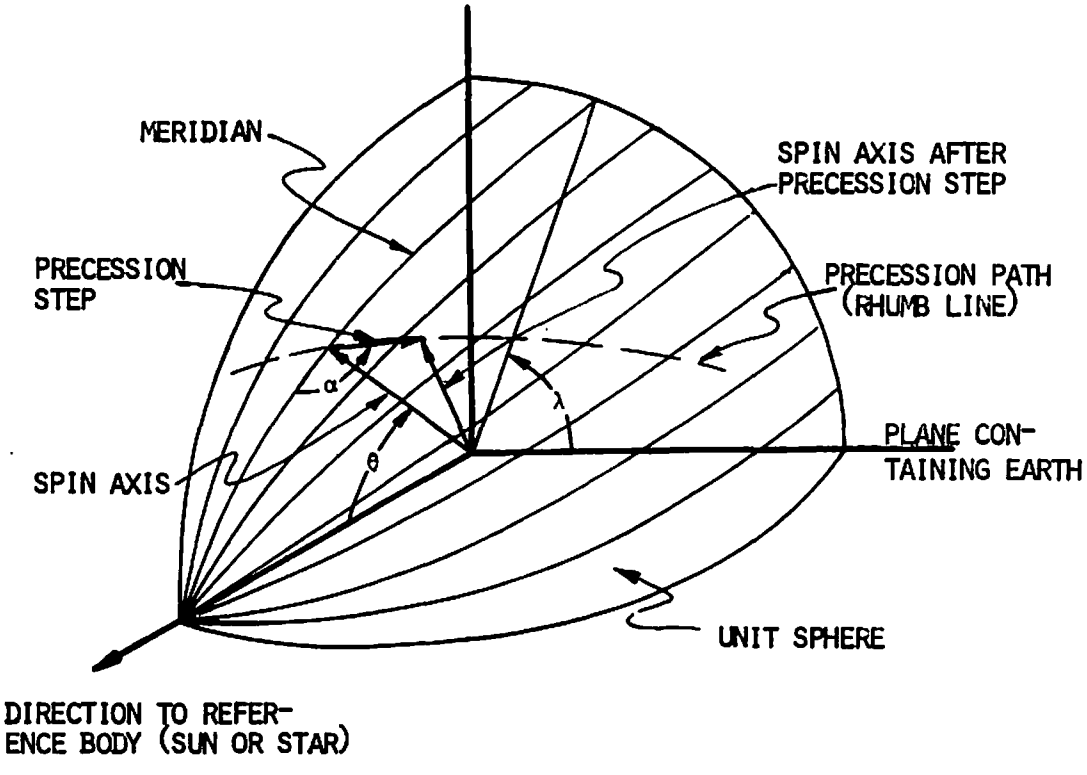


Figure 4.4-1. Geometry and Coordinates of Open Loop Precession

The rhumb line path is always longer than or equal to the great circle path connecting the initial and final angular momentum vectors. The angular length of the rhumb line path depends on the precession angle and the sun-spacecraft-earth geometry at the time of the precession maneuver. As part of a ΔV maneuver the spacecraft might have to be pressed through an angle as large as 90 degrees. However, it can be shown that the longest rhumb line path required for any precession maneuver is 107 degrees and occurs for a precession angle (α) of ≈73 degrees.

For the longest precession maneuver the actual precession path deviates from the rhumb line by about 0.47°. The actual path taken by Pioneer must be corrected for this deviation in what is called the "great circle program."

Figure 4.4-2 illustrates the rhumb line path geometry. Let us define the following quantities:

Symbol	Definition
$\bar{S}$	vector pointing from the spacecraft to the sun or other reference body
$\bar{H}_o$	initial angular momentum vector of the spacecraft (aligned to earth)
$\bar{H}_f$	final angular momentum vector of the spacecraft
ϕ_o	complement of angle between $\bar{S}$ and $\bar{H}_o$ $(90 - \theta_o)$
ϕ_f	complement of angle between $\bar{S}$ and $\bar{H}_f$ $(90 - \theta_f)$
θ_p	angle between $\bar{H}_o$ and $\bar{H}_f$ (precession angle)
λ	angle between the planes formed by the vectors $\bar{S}$ and $\bar{H}_o$ and the vectors $\bar{S}$ and $\bar{H}_f$
α	angle between meridian line and rhumb line path
ψ	angular distance of rhumb line path
Δ_ψ	angular step size of precession
θ_v	angle between the ΔV thrust vector and the initial spacecraft spin axis ($0° \leq \theta_v \leq 90°$ to cover entire sphere for ΔV)

It is assumed that the vehicle's angular momentum vector $\bar{H}_o$ is pointing at the earth at the start of the precession maneuver ($\lambda_o = 0$). After the precession maneuver the position of the final angular momentum vector $\bar{H}_f$ relative to the sun-spacecraft-earth plane is defined by the angles ϕ_o and λ.

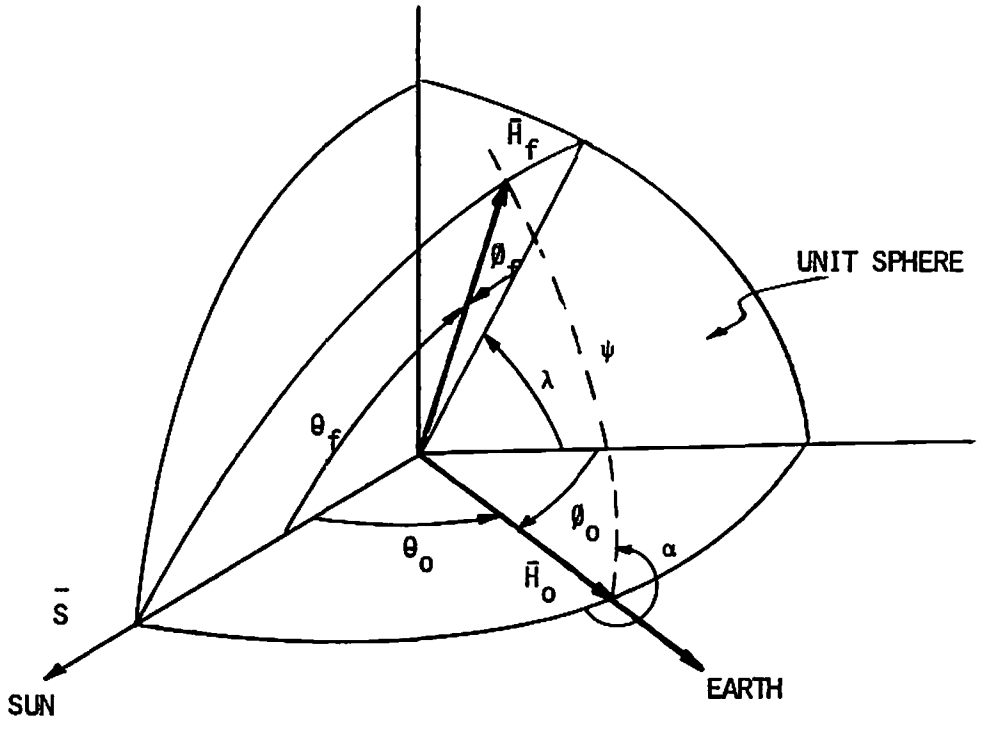


Figure 4.4-2. Rhumb Line Geometry

To obtain the values for storage in the ACS program storage to achieve an open loop precession, a number of steps must be taken. First, the magnitude and direction of the desired ΔV (or final pointing direction) must be determined in the coordinate system indicated in Figure 4.4-1. The ΔV (or pointing) direction will then be given in terms of λ and ϕ_f.

This information must be converted into the rhumb line parameters α and ψ and the time required for ΔV. An example Flow Chart is given in Figure 4.4-4. The precession direction (α) and magnitude (ψ) must be corrected for the quantization in the PSE, for thruster impulse and centroid performance data, for spin rate error, and for path distortion from a true Loxodrome caused by execution in the form of a sequence of great circles. The maneuver should also be modified to avoid regions of large error sensitivity--regions near the $\pm$ sun-spacecraft line. Sensitivities should be checked using system software prior to maneuver execution. Examples of sensitivity are shown in Figure 4.4-3.

Sensitivity Equation:

$$\frac{\partial \phi_f}{\partial \alpha} = -\psi \sin\alpha$$

$$\frac{\partial \lambda}{\partial \alpha} = \frac{1}{\cos^2\alpha} \ln \frac{\tan(\pi/4 + \phi_f/2)}{\tan(\pi/4 + \phi_o/2)} - \psi \frac{\sin\alpha \tan\alpha}{\cos\phi}$$

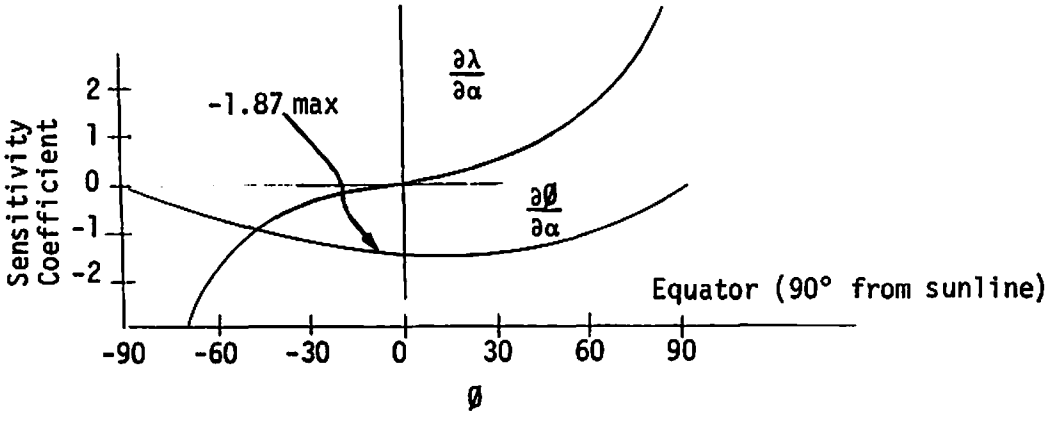


Figure 4.4-3. Nominal Sensitivity to Precession Errors (Worst Case Path Length)

From Figure 4.4-3 it is seen that the regions near the sun or the anti-sun are highly sensitive to error in execution of α. These regions should be avoided by breaking the ΔV into vector components. The SLA (sun look angle θ) is constrained as described in PC 263 by this requirement.

Once the final desired orientation is determined, the system software will transcribe this into the values to be entered into the Attitude Control System CEA-PSE. A flow chart for the operations to be performed by this software is shown in Figure 4.4-4. The operations shown by this figure are as follows:

1) The values of longitude (λ_o, λ_f) and latitude (ϕ_o, ϕ_f), which describe the initial and final orientation, are used to obtain a preliminary estimate of the rhumb line parameters, α and ψ. These describe the angle to the sun, and the length of the ideal rhumb line.

2) Given the nominal rhumb line parameters then the propulsion performance characteristics (impulse correction (k), obtain preliminary estimates of the number of angle bits (n) and the number of time increments (m) to be used in the ACS stored program for precession. This estimate of α is next corrected for impulse centroid location (δ_c and its correction factor c)

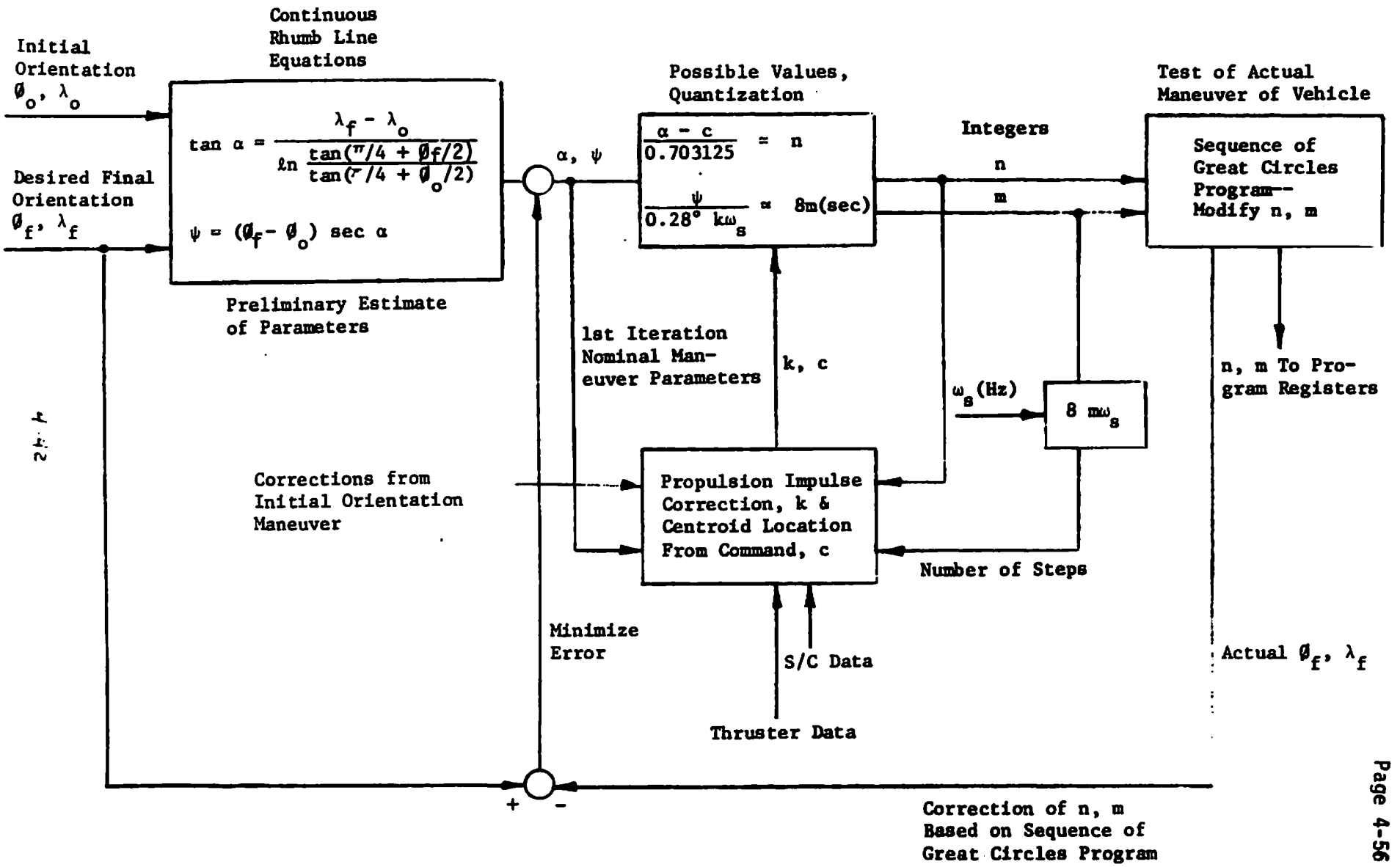


Figure 4.4-4. FLOW DIAGRAM TO DERIVE OPEN LOOP PRECESSION PROGRAM DATA FOR ACS

and for quantization in the α that can be stored in the PSE registers. (Refer to Section 3.3.1.3.2.) Since the impulse torque vector which produces the precession either leads or trails the roll index pulse ($\alpha = 0$) then the α used in the registers must be corrected depending on the thrusters and index pulse used:

$$\alpha_R = \alpha - \delta_c \pm 90^\circ$$

Additional explanation of this correction is given in Section 4.1.4. Note that α is measured from the sun spin axis plane (plane defined by $\bar{S}$ and $\bar{H}_0$) to the plane of precession and is described by the right hand rule.

3) A system test of the estimate of n and m are made via the sequence of great circles program which steps through the actual maneuver and produces error estimates from the desired orientation.

4) The information obtained from the sequence of great circles program is used to improve the estimates of n and m and obtain minimum error in the final orientation.

CAUTION: The continuous rhumb line equations given in Figure 4.4-4 do not completely define α for all possible geometries. Sign conventions must be carefully scrutinized to insure the proper value for α and prevent the complement of α or $\alpha \pm 180^\circ$ from being used. Although either α or $\alpha \pm 180^\circ$ may be mathematically correct for the equations, only one is correct for a given maneuver.

4.4.2 Precession Dynamics

In this section the precession equations for both the open loop maneuver and ΔV/SCT will be presented. The angular momentum step size and wobble angle as a function of spacecraft parameters will be developed. Also the phasing of the wobble envelope minimums will be discussed. In the last subsection a graphical procedure for determining the precession motion will be displayed.

4.4.2.1 Open Loop Precession. Precession of a spinning spacecraft such as Pioneer F/G is accomplished by firing gas jets parallel to the spin axis which results in an applied torque normal to the spin axis. Successive torque pulses will cause the angular momentum vector to precess and will induce a wobble motion of the spin axis about the angular momentum vector.

The angular momentum step size δ_p is given by the following equation:

$$\delta_p = \frac{F_v \tau E_T \ell_v}{I_z \omega_s} \text{ radians} \qquad (4.4\text{-}1)$$

where

$F_v \triangleq$ effective velocity precession thruster (VPT) thrust level, (lb)

$\ell_v \triangleq$ moment arm of VPT about the x_b-axis (ft)

$\tau \triangleq$ VPT pulse duration (seconds)

$E_T \triangleq$ efficiency of the VPT thrusters

$I_x, I_y, I_z \triangleq$ the spacecraft principal moments of inertia (slug-ft^2)

$\omega_s \triangleq$ the spin frequency of the spacecraft (rad/sec)

$F_v \tau E_T \triangleq$ thruster impulse (lb-sec)

The wobble angle η is defined to be the angle between the angular momentum vector and the spin axis (the z_b-axis), and can be expressed as a function of the pulse number i by the following equation:

$$\eta_p(i) = \delta_p \left| \frac{\sin ik\pi}{\sin k\pi} \right| \qquad (4.4\text{-}2)$$

where

$$k = \sqrt{k_x k_y}$$

and

$$k_x = \frac{I_z - I_y}{I_x} \qquad k_y = \frac{I_z - I_x}{I_y}$$

and the first pulse fired is denoted by i = 1. The maximum wobble is easily deduced to be

$$\max_i \eta_p(i) = \delta_p/\sin k\pi \qquad (4.4\text{-}3)$$

The wobble envelope is dependent on the phasing of the gas jet firings. For the open loop maneuver a torque pulse is fired every revolution and by definition the wobble envelope is a relative minimum for the i'th firing if

$$\eta_p(i - 1) > \eta_p(i) < \eta_p(i + 1)$$

In order to calculate this minimum wobble envelope, compute the number i_o given by

$$i_o = \frac{1}{1 - k} \qquad (4.4\text{-}4)$$

By definition, let j equal the greatest positive integer contained in the number i_o. Then the wobble envelope is a minimum for the integer j or j + 1. A minimum can be checked by evaluating

$$\eta_p(j - 1),\ \eta_p(j),\ \eta_p(j + 1),\ \eta_p(j + 2)$$

using equation (4.4-2).

4.4.2.2 ΔV/SCT. Activation of the spin control thrusters (SCT) as part of the alternate ΔV scheme is referred to as the ΔV/SCT maneuver. The ΔV/SCT maneuver is performed by firing the two spin control thrusters alternately; one thruster pulse every half revolution of the spacecraft. The spin control thrusters exert two undesirable torques on the vehicle while they are being used for ΔV thrusting. One torque causes spin variations and the other one generates precession and wobble motion of the spacecraft. The spin variations induced by the two thrusters tend to cancel, and cause a negligible angular error in the pulse firings. In this section the ΔV/SCT precession will be discussed.

The velocity increment ΔV, and the precession step size δ_Δ, are given by the following equations:

$$\Delta V = \frac{F_s \tau E_T}{M} \frac{ft}{sec}, \quad \text{neglecting mass changes}$$

$$\delta_\Delta = \frac{F_s \tau E_T \ell_s}{I_z \omega_s} \qquad (4.4\text{-}5)$$

where

$F_s \triangleq$ the spin control thruster (SCT) thrust level (lb)

$\ell_s \triangleq$ moment arm of SCT about the y_b axis (ft)

$\tau \triangleq$ SCT pulse duration (seconds)

$E_T \triangleq$ efficiency of the SCT thrusters--τ dependent

$M \triangleq$ mass of the spacecraft (slugs)

The maximum wobble angle as a function of the ΔV/SCT pulse number can be generated by the following equations:

$$\eta_\Delta(i) = \delta_\Delta \left(\frac{kI_x}{k_y I_y}\right) p(i) \qquad (4.4\text{-}6)$$

where

$$p^2(i) = J_1^2(i) + J_2^2(i)$$

and

$$J_1(i = 1) = 0 \qquad J_2(i = 1) = 1$$

$$J_1(i) = J_1(i - 1) + (-1)^i \sin ik\pi$$

$$J_2(i) = J_2(i - 1) + (-1)^i \cos ik\pi$$

where i = 1 corresponds to the first ΔV/SCT pulse. Fortunately, the wobble envelope can be approximately described by the open loop precession characteristics. Thus, the maximum wobble is given by

$$\max_i \eta_\Delta(i) \simeq 2\delta_\Delta \left(\frac{I_x k}{I_y k_y}\right) \frac{1}{\sin k\pi} \qquad (4.4\text{-}7)$$

For ΔV/SCT precession the number of pulses between envelope minimums is given by $2i_o$, i.e.,

$$2i_o = \frac{2}{1 - k} \qquad (4.4\text{-}8)$$

Nominal precession characteristics for both open loop and ΔV/SCT induced precession is presented in Table 4.4-2. The spacecraft nominal mass properties are given in Table 4.4-1. Note that the timing between wobble envelope minimums is independent of the torque pulse width, and is given by approximately seven spacecraft revolutions.

Table 4.4-1. Spacecraft Nominal Mass Properties

Parameter	Value
I_x, I_y, I_z	(284.5, 180.5, 433.9) slug-ft^2
M	17.0 slugs
ω_s	0.503 rad/sec
F_v(two thrusters)	2.2 lb
F_s	1.1 lb
ℓ_v	4.19 ft
ℓ_s	1.101 ft
$k_x = (I_z - I_y) I_x^{-1}$	0.8906
$k_y = (I_z - I_x) I_y^{-1}$	0.8277
$k = \sqrt{k_x k_y}$	0.8586
$(1 - k)^{-1} = i_o$	7.072
$\theta = r(1 - k)2\pi$	50.9 degrees (r = 1)
$\left(\frac{I_x k}{I_y k_y}\right)$	1.635

Table 4.4-2. Nominal Expected Precession Characteristics

Open Loop Precession					
Pulse Duration(τ)		E_T	Precession	Maximum Wobble Angle	
0.125 sec		0.1	0.3025°/step	0.7039°	
ΔV/SCT Induced Precession					
No. of SCT Pulses	τ(sec)	E_T	$\Delta V(\frac{\text{meters}}{\text{sec}})$	Precession	Max. Wobble
1	2.0	1.0	0.03944	0.6352°	4.834°
2	1.0	1.0	0.03944	0.6352°	2.417°
4	0.5	0.95	0.03747	0.6036°	1.148°
16	0.125	0.85	0.03352	0.540°	0.2568°
64	0.03125	0.65	0.02563	0.4129°	0.0491°

4.4.2.3 Graphical Construction of the Precession and Wobble Motion. Contained herein is a graphical construction of the precession motion for open loop and ΔV/SCT precession. The construction is based on a dynamics consideration of a symmetrical vehicle, and represents worst case for the asymmetrical vehicle.

A torque impulse will cause the angular momentum vector to change direction instantaneously in the direction of the torque impulse, and will induce the spin axis to cone about the "next" angular momentum vector. Thus, in inertial space the wobble cone angle η for the i-th pulse ($\eta(i)$) is given by the angular distance between the location of the spin axis and the new angular momentum vector. A typical open loop precession maneuver is displayed in Figure 4.4-5. The spacecraft orientation is determined by displaying the projection of the spacecraft spin axis onto the x_o, y_o inertial reference frame. Referring to Figure 4.4-5, it is assumed that initially the angular velocity and angular momentum vectors are aligned. Thus, the angular momentum vector is located at the origin initially. A precession pulse fired in the precession plane causes the initial angular momentum vector H_o to step δ_p degrees over to H_1. The spin axis then rotates in a counter-clockwise motion about H_1 until and next torque impulse is fired. Hence, the spin axis describes a circle with radius (wobble angle η_1) δ_p and centered at H_1. In order to construct the precession path it is only necessary to find the counter-clockwise angular rotation, ψ, that the spin axis cones between precession pulses. This counter-clockwise distance ψ is given by

$$\psi = r(1 + k)\ 2\pi \text{ radians}$$

where

$$r = \text{the number of revolutions between pulses}$$

and

$$k = \sqrt{k_x\ k_y}$$

For graphical construction, the effective clockwise rotation θ is more useful:

$$\theta = r(1 - k)\ 2\pi$$

For open loop precession one torque pulse is fired every revolution, thus, $\theta = (1 - k)\ 2\pi$; for $k = 0.858$, $\theta = 51$ degrees. Hence, when the second torque impulse is fired the spin axis is located at z_2 in Figure 4.4-5. The angular momentum vector steps from H_1 to H_2 and the wobble angle η_2 is equal to the angular distance between the spin axis (z_2) and the angular momentum vector (H_2). Thus after the second torque impulse, the spin axis will describe a circle with radius η_2 centered at H_2. Continuing on with this step wise procedure results in Figure 4.4-5.

In Figure 4.4-5 a typical open loop precession path is constructed using nominal parameters, as described above. Note that initially the angular momentum vector and angular velocity vector were aligned with the z inertial axis. In Figure 4.4-6 the precession path during a typical ΔV/SCT maneuver is displayed. For both constructions, the angle θ is the only quantity that has to be computed in order to construct the spin vector trajectory. Both inertial axes are scaled to the appropriate precession step size δ.

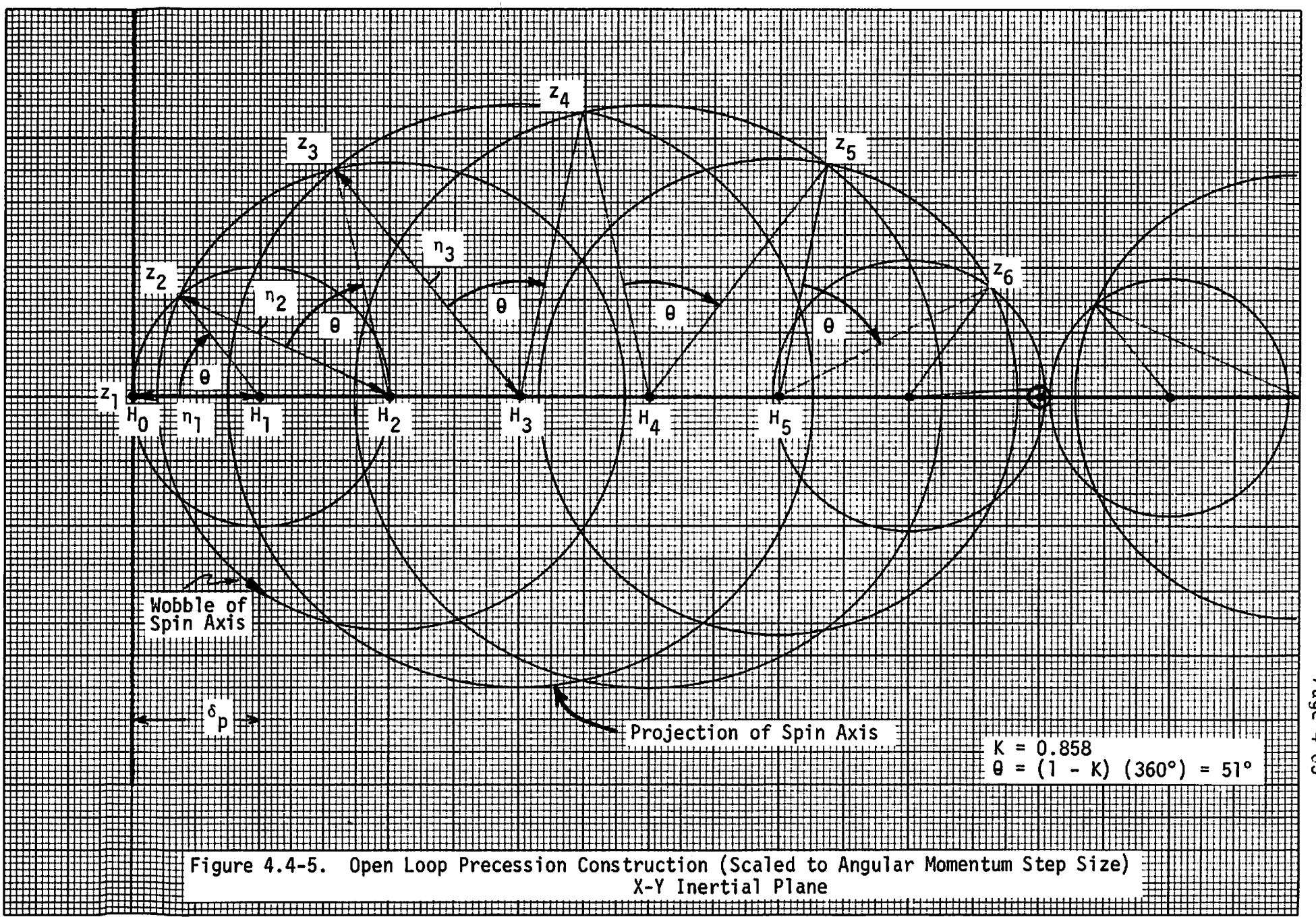


Figure 4.4-5. Open Loop Precession Construction (Scaled to Angular Momentum Step Size) X-Y Inertial Plane

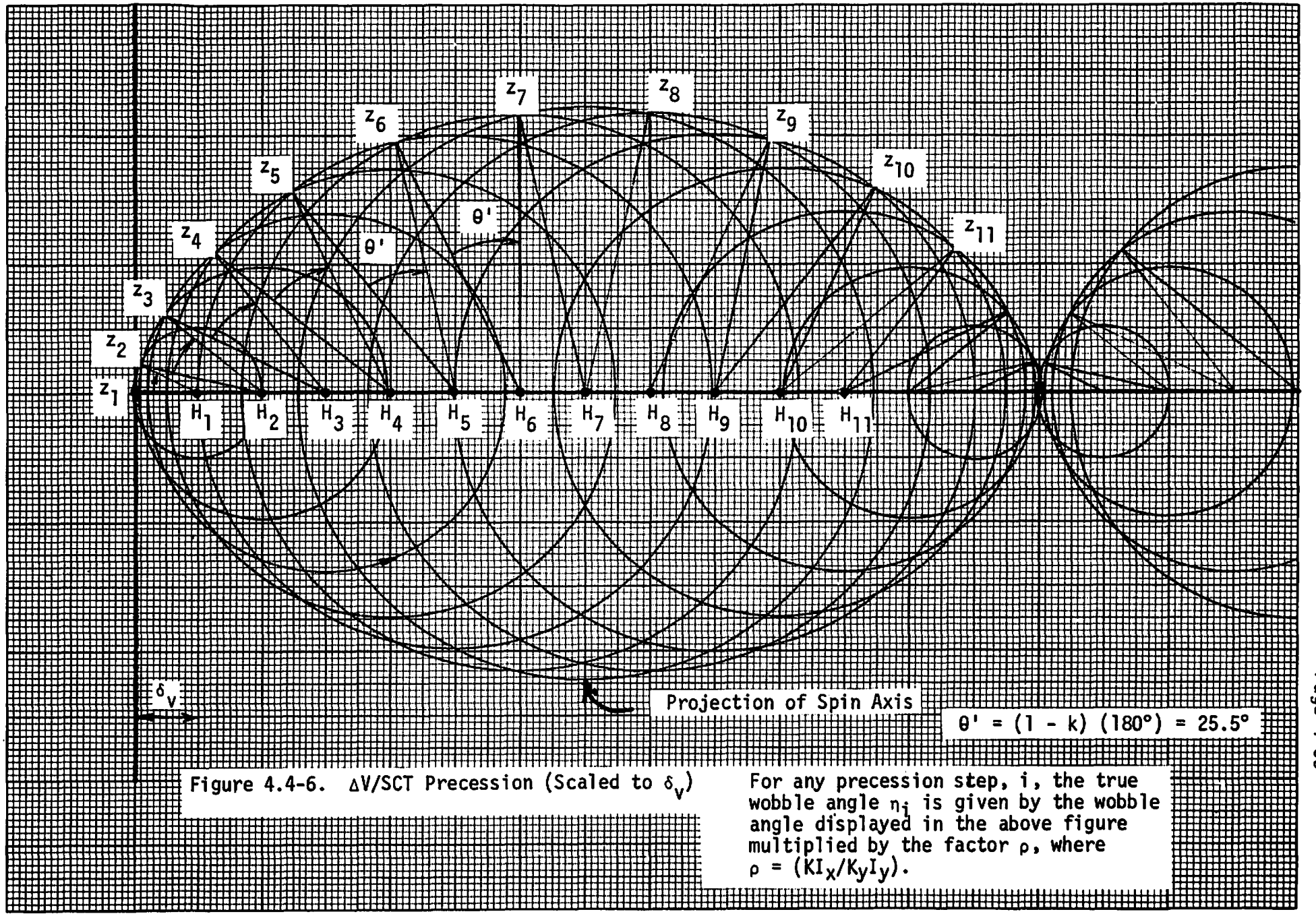


Figure 4.4-6. ΔV/SCT Precession (Scaled to δ_v)

For any precession step, i, the true wobble angle n_i is given by the wobble angle displayed in the above figure multiplied by the factor ρ, where $\rho = (KI_x/K_yI_y)$.

4.5 MIDCOURSE DELTA VELOCITY (ΔV) MANEUVERS

Velocity can be added to the spacecraft's trajectory in any selected direction by means of the "midcourse maneuver." The primary purposes of the midcourse maneuver are to correct for unavoidable injection error of the launch vehicle and to remove any injection bias which may be used to avoid biologically contaminating Jupiter. These interrelated factors cause an initial displacement from the desired trajectory which will be large compared with experiment requirements for planetary fly-by. Up to approximately 200 meters/second maneuver capability is available, depending upon final fuel loading and thruster efficiency.

Operationally, the ΔV maneuver consists of determining the trajectory after launch, evaluating the need for adjustment to achieve Jupiter encounter objectives, selecting the method of maneuver to be employed, and executing the appropriate commands.

Trajectory measurement and orbit determination are relatively standard processes developed by the Jet Propulsion Laboratory. Doppler rate data results from comparison of the signal returned from the spacecraft's coherent transponder with the uplink radio frequency upon which it is based. Accurate to approximately one millimeter per second, these data are combined with known motions of the Earth and its ground stations to produce, with the first two flight days, a precise projection of Jupiter encounter. As the spacecraft moves farther from Earth, tracking intervals necessary for maximum accuracy quickly increase to ten or twenty days.

The approximate magnitude and direction of velocity adjustment required will be calculated with a specially prepared midcourse maneuver computer program as soon after launch as trajectory estimates are available. While these data are being refined, strategy for the corrective maneuver will be developed using the capabilities of the maneuver program. One, two, or more successive maneuvers can be executed such that their vector sum produces the required correction. Considerations in selecting the course of action will include:

a) The encounter location objective and, if unreachable, the alternative to be selected.

b) The timing of encounter, and the epoch to which it will be adjusted.

(Timing of encounter will control visibility from the largest tracking antennas; imaging possibilities for satellites and their shadows, and for the red spot on Jupiter; and magnetic and particle measurements with respect to the satellite Io, belived to have a magnetic field.)

c) Precautions to be included in maneuver planning to prevent the possibility of impacting Jupiter.

d) Solar radiation danger to instrument sensors during the maneuver process; approach toward the Sun by the spacecraft spin axis which might compromise the roll orientation reference; and solar heating of the equipment compartment on the backside of the parabolic antenna.

e) The possibility of constraint such that the spacecraft antenna system will afford continuous communications with the ground station for monitoring and command control.

f) Advisability of demonstrating performance of the subsystem and appropriate components prior to committing a major proportion of the on-board fuel by programmed commands.

On the basis of these factors, comparisons will be made of alternative combinations of maneuvers which would achieve the necessary velocity vector. Total fuel consumption, loss of communications at large turn angles, and relative orientation of the Sun will be considered individually and in combinations.

Two methods of obtaining a velocity correction for Pioneer are available, both performed by the ACS subsystem using the propulsion thrusters and the spin period sector generator in the DTU. These two methods are, first, the primary ΔV (the spacecraft is precessed to the desired orientation, the ΔV is executed with the velocity-precession thrusters, and the vehicle is precessed back to within Conscan range of the original orientation) and second, the ΔV/SCT mode (ΔV is broken into components, one along the spin axis performed with the VPTs, and one perpendicular to the spin axis performed with the SCTs).

A selection of the method to perform the ΔV based on fuel consumption only depends on the magnitude of the maneuver to be performed. The break-even point on fuel consumption for ΔV by precession vs ΔV/SCT lies between 2 m/sec and 6 m/sec as shown in Figure 4.5-1. The fuel required depends on the angle from the earth-spacecraft line to the line of action of the desired ΔV (Velocity angle). The tradeoff is made between the fuel required for precession vs the inefficiency of using vector components.

4.5.1 ΔV via Open Loop Maneuvers

The Program Storage and Execution assembly (PSE) in the Attitude Control Subsystem was designed specifically to perform a ΔV. In the registers, data is stored to precess the spacecraft to any desired orientation, delay, fire a ΔV of any magnitude up to 200 m/sec, delay, then precess back to the original (or any other) orientation. The operation of the PSE for the ΔV is identical to that of any open loop maneuver as discussed in Sections 4.4 and 3.3.1.3.

The ΔV can be performed with either of 2 ΔV thruster pairs, facing the +z and -z directions on the spacecraft respectively. The spacecraft can be precessed to use either pair in any direction however, one pair should be selected as a preferred (calibrated) pair - that used for the calibration maneuver. The maximum angle of the spin axis to the earth (θ_V) should be less than 90°. The maximum length of the rhumb line (ψ) can be as much as 107°. Some characteristics of the ΔV capability are listed in the following table. (See Section 3.3.1.3.2)

Table 4.5-1. ΔV Characteristics

Commandable duration	8191 increments (13 bits)
Increment sizing (two VPTs)	1.0 sec @ 0.038 m/sec, or 0.117 ft/sec
Single pulse size	2.0 sec, 1 sec, 0.5 sec 0.125 sec or 0.03125 sec (selectable)
Precession duration	16,376 sec
Precession increment	8 sec
Maximum delay	8,128 sec
Time of delay increment	64 sec
Spin error due to 0.1 deg misalignment for total ΔV (5600 sec) capability (2 VPTs misaligned)	1.29 rpm
Precession error after ΔV performed with one VPT	3.73 deg
Wobble with ΔV performed with one VPT	4.8 deg

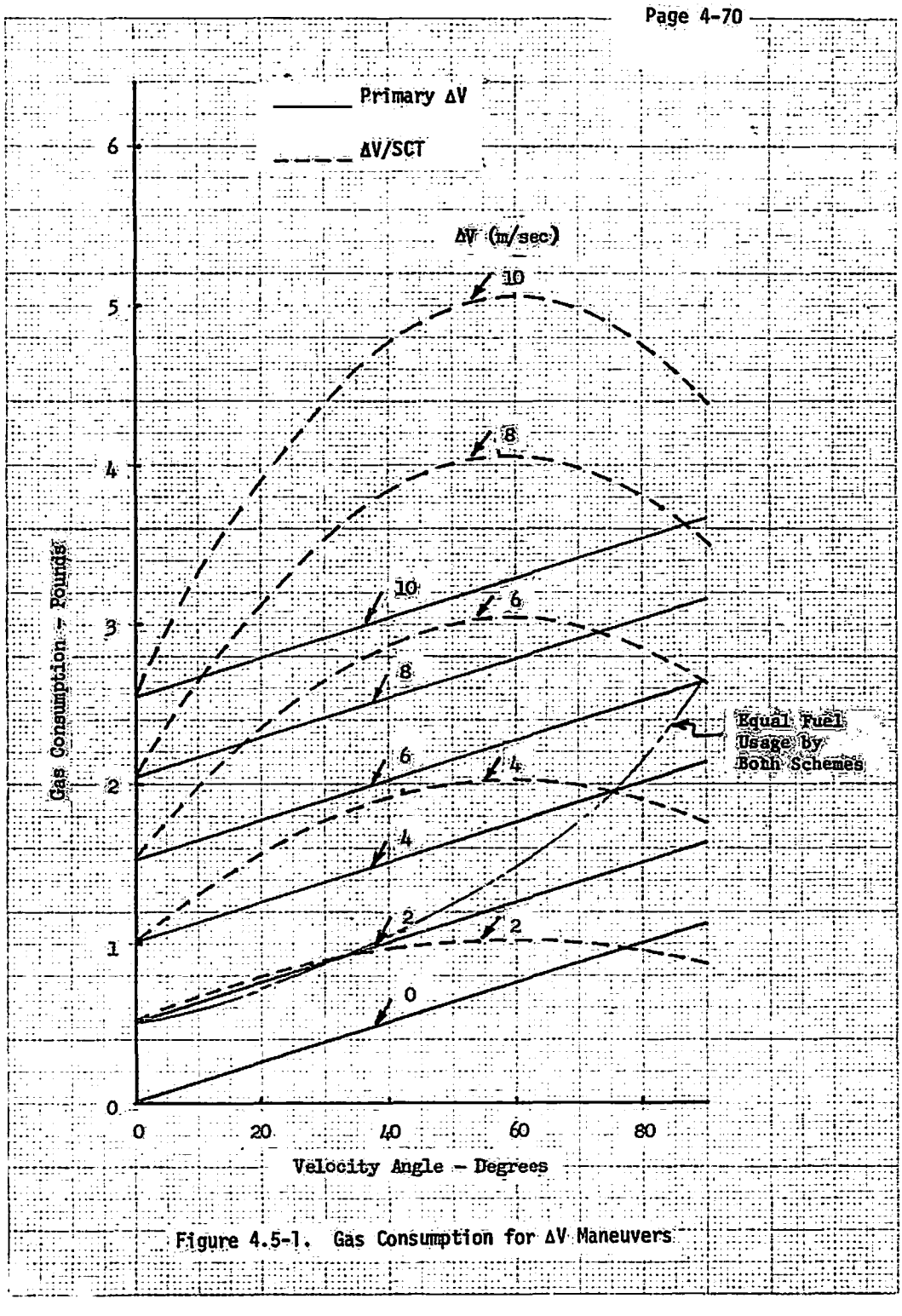


Figure 4.5-1. Gas Consumption for ΔV Maneuvers

Two important restrictions should be observed on the execution of the ΔV:

- Spin coupling must not allow the spacecraft to despin. Note from the above table, that a 0.5° misalignment of the ΔV thrusters could cause a spin rate change of 6.45 rpm. A measure of spin coupling can be obtained during the initial orientation maneuver (thruster calibration maneuver, Section 4.1). The ΔV should be broken into segments with spin control performed between segments such that error due to spin change is minimized. (See Section 3.3.2.2 for spin control)

- A ΔV must not be attempted within 10° of the sun or the anti-sun line. The anti-sun line is restricted because of sun sensor deadzone. Both the sun line and the anti-sun line are restricted because of high sensitivity to errors in precession path execution in these regions (Section 4.4).

4.5.2 ΔV/SCT

Two midcourse corrections are likely for Pioneer F/G during the early part of flight. The primary method for performing ΔV is to precess the spacecraft spin axis to the direction of the ΔV vector, execute, and then return the spacecraft to its original orientation (Sections 4.5.1, 4.4). Earth pointing errors after such a maneuver should be less than 8° to remain with the pointing control capability of Conscan using the medium gain antenna (range of approximately three astronomical units). When Pioneer is further from the earth than 3 A.U., the high gain antenna would normally be used for Conscan operation with a control capability of about two degrees.

A ΔV maneuver may become necessary at a late stage of the flight. Since after a precession maneuver the earth pointing error might be excessively large, the ΔV/SCT method for ΔV has been devised. The ΔV/SCT requires no reorientation of the spacecraft because the total ΔV is resolved into two perpendicular components. The velocity-precession thrusters are used for the component parallel to the vehicle's spin axis, while the spin control thrusters provide the thrust component perpendicular to the spin axis. Activation of the spin control thrusters for ΔV is called to ΔV/SCT maneuver in this document, however it is also referred to as a "no turn" maneuver. The ΔV/SCT maneuver fires the two spin control thrusters alternately; one thruster pulse every half revolution of the spacecraft.

The spin control thrusters exert two undesirable torques on the vehicle while being used for ΔV thrusting. One torque causes spin variations and the other generates precession and wobble motion of the spacecraft. The precession motion occurs because the spin thruster line of action is 1.1 feet above the vehicle c.g. which results in a torque (and precession) along the spacecraft y axis.

The ΔV/SCT maneuver is controlled by the stored program function of the PSE. That is, the same registers and counters used for the large angle precession are used for ΔV/SCT. The first precession angle and magnitude counters are used to determine the firing angle and the length of firing time for ΔV/SCT. The ΔV counter is set at zero, and the second precession counter is used to fire VPT precession pulses in the ΔV/SCT mode, not SCT pulses. This second precession may be used to precess the spin axis back to its original position. The spin axis wobble motion during two ΔV/SCT pulse sequences is shown in Figure 4.5-2 for the 0.125 sec SCT pulse width.

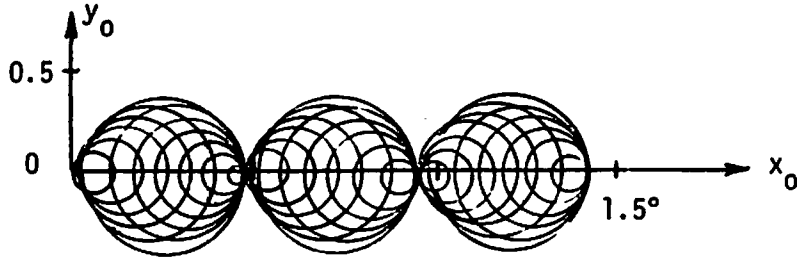


Figure 4.5-2. Precession and Wobble Motion During ΔV/SCT

The wobble amplitude (Section 4.4.2) varies with the number of torque pulses fired consecutively. For the lowest wobble amplitude, a sequence of 12, 24, 36, or 48 SCT pulses should be used. (The last statement is valid as long as no major changes in spacecraft mass properties occur). The precession and wobble amplitudes at the end of 12 SCT pulses are 0.48 and 0.033 degrees respectively when the pulse duration is 0.125 seconds. Note that this is nearly a 6 cycle (12 pulses or 6 pair) deadbeat system. The precession angle is corrected by firing the velocity-precession thrusters as programmed in the second precession registers. Before the next sequence of SCT pulses starts, the wobble motion caused by the VPT pulses should be allowed to die out, if the number of VPT pulses does not result in minimum wobble. A single SCT pulse of 0.125 sec could yield 0.0022 m/sec ΔV. By proper operation of this mode, a ΔV of 0.2 m/sec can easily be obtained. If timed commands from the ground are used this capability can be extended.

The ΔV/SCT effects will be described by examples. The angle and magnitude of the ΔV/SCT is entered into the first precession register and the ΔV/SCT Mode Command is sent (VEL 4). The DSL is set with a desired pulse length, say 0.125 sec (Command PUL 2). Upon receipt of command sequence execute (ACX 1), the program will begin by entering state 1 (delay), performing identically as for an open loop maneuver. After completion of state 1, the program enters state 2. In this state, SCT #1 will fire a single 1/8 second pulse at the angle entered in the angle register. Approximately 180° later, SCT #2 will fire a 1/8 second pulse. This continues for the duration of the time in the 1st precession magnitude register, or until the sequence is inhibited or stepped.

After the SCT #1 pulse, the spin rate will increase approximately 0.0128 rpm, depending on impulse. After 256 counts from the SPSG at 512 ppr, SCT #2 will fire a nearly identical pulse thus returning the spin rate to nominal at 4.8 rpm. The average spin rate for the cycle will be approximately 4.8064 rpm and this will be the rate sensed by the SPSG for the remainder of the ΔV/SCT. The two pulses fired will not be parallel because of this spin change by approximately 0.48°, since the spin rate has changed from that given by the SPSG during the first revoltuion. The third pulse will not be parallel to the first pulse, with an error proportional to the angle in the first precession angle register. This error will be about [(0.48°/512) x Binary number in Angle Register]. These small changes in direction of ΔV/SCT pulses can be fully accounted for in the support software if large SCT pulse widths are used, or neglected if they are small as in this example.

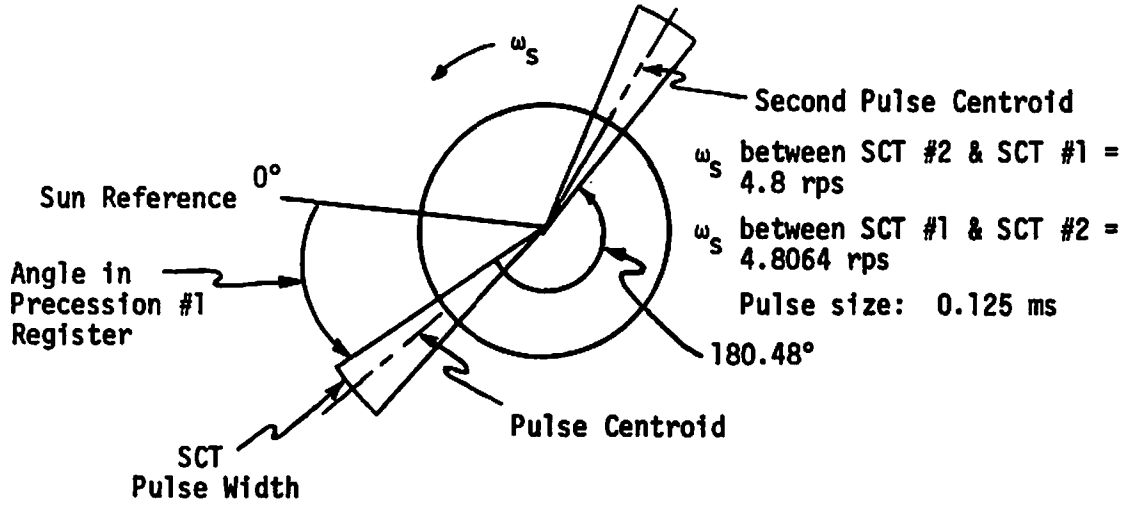


Figure 4.5-3

The precession motion is that which governs the size of the ΔV/SCT maneuver. It is assumed that mission constraints would limit precession to ± 2° from earth line (range of Hi-Gain Conscan) to prevent communication loss. In the example, assume the spin axis is initially pointed at the center of the tracking station. A ΔV/SCT of 48 pulses would precess to the precession angle limit. The precession motion for each pulse is in the direction of the +y body axis for SCT #1 and -y body axis for SCT #2. Since the SCTs are fired about 180° (256 counts) apart, the precession will add, not cancel.

In setting up a program to be executed for ΔV/SCT, several options are available.

1) Load program with maximum ΔV/SCT precession of 2° and precession #2 register with that to cancel ΔV/SCT precession. Program can be reloaded and executed by timed commands from the ground.

2) A large ΔV/SCT program can be loaded and interrupted by sequence step commands stored in the on-board command programmer. Such a program might be as follows:

Time	Command	Comment
Pre-execute	PSE9	PSE On
	ACX4	Reset
	SLA9	DSL-A On
	ACR1	
	VEL1	ΔV pair select, always select all pairs of thrusters even if not used so that state is known.
	PRE1	+x precession direction
	VEL4	ΔV/SCT Mode: always fires SCT #1 first, will be reset on PSE0 or ACX4.
	PUL2	Precession performed with 0.125 sec pulse
	ACS1	Enter ΔV/SCT of 0.167 m/sec (76 pulses, 38 pairs, or 475 seconds) at an angle of 28.1° (40 bits).
	ACS2	Enter delay of zero and zero ΔV. This is used in the example only. If commands for interruption of the program are to be stored in the CDU, the delay value is crucial since these commands must occur such that space-craft pointing error does not exceed 2° as in this example.
	ACS3	Enter precession of 3.0° to cancel ΔV/SCT precession magnitude error. (10 steps at 0.29°/step is 125 sec at 4.8 rpm). Angle of 118.1° (168 bits) will cancel ΔV/SCT precession direction since these precessions are 90° apart.
	CDS1,ACX1,0 CDS2,ACX5,272 CDS3,ACX5,1152 CDS4,ACX1,1280	Store commands in programmer for execution of the PSE program.

Time	Command	Comment
t=0	CDX1	The execute stored sequence command CDX1 will cause the sequence execute command to be issued. ΔV/SCT pulses will occur until the sequence step command (ACX5) 272 seconds later. At this time, 44 pulses, 1.74° of precession and 0.096 m/sec ΔV will have occurred. The program will go to the next non-zero state, the second precession. In this state, the spacecraft will precess back 2.9° (1.16° from the start) and then go to the program complete state. After wobble has died out, the next sequence step command returns the program to state zero where it is executed again. At this time the first precession register is the only one with data, the remaining part not previously executed. It has .071 m/sec ΔV, 1.3° precession, or 32 pulses in it. After execution the program will step to state complete since all other states are empty.

This simple example illustrates the capability of the system. Given a desired ΔV and the performance characteristics, a program can be developed to generate the ΔV. Use of the delay can be made to aid in selection of the proper stored programmer command times to properly advance the program and maintain earth pointing within 2°.

Various pulse widths may be used for the ΔV/SCT mode. Although the impulse varies with propulsion variables, the nominal values for a single SCT pulse are given in the following table.

Pulse Width sec	Spin Change rpm	Precession deg	ΔV m/sec	Wobble (peak)
0.03125	0.0032	0.0099	0.0008	0.06°
0.125	0.0128	0.0396	0.0022	0.25°
0.5	0.0514	0.158	0.0088	1.0°
1.0	0.103	0.317	0.0176	2.0°
2.0	0.205	0.634	0.0352	4.0°

The peak wobble occurs after about 3 pulse pairs. A minimum wobble will be obtained for 6 pulse pairs. Computations to compute the number of steps between minimum wobble amplitude are given in Section 4.4.2.2.

The ΔV/SCT mode has several operational alerts which are given below:

1) The SPSG should be in the ACS mode. Phase error will grow with any other SPSG mode which will cause the ΔV to shift with the phase change.

2) If the angle in the first precession register is greater than 180°, the ΔV/SCT pulses will occur every other revolution. Since wobble and accumulated error are both increased in magnitude in this operational state, and since operating time is doubled for a given ΔV, the angle should be constrained to less than 180°.

3) Because of thruster mismatch, spin speed changes will occur. With time entered into the registers, the possibility of an incorrect number of pulses is increased with this mode.

4) Upon selection of the ΔV/SCT mode by command (VEL4) the SCT-1 thruster is automatically fired first. To have SCT-2 fire first, then the command VEL4 must be followed by a command to select SCT #1, (command SPN3). Note that selecting SCT #1 causes SCT #2 to fire first in this mode.

4.6 ATTITUDE CHANGES DURING MISSION

In this section the various adverse control disturbance environments in which the vehicle must function are briefly described. The spacecraft will always be under the influence of environmental disturbing effects (torques). Several of the more significant disturbance sources are described below. Also in this section, the communication problem during sun-earth-spacecraft alignment will be discussed. The solution for this problem involves a spacecraft reorientation away from the earth.

The two most significant environmental disturbance sources for Pioneer F/G are:

- Solar radiation torques
- Induced magnetic torques

The effects of solar radiation pressure on attitude drift will be summarized in the first subsection. This disturbance will be most significant during the first 300 days of the mission. The effects of the Jovian magnetic field on the attitude and spin rate of the Pioneer spacecraft will be presented in the second subsection.

The third subsection will describe the various options available during syzygy, i.e., spacecraft-earth-sun alignment.

4.6.1 Solar Radiation Pressure

Bombardment of the various surfaces of the Pioneer spacecraft by photons emanating from the sun produces small, but significant, forces. Because of the center of gravity-center of pressure offset, the solar forces (pressure) will result in an unbalanced torque applied to the vehicle that will cause the momentum vector to drift.

The Pioneer spacecraft rotates about an axis perpendicular to the plane containing the reflector face. This spin axis is always directed towards the earth during the course of the spacecraft's mission from earth to Jupiter. The angle of incidence (γ) that the sun makes with the spacecraft changes continuously during its trajectory. In addition, as the spacecraft rotates, the angle of incidence is changed due to the changing orientation of the spacecraft. Further, certain portions of the spacecraft experience "shading" from the sun as the spacecraft rotates.

Taking into account all of the above factors, the total average torque per spin cycle can be calculated. Hence, the attitude drift rate as a function of mission time can be computed. The results of the solar torque induced attitude drift is displayed in Figure 4.6-1. The major cause of the initial high drift rate is due to the large angle between the spacecraft face and the sun vector. The direction of the steady precession (drift) caused by solar torque is positive about the S/C - sun line as given by the right hand rule. The spin axis - sun line angle will remain constant at the value of γ.

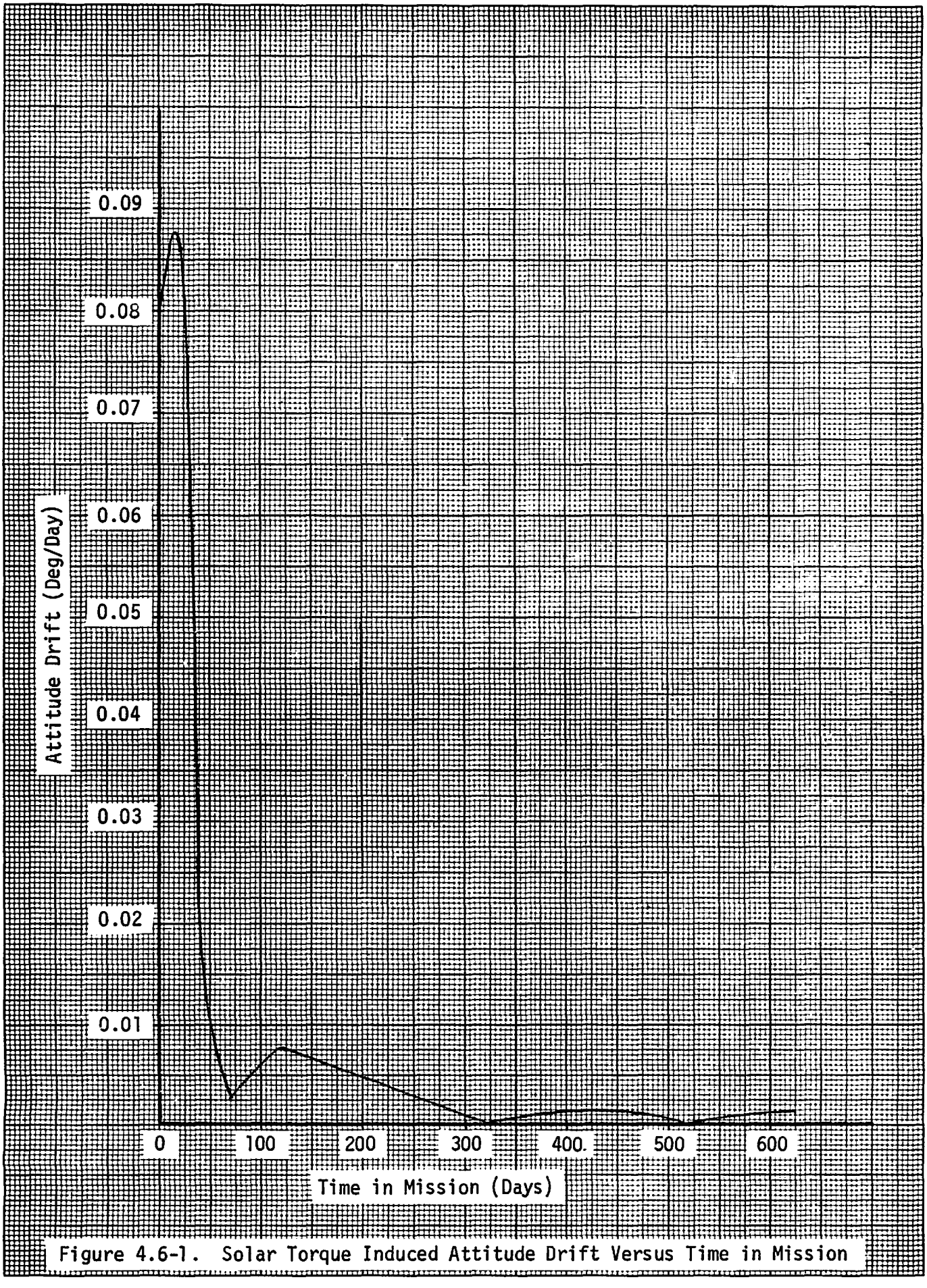


Figure 4.6-1. Solar Torque Induced Attitude Drift Versus Time in Mission

4.6.2 Magnetic Torques

In order to evaluate the effects of the Jovian magnetic field of the attitude and spin rate of the Pioneer spacecraft during its near-planetary passage, many uncertain parameters have to be estimated. The results that follow have been developed using the following assumptions:

- A peak planetary field of 1 gauss at 3 R_J from the planet center (i.e., at periapsis), in the equatorial plane.
- The Pioneer F trajectory at encounter lies in the equatorial plane of Jupiter (coincident with the magnetic equator).
- Characteristics of the spacecraft are estimated where necessary.

These assumptions tend to give a "worst case" situation.

The nature and magnitude of the effects considered are summarized in the following table.

Table 4.6-1. Effects of the Jovian Magnetic Field on Spacecraft Attitude and Spin Rate

	Interaction of Jovian Magnetic Field With:		
	Intrinsic magnetic moment of the spacecraft, due to ferro-magnetic materials and stray currents, with components:		Eddy currents induced in the spinning spacecraft by the Jovian magnetic field:
	Perpendicular to spin axis	Parallel to spin axis	
Nature of effect	negligibly small fluctuation of spin rate, at frequency = 1 per revolution	very small moment tending to precess spin axis in ecliptic plane	very small torque tending to retard the spinning motion
Accumulated magnitude of effect, integrated over the passage of the spacecraft past Jupiter	no change in spin rate	precession through 0.05° in ecliptic plane	reduction of spin rate by 0.002 times the initial value
Dependence on B, the Jovian magnetic field	$\propto B$	$\propto B$	$\propto B^2$

4.6.3 Communication During Syzygy

This section describes options available and various spin axis pointing maneuvers determined by these options; the objective is to maintain to as great an extent as possible up-and-down link communications and roll attitude information during periods of spacecraft-earth-sun alignment. The factors which make attitude determination and communication maintenance more difficult during times of small sun-probe-earth angles are:

- Performance degradation of SSA (sun sensor assembly)
- Extreme sensitivity of roll reference pulse accuracy to spin axis pointing uncertainties
- Necessity for using the spacecraft high gain antenna for probe-earth distances greater than 3 A.U. for conscanning and data transmission
- Severe attenuation of uplink and downlink signals by solar corona during occultation

Typical trajectories for Pioneer F and G have three syzygies. The first occurs approximately 70-80 days into the mission when the earth passes approximately between the spacecraft and the sun (transit). The second occurs about 310-320 days into the mission, when the sun passes between the spacecraft and the earth (occultation). The third syzygy is about 515-525 days into the mission and is a transit, similar to the first one. (Some 1975 launch dates have a fourth syzygy, which is an occultation.)

Spacecraft antenna pointing constraints are summarized in Table 4.6-2. The probe-earth distances correspond approximately to probe-earth separations at the various syzygies described above. (The 6.0 A.U. separation corresponds to a fourth syzygy which occurs for some 1975 launch dates.)

The problem, as indicated above, is to maintain approximate earth pointing and hence up-and-down link communication. The various alternatives are enumerated below. (It is assumed the spacecraft is nominally pointing in the earth direction prior to precession across the sun.)

1) The simplest situation is one in which the SRA (Stellar Reference Assembly) is operative and Canopus acquisition has been verified. The spin axis is then simply precessed from one side of the SSA deadzone to the other, in an open loop manner; the spin axis should not be left pointing in the SSA deadzone since roll attitude information would then be unavailable in the event of loss of Canopus lock.

   The minimum angle to be precessed through is, of course, the two-degree sun sensor deadzone; the commanded angle should be somewhat larger to account for uncertainties in the magnitude of the angle precessed and starting point uncertainty.

Table 4.6-2

Approximate Probe-Earth Distance		0.5 A.U.	3.5 A.U.	4.5 A.U.	6.0 A.U.
Conscan	85' Dish	±10° on MG	±2.5° on HG	±2.5° on HG	±2.5° on HG
	210' Dish	±10° on MG	±2.5° on HG	±2.5° on HG	±2.5° on HG
Downlink	85' Dish	±10° on MG	±2.0° @ 256 bps on HG	±2.0° @ 128 bps on HG	±2.0° @ 128 bps on HG
	210' Dish	±10° on MG	±3.0° @ 2048 bps on HG	±3.0° @ 2048 bps on HG	±2.0° @ 2048 bps on HG
Uplink	85' Dish	±10° on MG	±10° on MG	±3.0° on MG	±3.0° on MG
	210' Dish	±10° on MG	±10° on MG	±10° on MG	±10° on MG

The nominal angle of the precession plane is at zero degrees to the ecliptic. With the SRA operative, the "star delay" may be used to advance the roll reference pulse, and consequently the time of thruster firing, to correct for non-correspondence of the centroid of the firing pulse with a 0° (or 180°) roll attitude. Pulses may be commanded:

a) Singly (in real time)

b) By a program stored in the PSE

c) Using the stored command provision of the CDU

The spin period sector generator (SPSG) may be in either spin averaging, nonspin averaging, or ACS modes.

The constraints on antenna pointing are illustrated for the three syzygies of interest in Figures 4.6-1, 4.6-2, and 4.6-3. Each figure is a plot of the projection on the ecliptic of spacecraft-earth angle and spacecraft-sun angle, measured with respect to the spacecraft-earth (or spacecraft-sun) line at the time of syzygy; and the figures are for a particular mission, but can be considered representative.

2) If only the sun sensor is available for attitude reference, the situation is somewhat more complicated, mainly to the fact that when the spin axis is pointing in the sun sensor deadzone, roll index pulses are either non-existent or have severely degraded accuracy. For this reason, the syzygies of Figures 4.6-2, 4.6-3, and 4.6-4 below will be treated separately.

Figure 4.6-2: As indicated above, Figure 4.6-2 is a plot of the celestial longitude of the spacecraft-earth line and the spacecraft-sun line around the time of the transit at launch + 60 days. The zero reference longitude is the longitude at alignment (rather than the vernal equinox).

It is assumed that, prior to the time of sun crossing, the spin axis is nominally earth-pointing; this orientation would most typically have been attained by means of a Conscan maneuver. At alignment -1.5 days, an open loop maneuver is commanded at an α of 90° to cone about the sun. A precession step pulse width of either .125 sec or .03125 sec should be used so that wobble will not cause loss of the sun pulse. The ACS mode of the SPSG should be used. This maneuver will move the spin axis pointing direction to the opposite side of the sun. This maneuver can also be acheived by commanding an open loop maneuver straight through the sun. If this is done, the SPSG must be in the spin average mode, and must be placed in phase synch shortly (one minute) before commanding the maneuver since the sun pulses will be lost. Care must be exercised since the roll information will degrade quickly once the sun is lost. The maneuver must be complete before the next SPSG period. Large precession steps (1° or more) should be used. The entire maneuver could even be made in a single step. The precession commands may be made either as a stored program in the PSE or by using the

stored command provisions of the CDU (only five commands can be stored). When the spin axis emerges from the sun sensor deadzone, roll attitude information is once again available; also, since the medium gain antenna may be used for conscanning and downlink at this relatively small spacecraft-earth range, no problem is encountered in maintaining downlink or the ability to conscan during this transit. A third method of getting to the other side of the sun is by use of the fixed angle pulses. A combination of two to five pulses using mixed angles and pulse widths will achieve the desired result. If the first is a large one, and the remainder are for trim, the sun may not be lost at all.

Figure 4.6-3: As above, it will be assumed that a conscan has been performed around launch plus 312 days to obtain an earth-pointing attitude. It can be observed from the figure that for a period of about 1.5 days before and after alignment, both up and downlink communication is severely degraded due to solar corona attentuation as expected during this occultation. Effectively, this means that all precession commands must be transmitted before or after this interval; this complicates somewhat the sequence of maneuvers through alignment.

After approximate earth-pointing has been achieved with a conscan, the spin axis is precessed in a plane at an angle of about -40° to the ecliptic; this is performed prior to the time of severe corona attenuation. The magnitude of the angle precessed through is about two degrees, so that the spin axis is pointing outside the sun sensor deadzone, with the earth still in the hi-gain field-of-view. No maneuvers are commanded until the period of corona attenuation has passed. When this has occurred (at about launch plus 314D), an open loop maneuver of about 3.5° at zero degrees to the ecliptic is commanded. Since the sun pulse is the roll reference source, this command will cause the spin axis to precess through the sun and at some angle to the ecliptic; the magnitude of the angle precessed through (shown on Figure 4.6-3) represents the projection of the spin axis on the ecliptic. An angle of 3.5° will be sufficient to keep the spin axis out of the SSA deadzone and still keep the earth in the downlink field-of-view. The final maneuver is a conscan about seven days after alignment.

For all the open loop maneuvers above either the PSE, the fixed angle logic, and the stored command provision of the CDU may be used. The spin period sector generator must be spin averaging for maneuvers through the sun to provide an attitude reference as the spin axis moves through the sun sensor deadzone.

Figure 4.6-4: The final syzygy is treated similarly to that of Figure 4.6-2 above; an open loop maneuver of about 3.5° is commanded to the other side of the sun using either the PSE or the fixed angle logic and the stored command programmer as described.

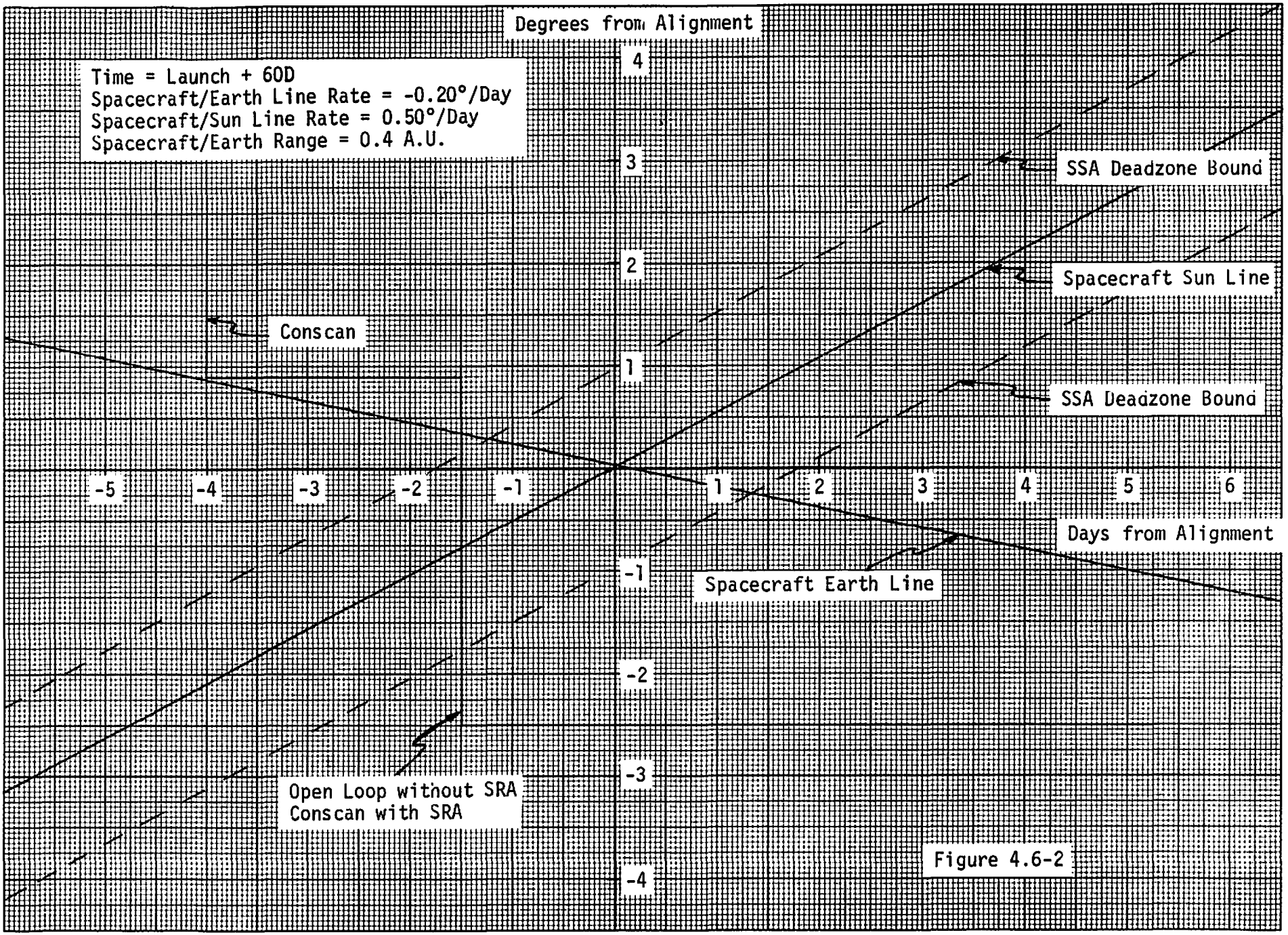

Degrees from Alignment
Time = Launch + 60D
Spacecraft/Earth Line Rate = -0.20°/Day
Spacecraft/Sun Line Rate = 0.50°/Day
Spacecraft/Earth Range = 0.4 A.U.
4
3
2
1
-1
-2
-3
-4
-5
-4
-3
-2
-1
1
2
3
4
5
6
SSA Deadzone Bound
Spacecraft Sun Line
Conscan
SSA Deadzone Bound
Days from Alignment
Spacecraft Earth Line
Open Loop without SRA
Conscan with SRA

Figure 4.6-2

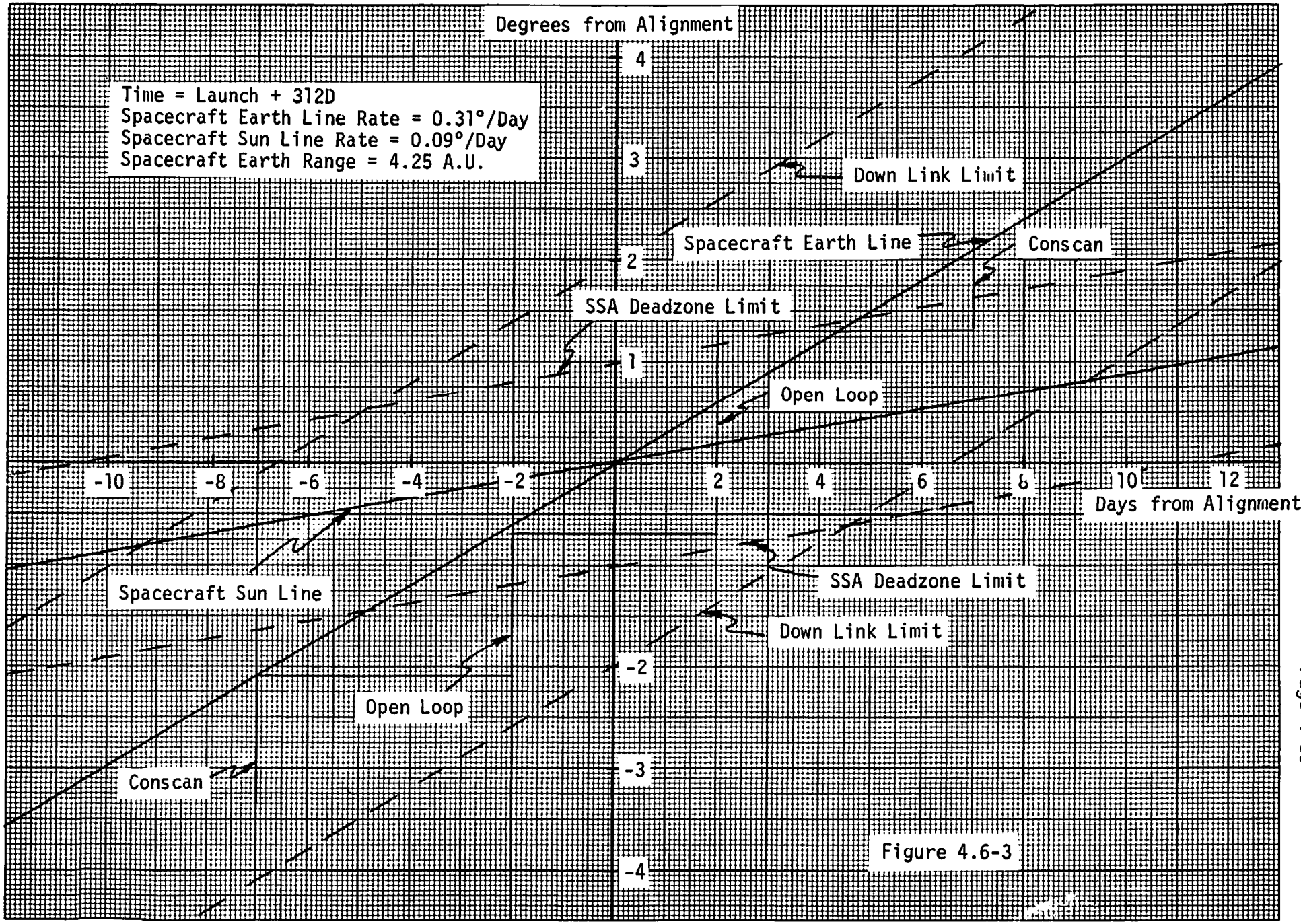

Degrees from Alignment
Time = Launch + 312D
Spacecraft Earth Line Rate = 0.31°/Day
Spacecraft Sun Line Rate = 0.09°/Day
Spacecraft Earth Range = 4.25 A.U.
4
3
2
1
-2
-3
-4
-10
-8
-6
-4
-2
2
4
6
8
10
12
Days from Alignment
Down Link Limit
Spacecraft Earth Line
Conscan
SSA Deadzone Limit
Open Loop
Spacecraft Sun Line
SSA Deadzone Limit
Down Link Limit
Open Loop
Conscan

Figure 4.6-3

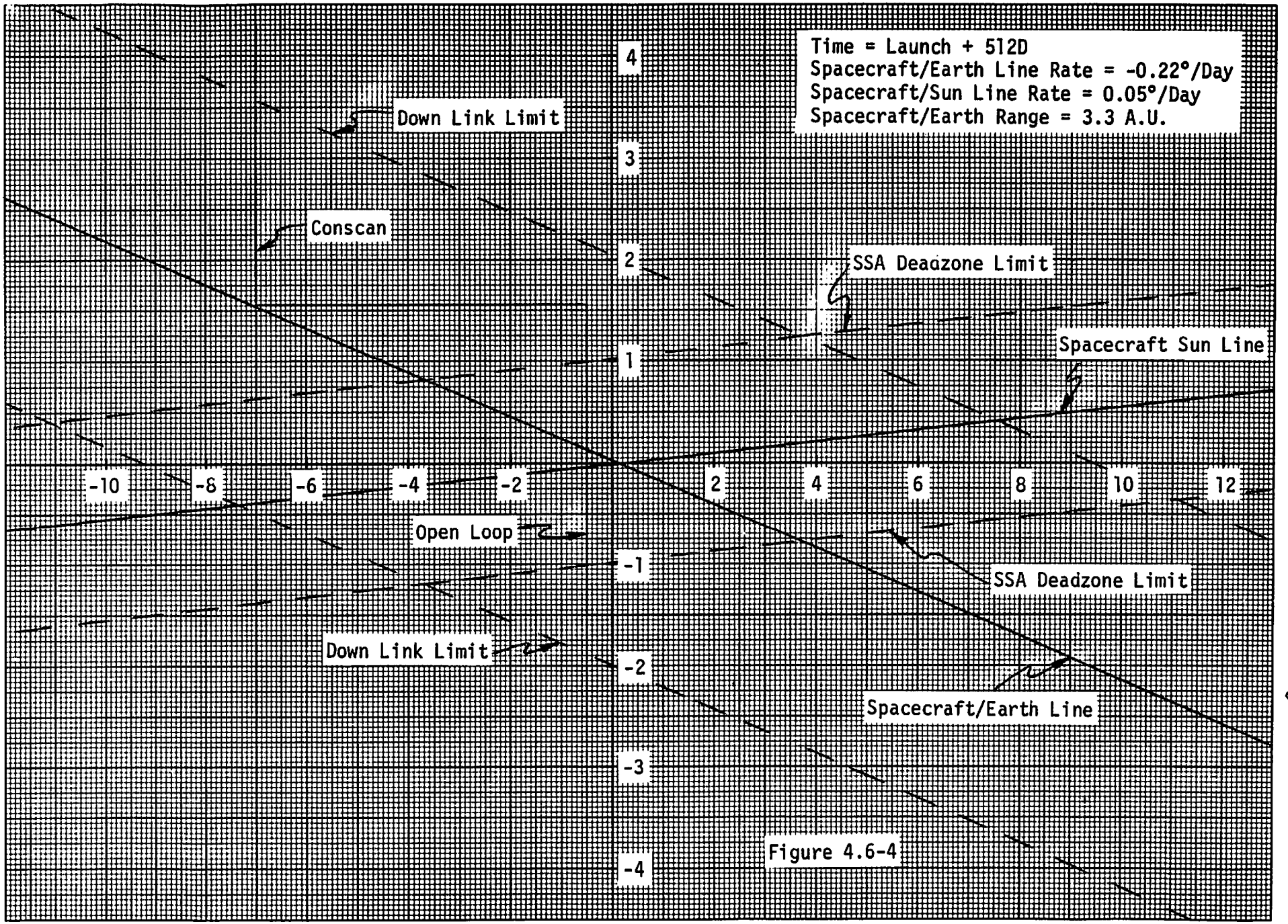

Time = Launch + 512D
Spacecraft/Earth Line Rate = -0.22°/Day
Spacecraft/Sun Line Rate = 0.05°/Day
Spacecraft/Earth Range = 3.3 A.U.
Down Link Limit
Conscan
SSA Deadzone Limit
Spacecraft Sun Line
Open Loop
SSA Deadzone Limit
Down Link Limit
Spacecraft/Earth Line
4
3
2
1
-1
-2
-3
-4
-10
-8
-6
-4
-2
2
4
6
8
10
12

Figure 4.6-4

4.7 SPECIFIC PREPARATION AND ENCOUNTER OPERATIONS

The operations to be performed prior to encounter include spin rate trimming, earth pointing trim, star lock maintenance, power margin review, and general exercising of spacecraft features (such as the DSU modes) which are critical for optimum encounter performance. The exercising of specific spacecraft capabilities will probably be performed within each subsystem and are discussed within the subsystem descriptions as applicable.

4.7.1 Spin Rate Trim

The desired spin rate at encounter is determined primarily by the requirements of the experiments. It is also important to maintain SRA lock on Canopus for an accurate roll reference, and spin speed control is needed to optimize SRA performance as discussed in Section 4.7.3. It is assumed that the desired spin rate at encounter is near 4.8 rpm. The spin rate selected should anticipate spin rate perturbations due to Jupiter. A slowing of about 0.2% (nominally 0.0096 rpm at 4.8 rpm ω_S for the entire encounter period) of the spin rate is expected because of eddy currents induced in the spacecraft by the Jovian magnetic field (see Section 4.6.2).

Since spin thruster firing produces a precession while precession thrusters are not strongly coupled to spin, the spin control maneuver should be performed before the earth pointing trim. A single pulse from the spin control thrusters will produce a nominal spin change of about 0.0032 rpm (with a precession of about 0.01°) using the 31.25 msec pulsewidth and assuming a full propellant tank. At 4.8 rpm, this pulse will give a period change of about 0.0083 sec. Somewhat finer resolution will probably occur at encounter since the fuel will most likely be somewhat depleted. If an even number of spin control pulses is to be used (the preferred method) precession may be cancelled by timing the pulses to be separated by 180° of spacecraft rotation. Since all spin control pulses are fired by real time commands, these pulses are timed on the ground.

4.7.2 Earth Pointing Trim

The earth pointing trim should be performed primarily to obtain accuracy in knowledge of spin axis pointing and to maximize communication capability. The trim should take into account expected drift due to solar torques and Jupiter perturbations during encounter. The trim should be performed at a time which permits sufficient wobble damping before use of the IPP experiment. The drift expected from solar torques is only about 0.0015 deg/day while magnetic moments will probably cause a drift of about 0.05° in the ecliptic plane (Section 4.6).

With a final earth pointing trim performed with the high-gain antenna conscan, an accuracy of about 0.3° can be expected. This pointing can be improved upon by open loop precession either via the PSE, the fixed angle pulses, or by real time pulses.

The wobble expected after conscan is nominally equal to the precession step size, 0.06°. The expected wobble damper time constant is about 8 min, thus, 16 minutes (2 time constants) should be sufficient to reduce wobble well below the 0.01° wobble tolerance.

4.7.3 Star Lock Maintenance

To insure lock on Canopus during encounter, the expected encounter trajectory (determined from tracking data) must be reviewed and correlated with the expected SRA and light shade performance.

The SRA saturation recovery circuit will normally blank out Jupiter completely, however, it is possible for pulses to occur on the trailing edge if the noise background is high. Reflected light from Jupiter in the SRA stray light shade could cause extra pulses and even loss of Canopus lock. The SRA stray light shade (discussed in Section 3.3.1.7) is designed to eliminate effects of reflected light and to minimize the regions where Jupiter and its satellites can look directly into (and trigger) the SRA. In performing this task, the shade is aided by the SRA recovery circuit and the time gate. The recovery circuit prevents the SRA electronics from remaining saturated more than 0.3 sec (8.7° at 4.8 rpm) if Jupiter or some other bright object is detected. Since Jupiter is expected to enter the SRA FOV before Canopus, the recovery time can be critical for some orbits if the circuit is to recover in time to permit Canopus detection. Further, the circuit must be desaturated before the opening of the time gate, or else the trailing edge of Jupiter could be detected if a trailing edge pulse occurs. This would not necessarily cause loss of lock on Canopus, but would certainly cause phase error in the roll index pulses from the SPSG.

Figure 4.7-1 shows how the direction to Jupiter and the apparent size of the planet and its sunlit portion vary during the encounter passage of the spacecraft past the planet. For the purpose of the Stellar Reference Assembly, the effect of Jupiter is felt only while the sensor is time-gated to be responsive to Canopus; therefore, locations are shown only at the time that Canopus is in the SRA field of view. The grid system of the figure is referenced to the SRA optical axis and field of view, and is shown as spacecraft body-fixed spherical polar coordinates* with the pole in the +Z (earth-pointing) direction, and the prime meridian coinciding with the sensor field of view.

The figure shows Jupiter locations corresponding to 30° intervals in the Jupiter-spacecraft direction in the encounter orbit plane, in the 300° variation from approach to departure. The interplanetary trajectory used in this example is defined by:

Launch date	March 4, 1972
Encounter date	December 1, 1973

It corresponds to approximately the earliest possible arrival date from the 1972 launch opportunity. The encounter trajectory is defined by:

$$R_{periapsis}/R_{Jupiter} = 3.0$$

$\theta = 14°$ (targeting angle, measured clockwise from the +T axis)**

---

* In a stereographic projection onto the plane of the figure.
** Defined in PC 292.

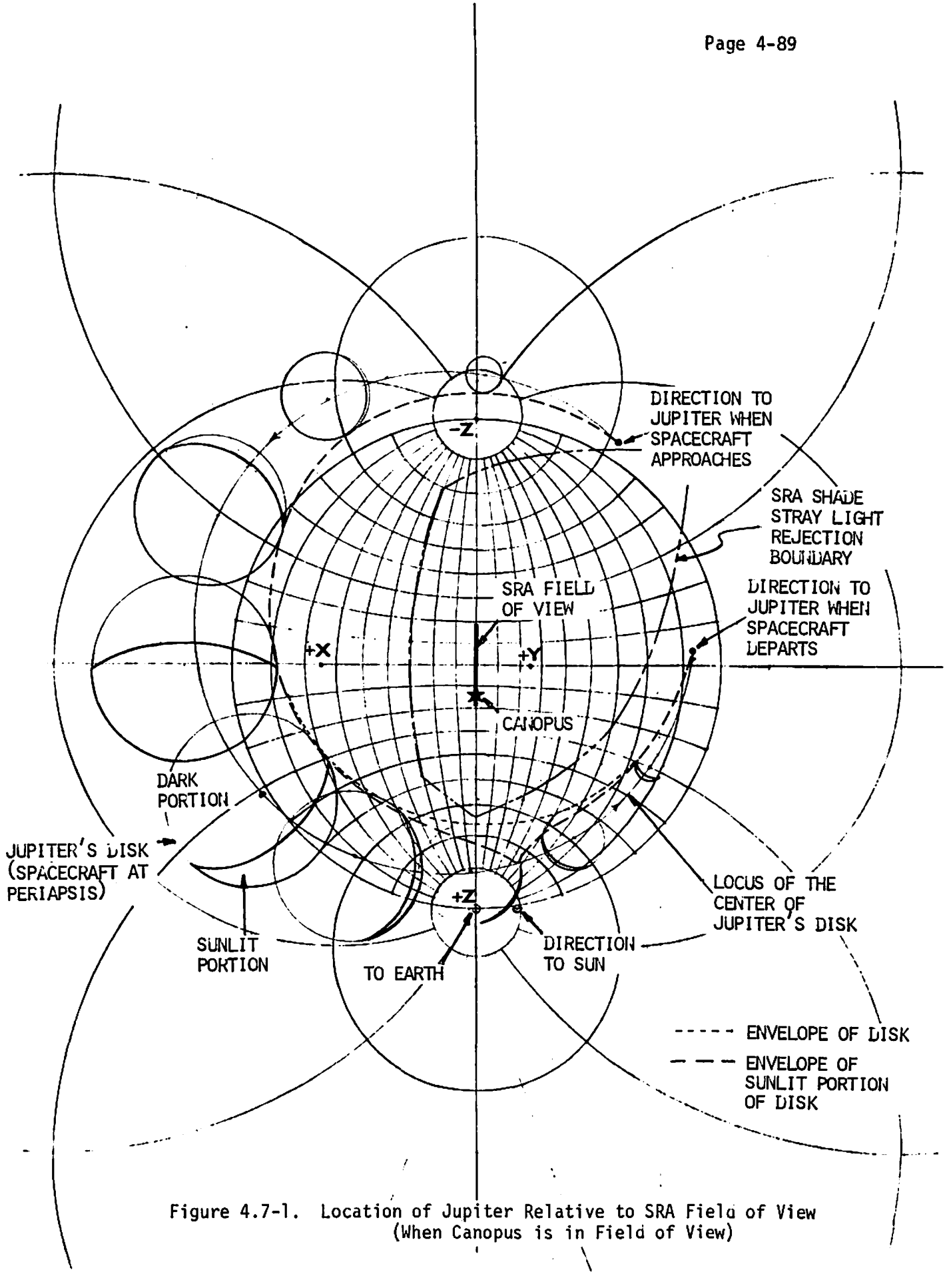


Figure 4.7-1. Location of Jupiter Relative to SRA Field of View (When Canopus is in Field of View)

This is the nominal encounter trajectory for the launch date used in the example.

In addition to the location and apparent size of Jupiter, the inner (side closer to the SRA optical axis) envelopes of the disk of Jupiter and of the sunlit portion of the disk are shown. It is the latter envelope for which a margin against interference is properly established by the SRAs light shield and recovery.

The time gate is quantized to 0.25 seconds. Since it is desired to maintain a minimum time gate opening (for the reasons given above) some spin control may be necessary. Slowing of the spacecraft (as expected from magnetic moments) would open the time gate at an angle closer to Jupiter, however, the slowing would aid recovery circuit performance. The time gate opening requires a minimum opening of 0.131 sec, consisting of the 0.125 sec hold time and 0.006 sec for SRA jitter and delay quantization (.172° at 4.8 rpm). The 1/4 second time gate quantization implies that the time gate should be set for about 12.25 seconds. With the minimum time gate opening, the period of the spacecraft must be at least 12.381 seconds (4.8461 rpm) to optimize time gate performance. The time gate can, however open 1/4 sec before the 12.25 sec value set (described in Section 3.3.1.5.5) giving a worst case time gate opening of 0.381 seconds. The time gate opening will therefore vary between 3.77° and 10.97° if the spin rate is optimized (4.8461 rpm) for time gate operation. At 4.8 rpm the time gate opening would range between 7.2° and 14.4°.

4.7.4 Power Margin Review

The electrical power subsystem can be evaluated by its performance in three of its capabilities.

1) Control and regulate RTG power output

2) Support the spacecraft load

3) Perform battery load sharing

With 4 RTGs operating, an excess of power over load requirement is expected for several years after encounter. With 3 RTGs operating, and with the expected RTG degradation, the power required in the commanding mode should be about equal to that available. In situations where RTG power does not exceed the load, DC bus regulation will be accomplished by battery discharge.

In view of the large number of conditions that can occur for the power subsystem at encounter, careful review of encounter power requirements must be made to allocate the remaining RTG-battery capabilities to maximize the return from the scientific instruments. The operation of the power subsystem is described in Section 3.7.

SECTION 4 NOTATION

A	Amplitude of Conscan Signal
ACS	Attitude Control Subsystem
AGC	Automatic Gain Control
A.U.	Astronomical Unit 150 x 10^6 kilometers
B	Magnetic Field Strength, Gauss
bps	Bits per second
c	Correction factor for pulse centroid calibration
CDU	Command Distribution Unit
CEA	Control Electronics Assembly
CKT	Circuit
CMD	Command
CSP	Conscan Signal Processor
dB	Decibels
dBm	Milliwatts expressed in decibels
DC	Direct Current
DEC	Declination
DSS	Deep space stations
DSL	Duration and Steering Logic (part of CEA)
DTU	Digital Telemetry Unit
EGSE	Electronic Ground Support Equipment
ERP	Effective Radiated Power
E_T	Efficiency of VPT thrusters
F	Thrust level, lbs
FF	Flip-Flop
FOV	Field of View
GND	Ground
H	Angular Momentum
HG, HGA	High Gain Antenna
Hz	Hertz, cycles per second
I	Inertia
IPP	Imaging Photopolarimeter Instrument
IRA	Initial Reference Axis
k	impulse uncertainty calibration correction factor
k	wobble factor based on Inertia, defined on page 4-58
km	kilometers
lb	pounds
m	number of bits in precession magnitude register
m	minutes
MECO	Main Engine Cutoff (Centaur)
MG	Medium Gain Antenna
ms	milliseconds
m/sec	meters/second
n	number of bits in angle register (ACS)
ORD	Ordnance
ppr	pulse per revolution
pps	pulse per second
PSE	Program Sorage and Execution Assembly
psi	pounds per square inch
r	number of revolutions between pulses
RA	Right Ascension
RF	Radio Frequency
R_J	Jovian Radii

SECTION 4 NOTATION

rpm	revolutions per minute
RTG	Radioisotope Thermoelectric Generator
s	seconds
S/C	Spacecraft
SCT	Spin Control Thrusters
SEQ	Sequence
SPSG	Spin Period Sector Generator
SRA	Stellar Reference Assembly
SSA	Sun Sensor Assembly
Sun Rt Sun Left	For an observer on the spacecraft looking at earth, and Canopus "down," sun right is defined as being on the observers right, and sun left is defined as on the observers left.
SW	Switch
t	time
TLM	Telemetry
TWT	Travelling Wave Tube Amplifier
Type I Type II	Motion where spin axis is precessed in orthogonal planes; defined in Section 5.4.2.2
V	Volts
VDC	Volts Direct Current
VPT	Velocity Precession Thruster
x_b, y_b, z_b	body axes

α	Rhumb angle measured from sun - spin axis plane to direction of desired precession (page 4-10)
α_e	executed α direction
α_R	angle α entered in precession angle register
γ	sun-spacecraft-spin axis angle (sun look angle)
$\delta\alpha$	error in α
δ_c	angle between leading edge of pulse command and pulse centroid
$\delta\psi$	error in ψ
ε	spin axis pointing error from S/C - Earth line
θ	pointing angle from conscan telemetry (defined in Section 3.6)
θ	Sun-Spacecraft-Earth angle
λ	longitude - angle from ecliptic plane measured about the sun line as given by right hand rule.
τ	pulse duration, seconds
ϕ	90° - θ, latitude
ψ	length of rhumb line path
ω	spin rate
ω_s	spin rate
ℓ	moment arm of thrusters
$\Delta\alpha$	α quantization
ΔV/SCT	Method of performing ΔV using SCT thrusters
$\Delta\psi$	precession step size

5. SPACECRAFT OPERATING MODES (BACKUP)

5.1 INITIAL OPERATIONS

The initial operations are those that encompass all actions of the spacecraft that are controlled by the on-board sequencer in the CDU. These events are described in Section 4.1. All of the sequencer events can be initiated by ground commands if the sequencer should fail. These commands are summarized in the following table. The start sequencer event is normally initiated by the separation switch closure.

Sequencer Initiated Event	Event Symbol	Event Time Reference (sec)	Commands
Start Sequencer Sequencer Enable Sequencer Start	t_o	0	 SEQ3 SEQ1
Start Despin	t'_o	64	SPN1
Arm Ordinance		64	ORD2
Thruster Verification	t_1	112	--
Initiate RTG Deployment RTG Deploy Redundant	t_2	360	DEP1 DEP2
Initiate Magnetometer Boom Deployment	t_3	1024	DEP3
Redundant Boom Deploy			DEP4
Start Initial Orientation maneuver	t_4	1408	ACX1
Stop Sequencer	t_5	any time after t_4	SEQ2

If these events are initiated by ground command, the times shown in the table should be allowed in each case except where performance or execution can be positively verified by telemetry.

Failure of the redundant separation switches may mean that the interstage structure is still attached. This can be verified during despin by noting the spin change rate, and assuming normal thruster operation. If the interstage structure is attached, the initial orientation maneuver will have to be re-programmed to account for the change in vehicle inertia.

The despin maneuver is terminated by the despin sensors, with no backup commands provided. Out of tolerance behavior of these switches should be compensated by ground spin control before initiation of the RTG and magnetometer deployment (see Section 3.1 for the tolerance on the spin rate for the deployment). If the initial orientation maneuver program should fail, acquisition can be performed using backup precession control capabilities discussed in Section 5.4.

5.2 EARTH TRACKING BACK-UP MODES

This section discusses alternatives to normal conscan closed loop spin axis pointing. The approach is first to determine spin axis pointing direction, second, transmit commands which precess the spin axis to an earth pointing attitude.

5.2.1 Determination of Spin Axis Attitude

Some methods for determining spin axis pointing are the following:

1) The antenna feed is offset and telemetered Conscan phase and amplitude from the spacecraft is used to determine spin axis pointing. Even if the total conscan function fails, it is possible that the telemetry of phase and amplitude of the pointing error will remain. This telemetry capability is described in Section 3.6.2.9.3.

2) The antenna feed is offset on the spacecraft and the ground conscan signal is recorded as a function of time. The phase of the signal relative to the roll pulse may be determined from the telemetered time of occurrence of the roll reference pulse, the spacecraft-earth separation distance (determined from earth tracking) and the spacecraft spin rate.

   (Note: The accuracy with which spin axis pointing may be determined using either of the above two methods degrades as the sun look angle decreases, if the Sun Sensor Assembly is being used as the roll reference source. This is because the deviation between the time of occurrence of the sun pulse and the time that the spacecraft reference line ascends through the plane containing the spacecraft and parallel to the ecliptic is increasingly sensitive to spin axis pointing uncertainty as the sun look angle decreases.)

3) Dead reckoning, an extrapolation of spin axis pointing relative to the spacecraft-earth line, may be made using the time of last Conscan (assuming earth pointing was achieved at that time), rate of change of orientation in inertial space of the spacecraft-earth line, and the magnitude and direction of solar torques acting on the spacecraft. This has the advantage of determining the pointing direction with the aid of neither uplink nor downlink.

   Drift rate caused by solar torque is described in Section 4.4.2. The spacecraft-earth line drift rate caused by earth motion is presented in Figure 5.2-1 for a representative March 4, 1972 launch date. Using this information on the two expected relative motions between the earth and the spacecraft, plots of the momentum vector (spin axis) location can be plotted to give predictions of spacecraft pointing.

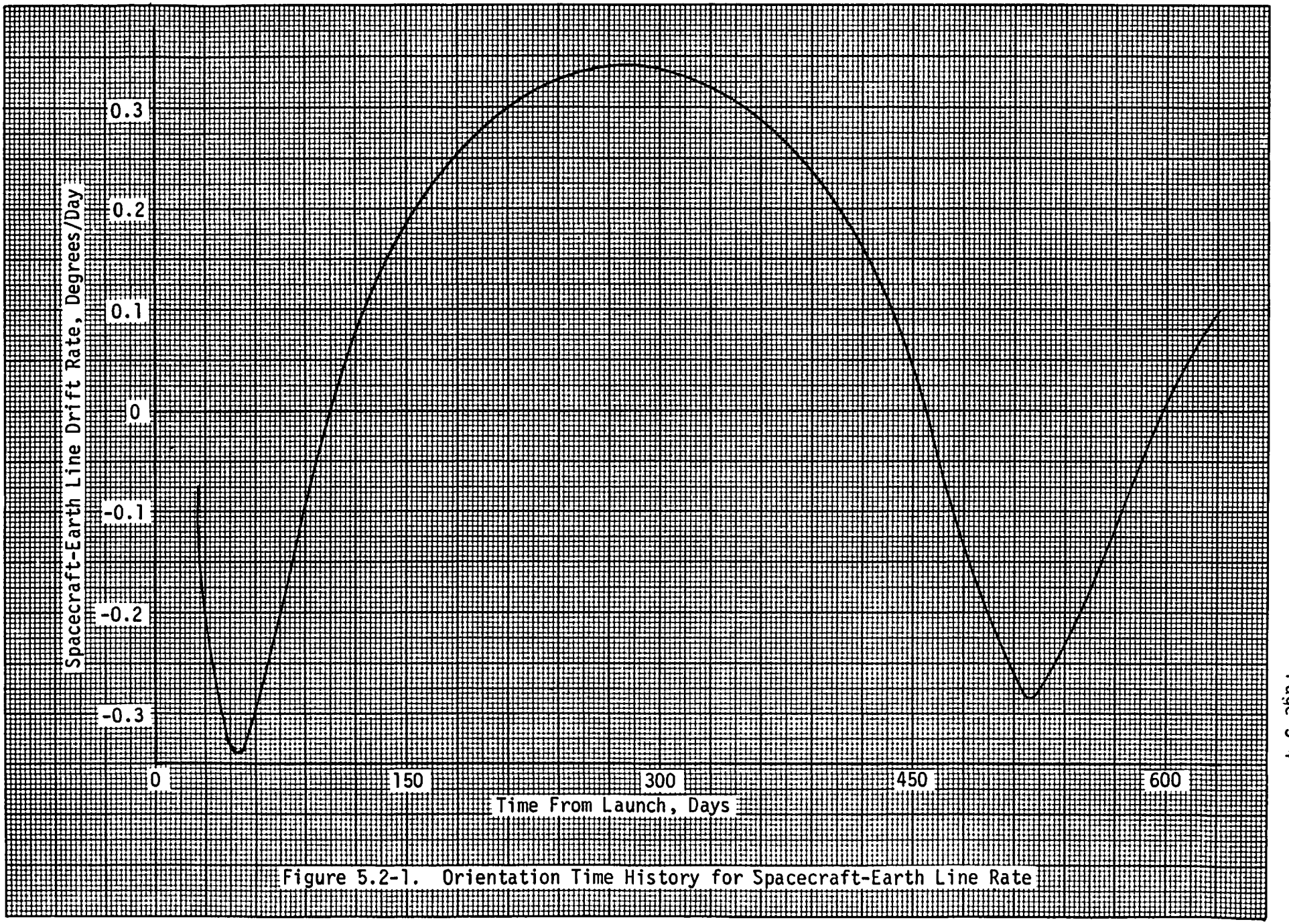


Figure 5.2-1. Orientation Time History for Spacecraft-Earth Line Rate

4) Pointing direction can be determined by trimming the spacecraft pointing to increase the downlink signal strength. This method can be systematized by precessing the spacecraft in each of two perpendicular directions (by about one degree for the medium gain antenna) and comparing the signal strengths with the downlink pattern. A single measurement of this type could yield both the amplitude and direction of the pointing error. Various search techniques can be used by making single precession steps. Measurements at three neighboring points on the pattern would enable the determination of the gradient of the signal strength to determine the spin axis direction. Precession in this direction would then be made until a maximum in signal strength is obtained. The search can then be repeated if additional improvement is desired.

5.2.2 Precession Modes for Conscan

After spin axis direction has been determined, precession to earth pointing may be achieved in one of the following ways:

1) Use of a PSE stored program to precess the spin axis along a rhumb-line toward earth (see Section 4.4).

2) Issuance of precession commands in real time: Commands are executed as received, so that transmission must be properly timed on the ground. Knowledge of spacecraft roll attitude is required.

3) "Quadrature Precession" using pulses initiated at roll angles of 0°, 90°, 180°, or 270°.

Additional explanation of these precession modes is given in Sections 4.4 and 5.4.

5.2.3 Reestablishing Communication After ΔV Maneuvers

A critical situation could arise if the stored program for a precession or ΔV maneuver were only partially executed. This situation could result in the spacecraft having an unknown orientation. However, earth-based transmitter power levels required to successfully transmit precession commands to the spacecraft can give information on pointing direction by comparison with the uplink antenna pattern power levels. The effects of medium gain antenna offset and antenna pattern peaks and nulls in the direction perpendicular to the spacecraft z-axis should be included in this determination.

5.3 STAR ACQUISITION BACKUP

5.3.1 Acquisition Without Sun Pulse

Without a sun reference, and without a star lock, the SPSG in the DTU would not give reliable 1 ppr and 8 ppr data. Without these references, the acquisition cannot use the star location logic of Sectors 1 and 8, nor the star count logic. The angle gate loses its acquisition capability since it also requires the DTU reference pulses and the octant generator. Without these capabilities, the time gate must be used for star acquisition.

When the time gate is used for locking onto a star, no direct information will be obtained in telemetry to indicate indexing of the star reference should it occur. If this indexing happens, then the spin period after this occurrence will change accordingly. If the SPSG is in the spin averaging mode, such a change may not be noticed on the ground. During those times when certain objects enter the SRA field of view (deterministic objects having a high probability of detection), periodic checks should be made to insure that Canopus is the reference. The "objects" above include planets, large asteroids, and some stars of prime concern which are listed below.

Early in Mission		Late in Mission	
Star	Probability	Star	Probability
Canopus	1	Canopus	1
Achernar	0.68	Agena	0.75
Vega	0.16	Rigel Kentaurus	0.55

The roll angle between Canopus and the sun pulse is well defined throughout the Pioneer mission. Measurement of this angle serves as positive identification of the received blip as belonging to Canopus. Without the sun pulse the operator must rely on the fact that Canopus is the only object in the field of view with a unity probability of detection, or on the fact that the angle between Canopus and the other stars can be measured using the star coincidence logic, or the real time star telemetry capability. Long observation times of the real time telemetry can be used not only to identify each of the stars detected, but also can be used to verify or correct the probability of detection table.

If Canopus is the only star in the FOV with a unity probability of detection,the time gate will eventually move around until Canopus is acquired. If, for example, the time gate has been set to include 1/2 sec less than one revolution and it has been enabled by some pulse other than Canopus', it will remain "locked" onto that pulse until the pulse disappears or the star advance command is used. A typical timing diagram is shown in Figure 5.3-1.

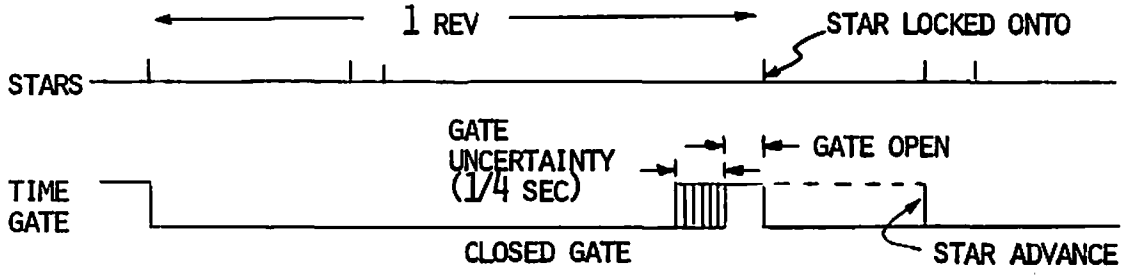


Figure 5.3-1. Star Advance Timing

Since it is the star pulse that closes the gate, the gate will close on the first available pulse that occurs after the gate opens. The dotted line shows what will happen if the original pulse does not reappear. If all the stars but Canopus in the FOV have a probability of detection less than unity the gate will eventually step around until it locks onto Canopus.

These facts suggest a procedure to be followed when attempting to find Canopus without the aid of the sun pulse. That is, merely set the time gate for a little less than one revolution then wait. How long? Given the probabilities of detection given in Table 4.3-1, on day 450 a wait of 43 revolutions (or about 9 minutes) yields a 0.99 probability of the time gate moving to Canopus, based on the stars in the field of view on that day. This assumes the worst case time to start the time gate, that is right after Canopus. Any other time raises the probability of locking on Canopus.

An important problem for the operator is the identification of the star held in lock by the time gate. Assuming that the real time telemetry has been observed before locking onto a star, the roll angles between the stars observed will be known. The corresponding delay times for these angles should be set into the delay and the star coincidence telemetry checked until identification is made. (The star delay capability is zero to 16 seconds with a quantization of 1/256 seconds or 3.906 milliseconds. At 4.8 rpm, this corresponds to an angular delay of 476° with a quantization of 0.1125°. If the delay is set higher than 16 seconds, it recycles to zero.) If the delayed pulse coincides with any other star pulse within 1/16 sec, a flip-flop will be set. The state of this flip-flop is telemetered to the ground, then reset by the trailing edge of telemetry word gate C-430. The angles corresponding to the telemetered delay may be compared with the star map to identify the stars. For example, on day 25, the angle between Canopus and Vega is 187.4. Therefore, if the delay is equivalent to this angle, the star is Canopus. If no other star pulses occur, star coincidence will be achieved when Canopus is delayed to itself--a 360° delay angle.

If the star identified is not Canopus, then the star advance command may then be issued to transfer the lock to next available pulse. A wait of a few minutes will allow the lock to transfer past random stars as discussed previously. Star angle measurements will then be repeated. This procedure will continue until Canopus is identified.

During the nominal 1972 mission considered there are no single stars that enter the field of view with a unity detection probability other than Canopus. However, at about 250-300 days a group of four stars enter the field of view; Acrux, γ-Crux, Agena, and Rigel Kentaurus. Two of them, Acrux and γ-Crux have very low detection probability. At the time that Rigel Kentaurus and Agena both appear in the FOV there is a roll angle of about 0.3° between them. This angle increases to about 1.4° on day 425. When they are very close they might appear as one bright star to the SRA with a unity probability of detection.

During other missions (Pioneer G) Sirius or Betelgeuse could appear in the FOV. Sirius always has a unity probability of detection, Betelgeuse only during its brighter stages. The SRA provides a discrete which is telemetered to the ground for identification of stars of brightness greater than 180% of Canopus. This serves to identify Sirius. Betelgeuse and the combined Rigel Kentaurus and Agena will have to be handled by means of angle measurement and the star advance command.

5.4 PRECESSION BACKUP

5.4.1 Capabilities Remaining with Clock Failure

DTU clocks A and B are critical to the existance of precession capability, both open and closed loop. Table 5.4-1 shows the capabilities associated with each of these clocks and indicates the remaining capability in the event of failure. Normally both clocks are functioning and all capabilities are available. However, it can be seen that the loss of DTU clock A will prevent the operation of the ACS fixed angle precession mode, the ACS star logic and Conscan. The loss of clock B prevents the operation of ACS programmed maneuvers. All other equipment receive these clocks in a manner that does not prevent the execution of the various functions due to the failure of a particular clock.

The objectives considered in the distribution of the DTU clocks were:

1) Preserve the fixed angle capability as a backup to the programmed precession capability.

2) Assure a reasonable backup for conscan.

3) Provide a reasonable alternative for performing maneuvers around syzygy of the spacecraft, earth and sun.

The first objective is accomplished by connecting the programmed precession logic and the fixed angle precession logic to opposite clocks. The second objective is accomplished by assigning the conscan and the programmed precession logic to opposite clocks.

The only variable remaining is the star logic. It is associated with the fixed angle logic for two reasons:

1) It allows the fixed angle capability to be variable by adjusting the star delay timing.

2) It provides a more precise knowledge of orientation to counterbalance the less precise precession capability when using the CDU memory and large precession pulses. This provides a reasonable alternative for syzygy maneuvers within the constraints.

5.4.2 Precession Backup Modes

In the event of failure of the PSE, Conscan or both, precession of the vehicle may be performed by either of two backup modes, real time precession pulses and fixed angle "quadrature pulses" depending on the failure mode.

5.4.2.1 Real Time Precession. Real time precession, ΔV or spin pulses must be timed from the ground since no on-board roll reference is supplied. These pulses will occur immediately upon receipt of the command. The roll attitude must be known to establish the direction of precession

Table 5.4-1. DTU Clock A/B Allocation

Unit	Clock A	Clock B
CEA/PSE Programmed Precession Logic	0*	X
CEA/SPC Fixed Angle Precession Logic	X	0
CEA/SPC Star Logic	X	0
CSP Conscan	X	0
**CEA/DSL A	X	X
**CEA/DSL B	X	X
**CDU Command Memory Event Sequencer	X	X
DDU A	X	0
DDU B	0	X
**SPSG A	X	0
**SPSG B	0	X
**Experiments	X	X

* Clock A is connected to the PSE, however its failure will result in loss of redundant registers in the PSE, but not in loss of any PSE function.

** Selected by command (both clocks always on in regard to other boxes)

X = Connected

0 = Not Connected

(see Section 4.5). The operation of real time pulses requires that one DSL be on and that the clock(s) driving the pulse generators must be operating (see Section 3.3.1.4.2). The commands necessary are for pulse length selection, PUL 1-5; thruster selection, PRE 1-2, VEL 1-2, SPN 3-4; and pulse execution, PRE 7, VEL 3, or SPN 2.

5.4.2.2 Fixed Angle "Quadrature" Precession. The quadrature precession pulses are initiated at roll angles of 0°, 90°, 180°, or 270° as selected by command, PRE 3-6. An on-board roll reference (SRA, SSA) must be operating. In addition, one DSL must be on, and the octant generator must be operating. The commands sent to execute quadrature pulses are first, select the precession pair, PRE 1-2; second, select the pulse length, PUL 1-5; and last send the desired roll angle execute command, PRE 3-6. These pulses will actually occur in the direction of the centroid of the pulse issued which depends on the pulse width (Section 3.3.3.3, page 3.3-57). Precession using these pulses is classified as Type I/Type II motion, where the spin axis is precessed in orthogonal planes. Type I motion lies in a plane close to the reference plane, but separated from it by the spacecraft roll angle from the thruster pulse initiation to the pulse centroid. Type II motion lies in the plane, containing the instantaneous spin axis, which is normal to the Type I plane. If the SSA is used for the reference, the reference plane is that plane defined by the sun and the spin axis. With the SRA (Canopus as the reference) then the reference plane is defined by the plane formed by Canopus and the spin axis. However, since the Canopus pulse can be delayed via the star delay logic, the reference plane can be controlled to be oriented at any desired angle about the spin axis. Type I/ Type II precession can thus be controlled to occur in any desired direction.

Without the star delay logic, complete control of the precession direction can be obtained by mixing angles and step sizes of the fixed angle logic, however wobble amplitude may become excessive if this is not done judiciously. Wobble can be minimized if the precession is performed in one plane, with precession pulses commanded steadily, once each 3rd revolution of the spacecraft.

Since the roll attitude of the quadrature pulse is predetermined, the stored command programmer can be used to issue up to five of those pulses, thus given the capability of a five pulse stored program. Using the largest pulse step size, this precession can be as much as 25.6° with a full tank (see thruster nominal performance summary, Section 3.3.3.3, page 3.3-57).

5.4.2.3 Spin Control Precession. The execution of real time spin control pulses are accompanied by a small precession. This precession may be cancelled by firing the pulses $(2n + 1)\pi$ degrees apart (n is a positive integer or zero) or the precession may be used to improve spacecraft pointing by properly timing the spin pulse(s).

5.5 DELTA VELOCITY (ΔV) MANEUVERS

In the event that a VPT thruster should fail, the ΔV maneuver can be performed using 2 alternatives.

1) Use the opposite VPT pair for ΔV. This may require a maximum precession of as much as 180° depending on the direction of the desired ΔV and on which thruster has failed. If such a large precession is required, the errors in the performance of the ΔV will be correspondingly increased. The dominant error source is the 4.8% (3σ estimate) error in precession impulse uncertainty which implies an error in the direction of the ΔV which is 4.8% of the precession magnitude required.

2) The ΔV can be performed using a single thruster. If this is done, then an error in precession will occur when the ΔV is executed. This precession is caused by the attend- and torque applied to the spacecraft when only one ΔV thruster is fired. This torque vector rotates with the spacecraft and will cause the ΔV to be executed in a direction which is about 3.7° from the pointing direction at the start of the ΔV.

The tradeoff between these two methods is based on the accuracy of the ΔV required, on the spin coupling that may occur with a single thruster, and on the increase in time required for the single thruster ΔV. The errors in ΔV using the thruster pair opposite the failed thruster are fully predictable but constitute an uncertainty (random error) in the direction of the ΔV. This uncertainty could result in an error of 0.048 (180) = 8.6°. If a single thruster is used, the error is a deterministic error if the roll attitude of the spacecraft is known at the instant of ΔV initiation. The transverse torque applied during the ΔV will cause the spacecraft to wobble as shown in Figure 5.5-1.

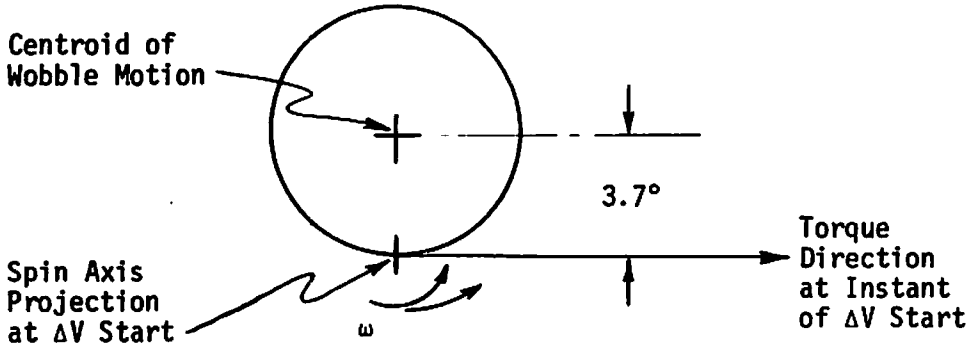


Figure 5.5-1

The centroid of the wobble motion will be the net direction of the ΔV. The angle between the spin axis position at the start of the ΔV and the centroid will be about 3.7° and the location of the centroid will be normal (+90 in a spin direction) to the direction of the torque direction at the start of the ΔV.

A comparison of these two methods indicates that if the precession magnitude for option 1 above is 77°, then the error in execution for the two methods is equal, however, the error for the ΔV using the 1st method is a 3σ random error while for the 2nd method, the error is more or less deterministic. The on orbit calibration of the thrusters may significantly reduce the errors for the 1st method. Any tradeoff between these two options should include these measured performance characteristics.

5.6 ELECTRICAL FAULT PROTECTION OPERATIONS

5.6.1 Overload Protection

If at any time the load exceeds the electrical capability, parts of the load will be dropped in the following order:

a) All scientific instruments.

b) All switched loads except the RF transmitter (TWT and driver).

c) RF transmitter (TWT and driver).

The overloads are sensed by loss of regulation and the load reductions occur automatically with delays between each of the above reductions to provide time to sense the elimination of the overload condition. The overload protection can be inhibited or enabled by ground command.

5.6.2 Overload Implementation

Loss of regulation is sensed by the PCU where an overload signal is generated and sent to the CDU. The CDU, in a timed sequence, then sequentially removes loads. Detailed operation is as follows:

PCU. When the +28 VDC bus decays, at a rate not to exceed 5V/sec to 26 $\pm$ 0.5 VDC and remains below that point for 0.1 to 0.3 seconds, a bi-level signal is generated and sent to the CDU. The signal goes from a level of 1.9 VDC max to a level of 14.6 VDC min. Output impedance is 10k ohm nominal.

CDU. When the CDU receives the overload signal from the PCU the CDU sequentially turns off the following loads with a time interval of 60 ms segments with a tolerance of 60 $\pm$ 30 ms.

All experiments T_0 + 60 ms

ACS Subsystem to standby T_0 + 180 ms

TWTAs T_0 + 300 ms

The load removal sequence terminates immediately when the overload control signal returns to the quiescent level.

5.6.3 Undervoltage CEA-CDU Reset

If the input voltage to either the CEA or CDU decays transitionally to a level which could cause logic changes, a reset sequence is initiated which precludes the possibility of executing erroneous commands or an erroneous pointing maneuver.

CEA Reset Implementation. When a circumstance arises such that the five volt input to the CEA drops below 3.5 $\pm$ 0.5 volts a "Master Reset" signal is developed. When the five volt input recovers and once again passes through the 3.5 volt level the

"Master Reset" signal causes the PSE to be reset and a precession maneuver to be discontinued with all thrusters off.

CDU Reset Implementation. When a circumstance arises such that the five volt power input to the CDU drops below a four volt level a "Load Reset" signal is developed. This signal immediately and simultaneously turns off all the experiments, places the ACS in standby, and turns the TWTAs off.

6.0 NORMAL TELEMETRY RESPONSE

For most telemetered quantities, the "normal" response depends on the operating mode and the mission phase of the spacecraft. Since it is difficult to define normal responses for all mission phases, the time immediately after injection (and before the sequencer starts despin) has arbitrarily been selected for the tabulation of the "normal" response. This tabulation is presented in Table 6-1 on the following pages. In this table a number of items are blank. This implies that the telemetry words in question are either unknown at this time (as is the case with the ACS stored program), or that the words are in the "don't care" category (such as star logic). Only engineering format telemetry is presented.

Table 6-1. Normal Telemetry Responses

Word	Bit	Measurement	Type	Range	Normal Response Before Despin	Remarks
		Electrical Distribution Subsystem				
C-108	1	H V Magnetometer Exp Power Status	B	Off/On	Off	
C-108	2	Plasma Analyzer Exp Power Status	B	Off/On	Off	
C-108	3	Charged Particle Exp Power Status	B	Off/On	Off	
C-108	4	Geiger Tube Tel Exp Power Status	B	Off/On	Off	
C-108	5	Cosmic Ray Exp Power Status	B	Off/On	Off	
C-120	1	Magnetometer Boom Status	B	Release/Stow	Stow	
C-120	2	RTG 1-2 Deploy Status	B	Deploy/Stow	Stow	
C-120	3	RTG 3-4 Deploy Status	B	Deploy/Stow	Stop	
C-120	4	S/C Separation Status	B	Sep/Attach	Sep	
C-124	1	Trapped Radiation Exp Power Status	B	Off/On	Off	
C-124	2	Ultraviolet Photo Exp Power Status	B	Off/On	Off	
C-124	3	Imaging Photo Polar Exp Power Status	B	Off/On	Off	
C-124	4	Infrared Radiometer Exp Power Status	B	Off/On	Off	
C-124	5	Asteoroid/Meteoroid Exp Power Status	B	Off/On	Off	
C-124	6	Meteoroid Detector Exp Power Status	B	Off/On	Off	
C-128	5	Arm/Safe Relay Status Prime	B	Safe/Arm	Arm	
C-128	6	Arm/Safe Relay Status Rednt	B	Safe/Arm	Arm	
C-132	2	Ordnance Safe/Arm, RTG 1/2	B	Safe/Arm	Arm	
C-132	3	Ordnance Safe/Arm, RTG 3/4	B	Safe/Arm	Arm	
C-132	4	Ordnance Safe/Arm, RTG 1/2 Redun	B	Safe/Arm	Arm	
C-132	5	Ordnance Safe/Arm, RTG 3/4 Redun	B	Safe/Arm	Arm	
C-132	6	Sequencer Status	B	Disable/Enable	Enabled	
C-305		Storage Mem Contents Time, 7 Bits	D			
C-306		Storage Mem Contents Cmd, 8 Bits	D			

Table 6-1. (continued)

Word	Bit	Measurement	Type	Range	Normal Response Before Despin	Remarks
C-307		Storage Mem Contents Register ID (3)	D			
C-324	1	Command Memory Status	B	Proc/Standby	Proc	
C-324	6	Command Execute	B			
C-332	1	Sequencer Power Status	B	Off/On	On	
C-332	2	Overload Protection Status	B	On/Off	On	
C-332	3	RCVR Reverse Status	B	Rev/Inhibit	Rev	
C-332	4	Cmd Processor Memory Status	B	B On/A On	A On	
C-332	5	Storage Mem Contents DTU ID	B	No/Yes		
C-332	6	CDU +5DC Bus Status	B	Ch B/Ch A		
		Data Handling Subsystem				
C-101		A/D Cal Voltage Low	A	168 MV		
C-102		A/D Cal Voltage Med	A	1.512 Volts		
C-103		A/D Cal Voltage High	A	2.424 Volts		
C-104		Ext S.C. ID	D			
C-112		Roll Attitude	D			
C-116		Roll Attitude	D			
C-120	5	Decoder A Status	B	Off/On		
C-120	6	Decoder B Status	B	Off/On		
C-132	1	Conv Code Gen Status	B	Off/On		
C-405		Spin Period	D			
C-406		Spin Period	D			
C-407		Spin Period	D			
C-408		Phase Error	D			

Table 6-1. (continued)

Word	Bit	Measurement	Type	Range	Normal Response Before Despin	Remarks
C-417	1	Roll Reference	B	0/180 Deg	0°	Conditions required for earth acquisition maneuver
C-417	2	Spin Averaging Mode	B	No/Yes	No	
C-417	3	ACS Operation	B	No/Yes	Yes	
		Thermal				
C-201		RTG No. 1 Fin Root Temperature	A	160 to 360°F	347°	
C-202		RTG No. 2 Fin Root Temperature	A	160 to 360°F	332°	
C-203		RTG No. 3 Fin Root Temperature	A	160 to 360°F	347°	
C-204		RTG No. 4 Fin Root Temperature	A	160 to 360°F	332°	
C-217		RTG No. 4 Hot Junction Temperature	A	880 to 1200°F	995°	
C-218		RTG No. 3 Hot Junction Temperature	A	880 to 1200°F	980°	
C-219		RTG No. 2 Hot Junction Temperature	A	880 to 1200°F	995°	
C-220		RTG No. 1 Hot Junction Temperature	A	880 to 1200°F	980°	
C-301		Temp Hex Platform -X Axis	A	0 to 140°F	94° ①	99° ② 133° ③
C-302		Temp Hex Platform -X, +Y	A	0 to 140°F	82°	88° 106°
C-304		Temp Hex Platform -X, -Y	A	0 to 140°F	76°	81° 93°
C-318		Temp Hex Platform +X Axis	A	0 to 140°F	88°	93° 110°
C-319		Temp Exp Platform +X Axis	A	-20 to 110°F	82°	82° 89°
C-320		Temp Exp Platform -Y Axis	A	-20 to 110°F	78°	78° 85°

① L--Launch ② +27 min ③ +4 hrs

Table 6-1. (continued)

Word	Bit	Measurement	Type	Range	Normal Response Before Despin	Remarks
		Attitude Control Subsystem				
C-303		SRA Temperature	A	-10 to +95°F	≈87°F	
C-317		SSA Temperature	A	-30 to +194°F	≈140°F	
C-403	1	Precession Pair	D			
C-403	2	Pulse Length Bit No. 1	D		0	0.125 sec pulse width for earth acquisition maneuver
C-403	3	Pulse Length Bit No. 2	D		1	
C-403	4	Pulse Length Bit No. 3	D		1	
C-403	5	Delta V-Pair	D			
C-403	6	Spin Control Direction	D		1	
C-404	1	Time Gate Register Bit No. 1	D			
C-404	2	Time Gate Register Bit No. 2	D			
C-404	3	Time Gate Register Bit No. 3	D			
C-404	4	Time Gate Register Bit No. 4	D			
C-404	5	Time Gate Register Bit No. 5	D			
C-404	6	Delta V SCT Mode	D		0	
C-410	1	Despin On	D		0	
C-410	2	Conscan On	D		0	Clock reset
C-410	3	Clock Select Bit No. 1	D		1	
C-410	4	Clock Select Bit No. 3	D		1	
C-410	5	Star Angle Gate Status	D	45 Deg/360 Deg		
C-410	6	Star Level 180 Deg Canopus	D	Yes/No		
C-411	1	No. 1 Precession Magnitude Reg Bit 1	D			
C-411	2	No. 1 Precession Magnitude Reg Bit 2	D			
C-411	3	No. 1 Precession Magnitude Reg Bit 3	D			
C-411	4	No. 1 Precession Magnitude Reg Bit 4	D			
C-411	5	No. 1 Precession Magnitude Reg Bit 5	D			
C-411	6	No. 1 Precession Magnitude Reg Bit 6	D			

Table 6-1. (continued)

Word	Bit	Measurement	Type	Range	Normal Response Before Despin	Remarks
C-412	1	No. 1 Precession Magnitude Reg Bit 7	D			
C-412	2	No. 1 Precession Magnitude Reg Bit 8	D			
C-412	3	No. 1 Precession Magnitude Reg Bit 9	D			
C-412	4	No. 1 Precession Magnitude Reg Bit 10	D			
C-412	5	No. 1 Precession Magnitude Reg Bit 11	D			
C-412	6	Delta V-Magnitude Reg Bit No. 1	D			
C-413	1	Delta V-Magnitude Reg Bit No. 2	D			
C-413	2	Delta V-Magnitude Reg Bit No. 3	D			
C-413	3	Delta V-Magnitude Reg Bit No. 4	D			
C-413	4	Delta V-Magnitude Reg Bit No. 5	D			
C-413	5	Delta V-Magnitude Reg Bit No. 6	D			
C-413	6	Delta V-Magnitude Reg Bit No. 7	D			
C-414	1	Delta V-Magnitude Reg Bit No. 8	D			
C-414	2	Delta V-Magnitude Reg Bit No. 9	D			
C-414	3	Delta V-Magnitude Reg Bit No. 10				
C-414	4	Delta V-Magnitude Reg Bit No. 11	D			
C-414	5	Delta V-Magnitude Reg Bit No. 12	D			
C-414	6	Delta V-Magnitude Reg Bit No. 13	D			
C-415	1	No. 2 Precession Magnitude Reg Bit 1	D			
C-415	2	No. 2 Precession Magnitude Reg Bit 2	D			
C-415	3	No. 2 Precession Magnitude Reg Bit 3	D			
C-415	4	No. 2 Precession Magnitude Reg Bit 4	D			
C-415	5	No. 2 Precession Magnitude Reg Bit 5	D			
C-415	6	No. 2 Precession Magnitude Reg Bit 6	D			
C-416	1	No. 2 Precession Magnitude Reg Bit 7	D			
C-416	2	No. 2 Precession Magnitude Reg Bit 8	D			
C-416	3	No. 2 Precession Magnitude Reg Bit 9	D			
C-416	4	No. 2 Precession Magnitude Reg Bit 10	D			
C-416	5	No. 2 Precession Magnitude Reg Bit 11	D			

Table 6-1. (continued)

Word	Bit	Measurement	Type	Range	Normal Response Before Despin	Remarks
C-416	6	Star Coincidence Indication	D	No/Yes		
C-419	1	Star Delay Bit No. 1	D			
C-419	2	Star Delay Bit No. 2	D			
C-419	3	Star Delay Bit No. 3	D			
C-419	4	Star Delay Bit No. 4	D			
C-419	5	Star Delay Bit No. 5	D			
C-419	6	Star Delay Bit No. 6	D			
C-420	1	Star Delay Bit No. 7	D			
C-420	2	Star Delay Bit No. 8	D			
C-420	3	Star Delay Bit No. 9	D			
C-420	4	Star Delay Bit No. 10	D			
C-420	5	Star Delay Bit No. 11	D			
C-420	6	Star Delay Bit No. 12	D			
C-421	1	Star Count Bit No. 1	D			
C-421	2	Star Count Bit No. 2	D			
C-421	3	Star Count Bit No. 3	D			
C-421	4	Power Status Bit No. 1	D		1	DSL-A and PSE on for
C-421	5	Power Status Bit No. 2	D		0	stored program execution
C-421	6	Power Status Bit No. 3	D		1	(earth acquisition)
C-422	1	Spacing Bits	D	0		
C-422	2	Spacing Bits	D	0		
C-422	3	Spacing Bits	D	0		
C-422	4	Spacing Bits	D	0		
C-422	5	Spacing Bits	D	0		
C-422	6	No. 2 Prec Rdnt Mag Reg Bit No. 1	D			

Table 6-1. (continued)

Word	Bit	Measurement	Type	Range	Normal Response Before Despin	Remarks
C-423	1	No. 2 Prec Rdnt Mag Reg Bit No. 2	D			
C-423	2	No. 2 Prec Rdnt Mag Reg Bit No. 3	D			
C-423	3	No. 2 Prec Rdnt Mag Reg Bit No. 4	D			
C-423	4	No. 2 Prec Rdnt Mag Reg Bit No. 5	D			
C-423	5	No. 2 Prec Rdnt Mag Reg Bit No. 6	D			
C-423	6	No. 2 Prec Rdnt Mag Reg Bit No. 7	D			
C-424	1	No. 2 Prec Rdnt Mag Reg Bit No. 8	D			
C-424	2	No. 2 Prec Angle Register Bit No. 1	D			
C-424	3	No. 2 Prec Angle Register Bit No. 2	D			
C-424	4	No. 2 Prec Angle Register Bit No. 3	D			
C-424	5	No. 2 Prec Angle Register Bit No. 4	D			
C-424	6	No. 2 Prec Angle Register Bit No. 5	D			
C-425	1	No. 2 Prec Angle Register Bit No. 6	D			
C-425	2	No. 2 Prec Angle Register Bit No. 7	D			
C-425	3	No. 2 Prec Angle Register Bit No. 8	D			
C-425	4	No. 2 Prec Angle Register Bit No. 9	D			
C-425	5	Delta V-Redundant Mag Reg Bit No. 1	D			
C-425	6	Delta V-Redundant Mag Reg Bit No. 2	D			
C-426	1	Delta V-Redundant Mag Reg Bit No. 3	D			
C-426	2	Delta V-Redundant Mag Reg Bit No. 4	D			
C-426	3	Delta V-Redundant Mag Reg Bit No. 5	D			
C-426	4	Delta V-Redundant Mag Reg Bit No. 6	D			
C-426	5	Delta V-Redundant Mag Reg Bit No. 7	D			
C-426	6	Delta V-Redundant Mag Reg Bit No. 8	D			
C-427	1	Delay Register Bit No. 1	D			
C-427	2	Delay Register Bit No. 2	D			
C-427	3	Delay Register Bit No. 3	D			

Table 6-1. (continued)

Word	Bit	Measurement	Type	Range	Normal Response Before Despin	Remarks
C-427	4	Delay Register Bit No. 4	D			
C-427	5	Delay Register Bit No. 5	D			
C-427	6	Delay Register Bit No. 6	D			
C-428	1	Delay Register Bit No. 7	D			
C-428	2	No. 1 Prec Rdnt Mag Reg Bit No. 1	D			
C-428	3	No. 1 Prec Rdnt Mag Reg Bit No. 2	D			
C-428	4	No. 1 Prec Rdnt Mag Reg Bit No. 3	D			
C-428	5	No. 1 Prec Rdnt Mag Reg Bit No. 4	D			
C-428	6	No. 1 Prec Rdnt Mag Reg Bit No. 5	D			
C-429	1	No. 1 Prec Rdnt Mag Reg Bit No. 6	D			
C-429	2	No. 1 Prec Rdnt Mag Reg Bit No. 7	D			
C-429	3	No. 1 Prec Rdnt Mag Reg Bit No. 8	D			
C-429	4	No. 1 Prec Angle Register Bit No. 1	D			
C-429	5	No. 1 Prec Angle Register Bit No. 2	D			
C-429	6	No. 1 Prec Angle Register Bit No. 3	D			
C-430	1	No. 1 Prec Angle Register Bit No. 4	D			
C-430	2	No. 1 Prec Angle Register Bit No. 5	D			
C-430	3	No. 1 Prec Angle Register Bit No. 6	D			
C-430	4	No. 1 Prec Angle Register Bit No. 7	D			
C-430	5	No. 1 Prec Angle Register Bit No. 8	D			
C-430	6	No. 1 Prec Angle Register Bit No. 9	D			
C-431	1	Sequencer Status Bit No. 1	D		0	
C-431	2	Sequencer Status Bit No. 2	D		0	
C-431	3	Sequencer Status Bit No. 3	D		0	
C-431	4	Star Time Gate Enable Status	D	Enable/Disable		
C-431	5	Reference Select Bit No. 1	D		1	
C-431	6	Reference Select Bit No. 2	D		0	

Table 6-1. (continued)

Word	Bit	Measurement	Type	Range	Normal Response Before Despin	Remarks
C-432	1	Star Location Bit No. 1	D			
C-432	2	Star Location Bit No. 2	D			
C-432	3	ACS Reg Inhibit State	D	Enable/Disable	0	
C-432	4	Arm Status Bit No. 1	D		0	
C-432	5	Arm Status Bit No. 2	D		0	
C-432	6	Arm Status Bit No. 3	D		0	
		Power Subsystem				
C-105		RTG 2 Current	A	0 to 10 Amps	9.75	
C-106		Battery Voltage	A	0-15 VDC	11.4	
C-107		DC Bus Voltage--Expanded	A	26-30 VDC	28.0	
C-109		Battery Charge Current	A	0-0.3 Amps	0.0	
C-110		RTG Voltage--RTG 1	A	0-6 VDC	3.9	
C-113		RTG Voltage--RTG 4	A	0-6 VDC	3.9	
C-114		RTG 3 Current	A	0 to 10 Amps	9.75	
C-115		Battery Temperature	A	-20 to +120	85	
C-117		TRF +5V CDU Output Ch A	A	0-6 Volts	5.3	
C-118		TRF +5V CDU Output Ch B	A	0-6 Volts	5.3	
C-119		DC Bus Voltage	A	0-30 VDC	28.0	
C-122		Shunt Bus Current	A	0-3 Amps	2.9	
C-123		RTG 4 Current	A	0 to 10 Amps	9.75	
C-125		RTG Voltage--RTG 2	A	0-6 VDC	3.9	

Table 6-1. (continued)

Word	Bit	Measurement	Type	Range	Normal Response Before Despin	Remarks
C-126		Battery Discharge Current	A	0 to 10 Amps	0.0	
C-127		RTG 1 Current	A	0 to 10 Amps	9.75	
C-128	1	Battery Charge Status	B	Auto/Cmd	Auto	
C-128	2	Battery Discharge Reg Status	B	Enable/Disable	Enable	
C-129		DC Bus Current	A	0 to 6 Amps	1.01	
C-131		RTG Voltage--RTG 3	A	0 to 6 VDC	3.9	
C-209		Shunt Bus Current	A	0 to 3 Amps	2.9	
		Communications Subsystem				
C-111		RCV 1 AGC Conscan	A	± 4 db AM	-4 db (off)	Reader at low end if off
C-121		RCV 2 AGC Conscan	A	± 4 db AM	-4 db (off)	
C-205		TWTA No. 1 Temperature	A	40 to 125°F	100°F	
C-206		Aux OSC Temp Driver A	A	20 to 110°F	85°F	
C-207		Converter Temperature TWTA A	A	40 to 125°F	100°F	
C-208		Cathode Current TWTA A	A	24 to 30 MA	29.26 ma	
C-211		Helix Current TWTA A	A	0 to 10 MA	6.0 ma	
C-212		Loop Stress, Receiver A	A	-100 to +100	0 KHz	
C-213		Signal Strength, Receiver B	A	-149 to -63	-149 dbm (off)	
C-214		Output Power Monitor TWTA B	A	26 to 40.4 DBM	26 (off)	
C-215		Cathode Current TWTA B	A	24 to 30 MA	24 (off)	

Table 6-1. (continued)

Word	Bit	Measurement	Type	Range	Normal Response Before Despin	Remarks
C-216		Helix Current TWTA B	A	0 to 10 MA	0 ma	
C-221		Converter Temperature TWTA B	A	40 to 125°F	40°F (off)	
C-222		Oscillator Temp Receiver A	A	20 to 110°F	85°F	
C-223		Aux OSC Temp Driver B	A	20 to 110°F	85°F	
C-224		Reference Voltage TWTA A	A	0 to 28 VDC	28 VDC	
C-227		Oscillator Temp Receiver B	A	20 to 110°F	85°F	
C-228		TWTA No. 2 Temperature	A	40 to 125°F	40°F (off)	
C-229		Loop Stress, Receiver B	A	-100 to +100		
C-230		Reference Voltage TWTA B	A	0 to 28 VDC	0 VDC (off)	
C-231		Output Power Monitor TWTA A	A	20 to 40 DBM	39 dbm	
C-232		Signal Strength, Receiver A	A	-149 to -63	-80 dbm	
C-308	1	RCVR A Signal Present	B	Absent/Pres	Present	
C-308	2	RCVR B Signal Present	B	Absent/Pres	Absent	
C-308	3	Osc Enable/Disable, RCVR A	B	Disable/Enable	Enable	
C-308	4	Osc Enable/Disable, RCVR B	B	Disable/Enable	Enable	
C-313	1	Phase Output Status	D	0 Deg/180 Deg	0 Deg	
C-313	2	Threshold Mode Status	D	Med/High	Med	
C-313	3	A Sin Bit 1 (LSB)	D		0	
C-313	4	A Sin Bit 2	D		0	
C-313	5	A sin Bit 3	D		0	
C-313	6	A sin Bit 4	D		0	

Table 6-1. (continued)

Word	Bit	Measurement	Type	Range	Normal Response Before Despin	Remarks
C-314	1	A Sin Bit 5	D		0	
C-314	2	A Sin Bit 6	D		0	
C-314	3	A Sin Bit 7	D		0	
C-314	4	A Sin Bit 8 (MSB)	D		0	
C-314	5	A Cos Bit 1 (LSB)	D		0	
C-314	6	A Cos Bit 2	D		0	
C-315	1	A Cos Bit 3	D		0	
C-315	2	A Cos Bit 4	D		0	
C-315	3	A Cos Bit 5	D		0	
C-315	4	A Cos Bit 6	D		0	
C-315	5	A Cos Bit 7	D		0	
C-315	6	A Cos Bit 8 (MSB)	D		0	
C-316	1	Conscan Status	B	Off/On	Off	
C-316	2	Conscan Threshold	B	Below/Above	Below	
		Antenna Subsystem				
C-316	4	Receiver Switch Position	B	Med/High	Medium	
C-316	5	Transmitter Switch Position	B	Med/High	Medium	
C-316	6	Antenna Feed Switch Position	B	Norm/Offset	Offset	
		Propulsion Subsystem				
C-210		Propellant Supply Pressure	A	0 to 600 psi	500 psia	
C-308	5	Spin Control Thruster 2 Pulse Count	B		0	
C-308	6	Spin Control Thruster 1 Pulse Count	B		0	
C-309		VPTCA No. 1 Prop Inlet Temperature	A	40 to 200°F	60	
C-310		TCA Propulsion Inlet Temperature	A	40 to 200°F	60	

Table 6-1. (continued)

Word	Bit	Measurement	Type	Range	Normal Response Before Despin	Remarks
C-311		VPT-1 Thruster Temp	A	400 to 1800°F	400 (off)	
C-312		VPT-2 Thruster Temp	A	400 to 1800°F	400 (off)	
C-321		Velocity Thruster No. 2 Pulse Count	D	0 to 63	0	
C-322		Velocity Thruster No. 4 Pulse Count	D	0 to 63	0	
C-325		VPT-4 Thruster Temp	A	400 to 1800°F	400 (off)	
C-326		VPTCA No. 2 Prop Inlet Temperature	A	40 to 200°F	60	
C-327		Propellant Supply Temperature	A	40 to 140°F	700	
C-328		VPT-3 Thruster Temp	A	400 to 1800°F	400 (off)	
C-329		Velocity Thruster No. 1 Pulse Count	D	0 to 63	0	
C-330		Velocity Thruster No. 3 Pulse Count	D	0 to 63	0	
C-409	1	Thruster Initiation Status VPT 1	B	No/Yes	No	
C-409	2	Thruster Initiation Status VPT 2	B	No/Yes	No	
C-409	3	Thruster Initiation Status VPT 4	B	No/Yes	No	
C-409	4	Thruster Initiation Status VPT 3	B	No/Yes	No	
C-409	5	Thruster Initiation Status VPT 1	B	No/Yes	No	
C-409	6	Thruster Initiation Status VPT 2	B	No/Yes	No	

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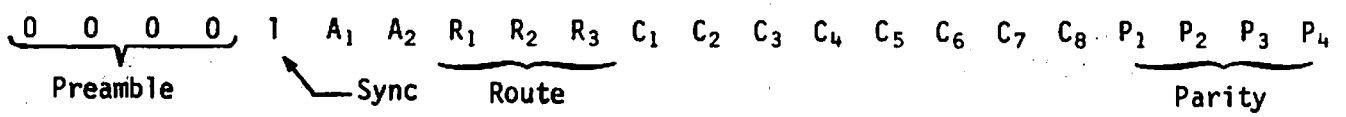
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APPENDIX A

PIONEER COMMANDS

COMMAND MESSAGE

The normal command message contains twenty-two bits as follows:



where:

(a) A_1 and A_2 are decoder addresses as follows:

A_1	A_2	Decoder
0	1	A
1	0	B

(b) R_1, R_2, and R_3 are routing addresses as follows:

R_1	R_2	R_3	
0	0	0	Not used
0	0	1	Real Time Command (all except DTU)
0	1	0	Stored Command (all except DTU)
0	1	1	Stored DTU Command (Item 2 of Attachment B)
1	0	0	Real Time DTU Command (Item 2 of Attach. B)
1	0	1	ACS Quantitative Command
1	1	0	Stored Command Time Delay (all except ACS Registers)
1	1	1	Not used

(c) C_1 through C_8 are command with C_1 "most significant bit" except for ACS registers

(d) P_1 through P_4 are parity bits having values satisfying following (even parity)

$$P_1 = R_1 \oplus R_2 \oplus R_3 \oplus C_1 \oplus C_2 \oplus C_3 \oplus C_4$$

$$P_2 = R_1 \oplus R_2 \oplus R_3 \oplus C_1 \oplus C_6 \oplus C_7 \oplus C_8$$

$$P_3 = R_1 \oplus R_2 \oplus C_2 \oplus C_3 \oplus C_5 \oplus C_6 \oplus C_7$$

$$P_4 = R_1 \oplus R_3 \oplus C_2 \oplus C_4 \oplus C_5 \oplus C_6 \oplus C_8$$

PIONEER F/G QUALITATIVE COMMANDS

ITEM NO	TITLE
1.	INSTRUMENTS
1.1	ALL
1.2	HELIUM VECTOR MAGNETOMETER
1.3	PLASMA ANALYZER
1.4	CHARGED PARTICLE INSTRUMENT
1.5	GEIGER TUBE TELESCOPE
1.6	COSMIC RAY TELESCOPE
1.7	TRAPPED RADIATION DETECTOR
1.8	ULTRAVIOLET PHOTOMETER
1.9	IMAGING PHOTO POLARIMETRY
1.10	INFRARED RADIOMETER
1.11	ASTEROID/METEOROID DETECTOR
1.12	METEOROID DETECTOR
2.	DATA HANDLING
2.1	OPERATING MODE
2.2	BIT RATE
2.3	FORMAT
2.4	ROLL-INDEX PULSE
3.	COMMUNICATION
3.1	ANTENNA SELECTION
3.2	TRANSMITTER POWER
3.3	COHERENT ON/OFF
4.	CONSCAN AND FEED
4.1	CONSCAN
4.2	MAIN FEED OFFSET

ITEM NO	TITLE
5.	ATTITUDE CONTROL AND PROPULSION
5.1	STORAGE REGISTERS
5.2	PRECESSION
5.3	PULSE LENGTH
5.4	VELOCITY
5.5	SPIN
5.6	SENSOR SELECT
5.7	STAR ACQUISITION-SECTOR
5.8	STAR ACQUISITION-TIME
5.9	PROGRAM STORAGE-PSE
5.10	DURATION/STEER LOGIC
5.11	CLOCK
6.	COMMAND DISTRIBUTION
6.1	STORAGE REGISTERS
6.2	CDU
7.	ELECTRICAL POWER
7.1	BATTERY
7.2	POWER SYSTEM PROTECTION
8.	SEPARATION SEQUENCE
8.1	SEQUENCER
8.2	ORDNANCE
8.3	DEPLOYMENT
9.	MISCELLANEOUS
9.1	TEST COMMANDS

PIONEER F/G QUALITATIVE COMMANDS

ITEM NO	TITLE	DESCRIPTION	OCTAL NO	DESIGNATOR
1.	INSTRUMENTS			
1.1	ALL			
1.1.1	Instrument Power	Turns off power to all instruments	344	INSO
1.2	HELIUM VECTOR MAGNETOMETER			
1.2.1	Instrument off	Turns off power to HVM	214	HVMO
1.2.2	Instrument on	Turns on power to HVM	302	HVM9
1.2.3	Manual Range On/Off	Transfers to Manual Range or Non-Data Interchange	063	HVM1
1.2.4	Range Decrement	Decrements Manual Range by one level	030	HVM2
1.2.5	Range Increment	Increments Manual Range by one level or transfers to automatic range, or institutes calibration	041	HVM3
1.2.6	Data Interchange	Transfers to data interchange	052	HVM4
1.3	PLASMA ANALYZER			
1.3.1	Instrument Off	Turns off power to PA	225	PAYO
1.3.2	Instrument On	Turns on power to PA	313	PAY9
1.3.3	FSM Spacecraft Rev/HV	Steps through four "Rev/HV" when in FSM mode	046	PAY1
1.3.4	Data Source Mode/CCM	Steps through four modes (HFM) or three modes (FSM)	057	PAY2
1.3.5	In Flight Calibration	Selects calibration mode	071	PAY3
1.3.6	Detector A HV Step	Steps through three "energy range/step" values	035	PAY4
1.3.7	Detector A HV/CCM Bias	Toggles between "HV to CCM bias" on and off	106	PAY5
1.3.8	Left CCM Bias Supply Step	Steps through eight HV levels on CCM bias	121	PAY6

ITEM NO	TITLE	DESCRIPTION	OCTAL NO	DESIGNATOR
1.3.9	Right CCM Bias Supply Step	Steps through eight HV levels on right	140	PAY7
1.3.10	Integration Period	Steps through four "integration" periods	132	PAY8
1.3.11	Target Suppression	Toggles between "Target Suppression" on and off	013	PAW1
1.3.12	Detector B HV Step	Toggles between 32 and 64 "energy levels" per cycle	024	PAW2
1.3.13	Detector B HV Limit	Toggles between HV limit on and off	060	PAW3
1.3.14	Detector B HV	Toggles between HV on and off	117	PAW4
1.4	CHARGED PARTICLE INSTRUMENT			
1.4.1	Instrument Off	Turns off power to CPI	203	CPD0
1.4.2	Instrument on	Turns on power to CPI	335	CPD9
1.4.3	Priority Mode Control	Disables priority mode	103	CPD1
1.4.4	Disable Detector D2	Disables Detector D2	114	CPD2
1.4.5	Disable Detector D1	Disables Detector D1	150	CPD3
1.4.6	Disable Detector D7	Disables Detector D7	136	CPD4
1.4.7	Calibration	Places instrument in calibrate mode	147	CPD5
1.4.8	Logic Reset	Enables disabled mode and detectors, returns instrument to normal mode	125	CPD6
1.5	GEIGER TUBE TELESCOPE			
1.5.1	Instrument off	Turns off power to GTT	231	GTT0
1.5.2	Instrument on	Turns on power to GTT	346	GTT9
1.5.3	Redundant Logic Select	Toggles between logic A and B	162	GTT1
1.6	COSMIC RAY TELESCOPE			
1.6.1	Instrument off	Turns off power to CRT	236	CRT0
1.6.2	Instrument on	Turns on power to CRT	324	CRT9
1.6.3	Calibrate	Toggles between normal and calibrate	130	CRT1
1.6.4	Mode Select Step	Increments one step through "Lo Pwr. Sec. Sync Off"; "Hi Pwr, Sec. Sync on"; "Lo Pwr, Sec. Sync On"; "Hi Pwr, Sec. Sync Off"	141	CRT2

PIONEER F/G QUALITATIVE COMMANDS

ITEM NO	TITLE	DESCRIPTION	OCTAL NO	DESIGNATOR
1.7	TRAPPED RADIATION DETECTOR			
1.7.1	Instrument Off	Turns off power to TRD	253	TRD0
1.7.2	Instrument On	Turns on power to TRD	320	TRD9
1.7.3	High Voltage Mode Select	Selects high-voltage mode	105	TRD1
1.7.4	Low Voltage Mode Select	Selects low-voltage mode	116	TRD2
1.8	ULTRAVIOLET PHOTOMETER			
1.8.1	Instrument Off	Turns off power to UVP	220	UVP0
1.8.2	Instrument On	Turns on power to UVP	361	UVP9
1.8.3	Remove Cover	Fires piston actuator to remove cover	100	UVP1
1.9	IMAGING PHOTOPOLARIMETRY			
1.9.1	Instrument Off	Turns off power to IPP	264	IPY0
1.9.2	Instrument On	Turns on power to IPP	372	IPY9
1.9.3	Look Angle Step Inhibit	Inhibits telescope stepping	006	IPY1
1.9.4	Start Look Angle Reset	Resets telescope to reference position	040	IPY2
1.9.5	Start Look Angle Increment	Sets telescope to selected starting look angle	051	IPY3
1.9.6	Look Angle Step Reverse	Reverse direction of telescope stepping	165	IPY4
1.9.7	Low Sample Rate	Reduces sample rate by half	176	IPY5
1.9.8	Start Data at Threshold	Syncs data taking to sensor threshold level	017	IPY6
1.9.9	Spoke Advance Fine	Advances spoke wheel 1/64th roll	142	IPY7
1.9.10	Spoke Advance Coarse	Advances spoke wheel 1/16th roll	153	IPY8
1.9.11	Gain Increment	Increases channeltron gain setting	062	IPW1
1.9.12	Gain Decrement	Reduces channeltron gain setting	073	IPW2
1.9.13	Mode 2 On	Selects zodiacal Light Mode	104	IPW3
1.9.14	Mode 3 On	Selects Jupiter Photopolarimetry Mode	111	IPW4
1.9.15	Mode 4 ON	Selects Jupiter Imaging Mode	120	IPW5

ITEM NO	TITLE	DISCRIPTION	OCTAL NO	DESIGNATOR
1.10	INFRARED RADIOMETER			
1.10.1	Instrument Off	Turns off power to IRR	275	IRR0
1.10.2	Instrument On	Turns on power to IRR	342	IRR9
1.10.3	Spoke Wheel Advance	Change start of data taking cycle	070	IRR1
1.11	ASTEROID METEOROID DETECTOR			
1.11.1	Instrument Off	Turns off power to AMD	247	AMD0
1.11.2	Instrument On	Turns on power to AMD	353	AMD9
1.11.3	Threshold Level Normal	Selects normal-level threshold	005	AMD1
1.11.4	Threshold Level High	Selects high-level threshold	016	AMD2
1.11.5	Narrow Bandwidth	Selects narrow bandwidth for amplifier	161	AMD3
1.11.6	Medium Bandwidth	Selects medium bandwidth for amplifier	027	AMD4
1.11.7	Wide Bandwidth	Selects wide bandwidth for amplifier	172	AMD5
1.11.8	Star Exclusion	Toggles between star exclusion enable and disable	127	AMD6
1.11.9	Data Readout	Toggles between data readout enable and disable	174	AMD7
1.12	METEOROID DETECTOR			
1.12.1	Instrument Off	Turns off power to MD	257	MED0
1.12.2	Instrument On	Turns on power to MD	375	MED9
2.	DATA HANDLING			
2.1	OPERATING MODE			
2.1.1	Real Time Mode		060	RTM1
2.1.2	Telemetry Store Mode		062	TSM1
2.1.3	Memory Readout		061	MRM1
2.1.4	Memory Halt		064	MHM1
2.1.5	Memory Continue		065	MCM1
2.1.6	Coded Data		070	CCM1
2.1.7	Uncoded Data		071	UCM1
2.1.8	Redundancy A	Selects DTU redundancy configuration A	044	DTA1
2.1.9	Redundancy B	Selects DTU redundancy configuration B	155	DTB1

PIONEER F/G QUALITATIVE COMMANDS

ITEM NO	TITLE	DESCRIPTION	OCTAL NO	DESIGNATOR
2.2	BIT RATE			
2.2.1	16 bps		040	BIT1
2.2.2	32 bps		041	BIT2
2.2.3	64 bps		042	BIT3
2.2.4	128 bps		043	BIT4
2.2.5	256 bps		044	BIT5
2.2.6	512 bps		045	BIT6
2.2.7	1024 bps		046	BIT7
2.2.8	2048 bps		047	BIT8
2.3	FORMAT			
2.3.1	Format A		010	FMA1
2.3.2	Format B		001	FMB1
2.3.3	Format C		002	FMC5
2.3.4	Format C-1		004	FMC1
2.3.5	Format C-2		005	FMC2
2.3.6	Format C-3		006	FMC3
2.3.7	Format C-4		007	FMC4
2.3.8	Format A/D-1		030	FAD1
2.3.9	Format A/D-2		031	FAD2
2.3.10	Fromat A/D-3		032	FAD3
2.3.11	Format B/D-1		020	FBD1
2.3.12	Format B/D-2		021	FBD2
2.3.13	Format B/D-3		022	FBD3
2.4	ROLL-INDEX PULSE			
2.4.1	Roll Reference 0 Deg.	Provides roll pulse when sensor stimulated	066	RIP1
2.4.2	Roll Reference 180 Deg.	Delays roll pulse 1/2 revolution after sensor stimulated	067	RIP2
2.4.3	Averaging Mode	Selects Averaging Mode for SPSG	077	RIP3
2.4.4	Non-Averaging Mode	Selects Non-Averaging Mode for SPSG	076	RIP4
2.4.5	ACS Mode	Selects ACS Mode for SPSG	074	RIP5

ITEM NO	TITLE	DESCRIPTION	OCTAL NO	DESIGNATOR
3.	COMMUNICATION			
3.1	ANTENNA SELECTION			
3.1.1	TWTA A to HG Antenna	Connects TWTA A to High Gain, TWTA B to Med. Gain	011	ANT1
3.1.2	TWTA A to MG Antenna	Connects TWTA A to Med. Gain, TWTA B to High Gain	122	ANT2
3.1.3	Rcvr A to HG Antenna	Connects Rcvr A to High Gain, Rcvr B to Med. Gain	001	ANT3
3.1.4	Rcvr A to MG Antenna	Connects Rcvr A to Med. Gain, Rcvr B to High Gain	235	ANT4
3.1.5	Rcvr Reverse Enable	Resets time and enables automatic Rcvr/ Antenna Reverse	134	ANT5
3.1.6	Rcvr Reverse Inhibit	Inhibits automatic Rcvr/Antenna Reverse	202	ANT6
3.2	TRANSMITTER POWER			
3.2.1	Trans A Off	Turns off power to TWTA A and Driver A	232	TRA0
3.2.2	Trans A On	Turns on power to TWTA A and Driver A	223	TRA9
3.2.3	Trans B Off	Turns off power to TWTA B and Driver B	331	TRB0
3.2.4	Trans B On	Turns on power to TWTA B and Driver B	310	TRB9
3.3	COHERENT ON/OFF			
3.3.1	Coherent Mode Off - Rcvr A	Inhibits coherent output of Rcvr A	354	COA1
3.3.2	Coherent Mode On - Rcvr A	Allows coherent output of Rcvr A TCXO stays on	343	COA2
3.3.3	Coherent Mode Off - Rcvr B	Inhibits coherent output of Rcvr B	212	COB1
3.3.4	Coherent Mode On - Rcvr B	Allows coherent output of Rcvr B	226	COB2

PIONEER F/G QUALI IVE COMMANDS

ITEM NO	TITLE	DESCRIPTION	OCTAL NO	DESIGNATOR
4.	CONSCAN AND FEED			
4.1	CONSCAN			
4.1.1	Conscan Off	Turns power off to conscan	133	CNS0
4.1.2	Conscan On	Turns power on to conscan	026	CNS9
4.1.3	Conscan Threshold High	Sets conscan threshold high	234	CNS1
4.1.4	Conscan Threshold Low	Sets conscan threshold low	321	CNS2
4.1.5	Conscan 0° Output	Selects conscan output for use with Precession Pair 2	007	CNS3
4.1.6	Conscan 180° Output	Selects conscan output for use with Precession Pair 1	144	CNS4
4.1.7	Conscan to DSL Enable	Allows conscan pulses to enter DSL	352	CNS5
4.1.8	Conscan to DSL Inhibit	Inhibits conscan pulses from entering DSL	363	CNS6
4.2	MAIN FEED OFFSET			
4.2.1	Main Feed (A) Normal	Turns power off to Feed Movement heater A	356	MFA0
4.2.2	Main Feed (A) Offset	Turns power on to Feed Movement heater A	210	MFA9
4.2.3	Main Feed (B) Normal	Turns power off to Feed Movement heater B	201	MFB0
4.2.4	Main Feed (B) Offset	Turns power on to Feed Movement heater B	364	MFB9
5.	ATTITUDE CONTROL AND PROPULSION			
5.1	STORAGE REGISTERS			
5.1.1	Arm Register 1	Enables data entry into first precession register	233	ACS1
5.1.2	Arm Register 2	Enables data entry into ΔV register	244	ACS2
5.1.3	Arm Register 3	Enables data entry into second precession register	255	ACS3
5.1.4	Sequence Execute	Starts program sequence	222	ACX1
5.1.5	Sequence Inhibit	Inhibits continuation of program sequence	345	ACX2
5.1.6	Sequence Override	Overrides "Sequence Inhibit," starts at inhibit point	010	ACX3
5.1.7	Sequence Reset	Returns sequence to initial state	101	ACX4
5.1.8	Sequence Step	Advances sequence one step	042	ACX5

ITEM NO	TITLE	DESCRIPTION	OCTAL NO	DESIGNATOR
5.2	PRECESSION			
5.2.1	Precession Pair 1	Selects precession pair 1 (positive precession)	266	PRE1
5.2.2	Precession Pair 2	Selects precession pair 2 (negative precession)	322	PRE2
5.2.3	0° Precession Pulse	Single precession pulse at nominally 0° roll	230	PRE3
5.2.4	90° Precession Pulse	Single precession pulse at nominally 90° roll	241	PRE4
5.2.5	180° Precession Pulse	Single precession pulse at nominally 180° roll	252	PRE5
5.2.6	270° Precession Pulse	Single precession pulse at nominally 270° roll	263	PRE6
5.2.7	Real Time Precession Pulse	Single precession pulse at instant of receipt at S/C	274	PRE7
5.3	PULSE LENGTH			
5.3.1	Pulse Length 1	Selects 0.0312 sec pulse length	022	PUL1
5.3.2	Pulse Length 2	Selects 0.125 sec pulse length	213	PUL2
5.3.3	Pulse Length 3	Selects 0.5 sec pulse length	143	PUL3
5.3.4	Pulse Length 4	Selects 1.0 sec pulse length	301	PUL4
5.3.5	Pulse Length 5	Selects 2.0 sec pulse length	055	PUL5
5.4	VELOCITY			
5.4.1	ΔV Pair 1	Selects aft-thrust pair (positive ΔV)	277	VEL1
5.4.2	ΔV Pair 2	Selects forward-thrust pair (negative ΔV)	333	VEL2
5.4.3	Real Time ΔV Pulse	Single ΔV pulse at instant of receipt at S/C	205	VEL3
5.4.4	ΔV/SCT Mode	Enables ΔV pulse pair perpendicular to spin axis	216	VEL4
5.5	SPIN			
5.5.1	Despin Start	Ground backup to initiate despin	330	SPN1
5.5.2	Real Time Spin Pulse	Produces single spin pulse at instant of receipt at S/C	227	SPN2
5.5.3	Spin Direction Up	Select SCT1 to increase spin rate	224	SPN3
5.5.4	Spin Direction Down	Select SCT2 to decrease spin rate	312	SPN4

PIONEER F/G QUAL.,ATIVE COMMANDS

ITEM NO	TITLE	DESCRIPTION	OCTAL NO	DESIGNATOR
5.6	SENSOR SELECT			
5.6.1	Sensor Select Sun A	Selects A output of Sun Sensor	303	SEN1
5.6.2	Sensor Select Sun B	Selects B output of Sun Sensor	314	SEN2
5.6.3	Sensor Select Star	Selects star sensor output for reference	325	SEN3
5.7	STAR ACQUISITION - SECTOR			
5.7.1	Star Gate Enable 45° Sector	Blanks star pulse for 315° of roll	256	SAS1
5.7.2	Star Gate Enable 360° Sector	Allows 360° star scan output	334	SAS2
5.7.3	Star Delay Up - Least Significant	Adds one increment to star pulse delay register (increases 3.91 ms)	245	SAS3
5.7.4	Star Delay Up - Most Significant	Adds 64 increments to star pulse delay register (increases 250.00 ms)	373	SAS4
5.7.5	Star Delay Down - Least Significant	Subtracts one increment from star pulse delay register (decreases 3.91 ms)	323	SAS5
5.7.6	Star Delay Down - Most Significant	Subtracts 64 increments to star pulse delay register (decreases 250.00 ms)	337	SAS6
5.8	STAR ACQUISITION - TIME			
5.8.1	Star Time Gate Enable	Accepts star pulse during time gate	340	SAT1
5.8.2	Star Time Gate Up	Adds one time increment to gate register	351	SAT2
5.8.3	Star Time Gate Down	Subtracts one time increment to gate register	362	SAT3
5.9	PROGRAM STORAGE - PSE			
5.9.1	Program Storage and Exec. Off	Turns off power to PSE	025	PSE0
5.9.2	Standby Power On	Turns on power to PSE and removes power from both DSLs	166	PSE9
5.10	DURATION-STEER LOGIC			
5.10.1	Standby Power	Turns off power to DSL A and B	166	SLP0
5.10.2	Duration/Steer Logic A On	Turns on power to DSL A, turns off power to DSL B, resets PSE register	036	SLA9
5.10.3	Duration/Steer Logic B On	Turns on power to DSL B, turns off power to DSL A, resets PSE register	047	SLB9

ITEM NO	TITLE	DESCRIPTION	OCTAL NO	DESIGNATOR
5.11	CLOCK			
5.11.1	Time Frequency Select A	Selects DTU clock output A	336	ACA1
5.11.2	Time Frequency Select B	Selects DTU clock output B	347	ACB1
5.11.3	Clock Reset	Resets clock-enable	206	ACR1
6.	COMMAND DISTRIBUTION			
6.1	STORAGE REGISTERS			
6.1.1	Select Register No. 1	Enables data entry to register 1	102	CDS1
6.1.2	Select Register No. 2	Enables data entry to register 2	113	CDS2
6.1.3	Select Register No. 3	Enables data entry to register 3	124	CDS3
6.1.4	Select Register No. 4	Enables data entry to register 4	135	CDS4
6.1.5	Select Register No. 5	Enables data entry to register 5	146	CDS5
6.1.6	Reset Storage Registers	Resets all storage registers to zero	355	CDR1
6.1.7	Reset "Select Register"	Disables all registers for additional data entry	267	CDR2
6.1.8	Execute Stored Sequence	Enables stored data to be executed	254	CDX1
6.1.9	Stop Stored Sequence	Stops execution of stored data	332	CDX2
6.2	CDU			
6.2.1	Cmd Proc 1 Select	Selects storage output to CP 1	012	CDU1
6.2.2	Cmd Proc 2 Select	Selects storage output to CP 2	023	CDU2
6.2.3	CDU Select 5V dc Bus A	Selects Bus A	020	CDU3
6.2.4	CDU Select 5V dc Bus B	Selects Bus B	131	CDU4

PIONEER F/G QUALITATIVE COMMANDS

ITEM NO	TITLE	DESCRIPTION	OCTAL NO	DESIGNATOR
7.	ELECTRICAL POWER			
7.1	BATTERY			
7.1.1	Battery Heater Off	Turns off battery heater	126	BAT0
7.1.2	Battery Heater On	Turns on battery heater	015	BAT9
7.1.3	Battery Auto Charge	Places battery in automatic-charge mode	043	BAT1
7.1.4	Battery Float Charge	Places battery in float-charge	164	BAT2
7.1.5	Battery Discharge Enable		004	BAT3
7.1.6	Battery Discharge Disable		115	BAT4
7.2	POWER SYSTEM PROTECTION			
7.2.1	Relay Reset	Resets four RTG relays	204	PSP1
7.2.2	Overload Protection Off	Prevents overload protection	246	PSP2
7.2.3	Overload Protection On	Automatically removes electrical load when voltage drops	123	PSP3
8.	SEPARATION SEQUENCE			
8.1	SEQUENCER			
8.1.1	Start Sequencer	Ground backup to start sequencer	112	SEQ1
8.1.2	Sequencer Inhibit	Inhibit sequencer outputs but not sequencer clock	061	SEQ2
8.1.3	Sequencer Enable	Enables sequencer outputs	072	SEQ3
8.2	ORDNANCE			
8.2.1	Ordnance Safe	Sets ordnance circuitry to safe	365	ORD1
8.2.2	Ordnance Arm	Sets ordnance circuitry to arm	243	ORD2

ITEM NO	TITLE	DESCRIPTION	OCTAL NO	DESIGNATOR
8.3	DEPLOYMENT			
	RTG Deploy	Ground backup to deploy RTG	200	DEP1
	RTG Deploy Redundant	Redundant ground backup to deploy RTG	300	DEP2
	Boom Deploy	Ground backup to deploy magnetometer boom and uncage main feed	211	DEP3
	Boom Deploy Redundant	Redundant ground backup to deploy magnetometer boom and uncage main feed	311	DEP4
9.	MISCELLANEOUS			
9.1	TEST COMMANDS			
9.1.1	Zero Cmd	Also used to fill registers when no activity desired	000	ZERO
9.1.2	One Cmd		111	ONE1

QUANTITATIVE COMMANDS

1. CDU STORAGE REGISTER

The CDU contains five storage registers. Five commands are available for selecting the storage register to be loaded with data (see Item 6.1 of Attachment B). The data consist of the qualitative command to be executed and a time delay from the time of executing the qualitative command in the previous register or the "Execute Stored Sequence" command (when considering first storage register). After selecting the register, two commands, a qualitative command and a quantitative time delay command (in that order), are required to load the register. The time delay command uses only seven bits; the first bit of the command is not used. The resolution of the time delay command is a function of the range to be covered as follows:

Range	Resolution
0 to 32 sec	1 sec
32 to 288 sec	8 sec
288 to 384 sec	96 sec
384 to 8320 sec	128 sec

Typical values of the time delay command for various time delays are as follows:

Time Delay, Sec.	Time Delay Command								Octal Equivalent
	MSB							LSB	
0	0	0	0	0	0	0	0	0	00
1	0	0	0	0	0	0	0	1	01
31	0	0	0	1	1	1	1	1	37
32	0	0	1	0	0	0	0	0	40
40	0	0	1	0	0	0	0	1	41
280	0	0	1	1	1	1	1	1	77
288	0	1	0	0	0	0	0	0	100
384	0	1	0	0	0	0	0	1	101
512	0	1	0	0	0	0	1	0	102
8320	0	1	1	1	1	1	1	1	177

The following table presents the CDU time delay commands for all selectable times. The time in seconds is shown versus the octal code for each command.

CDU TIME DELAY COMMANDS

OCTAL CODE	TIME (SECONDS)	OCTAL CODE	TIME (SECONDS)	OCTAL CODE	TIME (SECONDS)
0	0	53	120	126	3072
1	1	54	128	127	3200
2	2	55	136	130	3328
3	3	56	144	131	3456
4	4	57	152	132	3584
5	5	60	160	133	3712
6	6	61	168	134	3840
7	7	62	176	135	3968
10	8	63	184	136	4096
11	9	64	192	137	4224
12	10	65	200	140	4352
13	11	66	208	141	4480
14	12	67	216	142	4608
15	13	70	224	143	4736
16	14	71	232	144	4864
17	15	72	240	145	4992
20	16	73	248	146	5120
21	17	74	256	147	5248
22	18	75	264	150	5376
23	19	76	272	151	5504
24	20	77	280	152	5632
25	21	100	288	153	5760
26	22	101	384	154	5888
27	23	102	512	155	6016
30	24	103	640	156	6144
31	25	104	768	157	6272
32	26	105	896	160	6400
33	27	106	1024	161	6528
34	28	107	1152	162	6656
35	29	110	1280	163	6784
36	30	111	1408	164	6912
37	31	112	1536	165	7040
40	32	113	1664	166	7168
41	40	114	1792	167	7296
42	48	115	1920	170	7424
43	56	116	2048	171	7552
44	64	117	2176	172	7680
45	72	120	2304	173	7808
46	80	121	2432	174	7936
47	88	122	2560	175	8064
50	96	123	2688	176	8192
51	104	124	2816	177	8320
52	112	125	2944		

2. ACS STORAGE REGISTER

The ACS contains three storage registers. Three commands are available for selecting the storage register to be loaded with data (see Item 5.1 of Attachment B). The first and third registers contain data controlling the "Open Loop Precession." The second register contains data controlling the ΔV velocity correction and the time delay. After selecting a register, four commands are required to load the register. The register uses only seven bits from each command; the first bit is not used. Also, in contrast to the CDU storage registers as well as all qualitative commands, quantitative commands for the ACS storage registers have the "least significant bit" first and the "most significant bit" last. The registers are shown schematically as follows:

FIRST AND THIRD REGISTERS

Parameter Bit No.	Precession Magnitude 1 2 3 4 5 6 7 8 9 10 11	Redundant Magnitude 1 2 3 4 5 6 7 8	Precession Angle 1 2 3 4 5 6 7 8 9
Cmd. No. Bit No. of Cmd.	1 1 1 1 1 1 1 2 2 2 2 2 3 4 5 6 7 8 2 3 4 5	2 2 2 3 3 3 3 3 6 7 8 2 3 4 5 6	3 3 4 4 4 4 4 4 4 7 8 2 3 4 5 6 7 8
Bit Weighting* Resolution	0 1 2 3 4 5 6 7 8 9 10 8 secs.	0 1 2 3 4 5 6 7 64 secs.	0 1 2 3 4 5 6 7 8 360°/512

* Bit weighting gives the value of the bit expressed as powers of 2. The decimal value of the bit stream is then multiplied by the "resolution" to give the total value stored in the register.

SECOND REGISTER

Parameter Bit No.	ΔV Magnitude 1 2 3 4 5 6 7 8 9 10 11 12 13	Redundant Magnitude 1 2 3 4 5 6 7 8	Time Delay 1 2 3 4 5 6 7
Cmd. No. Bit No. of Cmd.	1 1 1 1 1 1 1 2 2 2 2 2 2 2 3 4 5 6 7 8 2 3 4 5 6 7	2 3 3 3 3 3 3 3 8 2 3 4 5 6 7 8	4 4 4 4 4 4 4 2 3 4 5 6 7 8
Bit Weighting* Resolution	0 1 2 3 4 5 6 7 8 9 10 11 12 1 sec.	0 1 2 3 4 5 6 7 32 secs.	0 1 2 3 4 5 6 64 secs.

* Bit weighting gives the value of the bit expressed as powers of 2. The decimal value of the bit stream is then multiplied by the "resolution" to give the total value stored in the register.

It is noted that in contrast to the quantitative command for the CDU storage registers, the resolution of each quantitative command for the ACS storage registers is the same throughout its range.

SAMPLE COMMAND MESSAGES

1. INTRODUCTION

Shown below are sample command messages covering the various cases to be encountered. For simplicity of reading, the indicated bit stream has a space between the "Preamble," "Sync," "Decoder Address," "Routing Address," "Command," and "Parity." However, when transmitted to the spacecraft there are no gaps in the bit stream; the message is transmitted at one bit per second. Generally speaking, the "Decoder Address" will not change from one command message to the next and, therefore, should be entered in a "general command" header into the computer preparing the command message. In the samples below, 01 has been selected for the "Decoder Address." The "Routing Address," "Command," and "Parity" differ for each command and, therefore, the corresponding bit stream must be separately generated by the computer in accordance with the selected "Command Designator." Each "Command Designator" has a unique arrangement of bits for the combined "Routing Address," "Command," and "Parity." In addition, a given combination of bits for "Routing Address" and "Command" will correspond to a single "Command Designator." Thus, the command will be entered into the computer using the "Command Designator."

2. REAL TIME COMMAND (EXCEPT DTU)

Title: Conscan On (Item 4.1.2 of Attachment B)
Designator: CNS9

Bit Stream: 0000 1 01 001 00010110 0101

3. REAL TIME COMMAND - DTU (ITEM 2 OF ATTACHMENT B)

Title: 256 bps (Item 2.2.5 of Attachment B)
Designator: BIT5

Bit Stream: 0000 1 01 100 00100100 0010

4. STORED COMMAND (EXCEPT DTU)

Title: Store in Register 3, Main Feed (A) Offset, Time Delay 4795 Sec.
Designator: CDS3, MFA9, 4795

Bit Stream:	1st.					Last
Cmd. (1)	0000	1	01	001	01010100	1000
Cmd. (2)	0000	1	01	010	10001000	0001
Cmd. (3)	0000	1	01	110	01100011	0011

Notes:

(1) The three messages should be transmitted in a single continuous stream.

(2) If desired, the first three zeros in Cmd. (2) and Cmd. (3) need not be generated. (See section 6.3.3.1 of PC-224.00, Contiguous Commands.)

(3) The Routing Addresses for the three commands are established by the command in Cmd. (1) which selects a Store Register and in Cmd. (2) which selects a "non-DTU" command.

(4) The resolution for time delay does not provide for 4800 sec. delay. The closest possible value is 4736 sec. and the computer shall select this value and generate the corresponding bit stream.

5. STORED COMMAND - DTU

Title: Store in Register 1, Telemetry Store Mode, Time Delay 7168 sec.
Designator: CDS1, TSM1, 7168

Bit Stream:	1st.					Last
Cmd. (1)	0000	1	01	001	01000010	0000
Cmd. (2)	0000	1	01	011	00110010	0110
Cmd. (3)	0000	1	01	110	01110110	1000

Notes:

(1) See Note in Section 4.

(2) The resolution of the time delay provides for 7168 sec.

6. STORED COMMAND - ACS STORAGE REGISTER 1

Title: Store in Register 1, Mag. 795 sec., Red. Mag. 891 sec., Angle 134.6°
Designator: ACS1, 795, 891, 134.6

Bit Stream:	1st.					Last
Cmd. (1)	0000	1	01	001	10011011	1000
Cmd. (2)	0000	1	01	101	01100011	0000
Cmd. (3)	0000	1	01	101	00000011	0000
Cmd. (4)	0000	1	01	101	01000011	1010
Cmd. (5)	0000	1	01	101	01111010	1111

Notes:

(1) The five messages should be transmitted in a single continuous stream.
(2) If desired, the first three zeros in Cmd. (2) through (5) need not be generated. (See Section 6.3.3.1 of PC-224.00, Contiguous Commands.)
(3) The Routing Addresses for the five commands are established by the command in Cmd. (1).
(4) The resolution of the parameters does not provide for the exact values desired. The closest values are 792 sec, 896 sec, and 134.3°; the computer shall select the closest value and generate the corresponding bit stream.

7. STORED COMMAND - ACS STORAGE REGISTER 2

Title: Store in Register 2, ΔV Mag. 4750, Red. Mag. 4757, Time Delay 3600 sec.
Designator: ACS2, 4750, 4757, 3600

Bit Stream:	1st.					Last
Cmd. (1)	0000	1	01	001	10100100	1100
Cmd. (2)	0000	1	01	101	00111000	0010
Cmd. (3)	0000	1	01	101	01010010	0011
Cmd. (4)	0000	1	01	101	01101001	1110
Cmd. (5)	0000	1	01	101	00001110	0000

Notes:

(1) See Notes (1) through (3) in Section 6.
(2) The resolution of the Redundant Magnitude and Time Delay parameters does not provide for the exact values desired. The correct values are 4800 and 3584, respectively; the computer shall select the closest value and generate the corresponding bit stream.

CDU COMMAND MATRIX

$\overline{G}\,\overline{H} \Rightarrow 00$

	$\overline{A}\,\overline{B}\,\overline{C}$ 0	$A\,\overline{B}\,\overline{C}$ 1	$\overline{A}\,B\,\overline{C}$ 2	$A\,B\,\overline{C}$ 3	$\overline{A}\,\overline{B}\,C$ 4	$A\,\overline{B}\,C$ 5	$\overline{A}\,B\,C$ 6	$A\,B\,C$ 7
$\overline{D}\,\overline{E}\,\overline{F}$ 00	NOT USED	RCVR. 1 TO HG ANTENNA HP	H	Sc	BATTERY DISCHARGER ENABLE	EXP J THRESHOLD LEVEL NORMAL	EXP H LOOK ANGLE STEP INHIBIT	CONSCAN 0° OUT
$\overline{D}\,\overline{E}\,F$ 10	C12 ACS SEQUENCE OVERRIDE	TWTA 1 TO HG ANTENNA HP	COMMAND PROCESSOR 1 SELECT	EXP B TARGET SUPR. ON/OFF	Sc	BATTERY HEATER ON	EXP J THRESHOLD LEVEL HIGH	EXP H START DATA THRESHOLD
$\overline{D}\,E\,\overline{F}$ 20		H	C14 ACS 31 MS PULSE LENGTH	COMMAND PROCESSOR 2 SELECT	EXP B DET. B H.V. STEP	C5 ACS PS&E OFF	CONSCAN POWER ON	EXP J MEDIUM B/W
$\overline{D}\,E\,F$ 30	EXP A RANGE DECREMENT	Sc	Sc			EXP B DETECTOR A H.V. STEP	C3 ACS DSL LOGIC 1 ON	
$D\,\overline{E}\,\overline{F}$ 40	EXP H START LOOK ANGLE RESET	EXP A RANGE INCREMENT	C1 ACS SEQUENCE STEP	BATTERY AUTO CHARGE	DTU REDUNDANCY A		EXP B FSM S/C REV. HV STEP	C4 ACS DSL LOGIC 2 ON
$D\,\overline{E}\,F$ 50		EXP H LOOK ANGLE INCREMENT	EXP A DATA INTERCHANGE	Sc	Sc	C18 ACS 2 SEC PULSE LENGTH		EXP B DATA SOURCE MODE.CCM
$D\,E\,\overline{F}$ 60	EXP B DETECTOR B H.V. LIMIT	CDU SEQUENCER INHIBIT	EXP H PMT GAIN INCREMENT	EXP A MANUAL RANGE ON/OFF		Sc	Sc	
$D\,E\,F$ 70	EXP I SPOKE ADVANCE	EXP B IN FLIGHT CALIBRATE	CDU SEQUENCER ENABLE	EXP H PMT GAIN DECREMENT		Sc	Sc	

Sc: Science Spare
H: High Level State
HP: High Level Pulse
S: Low Level State

CDU COMMAND MATRIX (Continued)

$\bar{G}\bar{H}$ ⇒ 00

DEF ⇓	$\bar{A}\bar{B}\bar{C}$ 0	$A\bar{B}\bar{C}$ 1	$\bar{A}B\bar{C}$ 2	$AB\bar{C}$ 3	$\bar{A}\bar{B}C$ 4	$A\bar{B}C$ 5	$\bar{A}BC$ 6	ABC 7
$\bar{D}\bar{E}\bar{F}$ 00	EXP G DEPLOY COVER	C13 ACS SEQUENCE RESET	CDU SELECT REG 1	EXP C PRIORITY MODE CONTROL	EXP H MODE 2 ON/OFF	EXP F HV MODE SELECT	EXP B DETECTOR A HV/CCM BIAS	
$\bar{D}E\bar{F}$ 10		EXP H MODE 3 ON/OFF	CDU START SEQUENCER	CDU SELECT REG 2	EXP C DISABLE DETECTOR D-2	BATTERY DISCHARGER DISABLE	EXP F LV MODE SELECT	EXP B DET B HV ON/OFF
$\bar{D}\bar{E}F$ 20	EXP H MODE 4 ON/OFF	EXP B LEFT CCM BIAS SUPPLY	TWTA 1 TO MG ANTENNA HP	CDU U-V NORMAL	CDU SELECT REG 3	EXP C LOGIC RESET	BATTERY HEATER OFF	EXP J STAR EXC. ENABLE/ DISABLE
$\bar{D}E\bar{F}$ 30	EXP E CALIBRATE		EXP B INTEGR. PERIOD	CONSCAN POWER OFF	SIGNAL PRESENT RESET	CDU SELECT REG 4	EXP C DISABLE DETECTOR D-7	
$\bar{D}\bar{E}\bar{F}$ 40	EXP B RIGHT CCM BIAS SUPPLY	EXP E MODE SELECT	EXP H SPOKE ADVANCE FINE	C16 ACS 0.5 SEC PULSE	CONSCAN 180° OUT	H	CDU SELECT REG 5	EXP C CALIBRATE
$\bar{D}E\bar{F}$ 50	EXP C DISABLE DETECTOR D-1	Sc	Sc	EXP H SPOKE ADVANCE COARSE	H	DTU REDUNDANCY B	Sc	Sc
$\bar{D}E\bar{F}$ 60	Sc	EXP J NARROW B/W	EXP D REDUNDANT LOGIC SELECT		BATTERY FLOAT CHARGE	EXP H LOOK ANGLE REVERSE STEP	C2 ACS PS&E REG ON	
$DE\bar{F}$ 70	Sc	Sc	EXP J WIDE B/W		EXP J DATA READOUT EN/DIS	Sc	EXP H LOW SAMPLE RATE	

Sc: Science Spare
H: High Level State
HP: High Level Pulse
S: Low Level State

CDU COMMAND MATRIX (Continued)

$\bar{G}H \Rightarrow 20\bar{0}$

	$\bar{A}\bar{B}\bar{C}$ 0	$\bar{A}\bar{B}C$ 1	$\bar{A}B\bar{C}$ 2	$\bar{A}BC$ 3	$A\bar{B}\bar{C}$ 4	$A\bar{B}C$ 5	$AB\bar{C}$ 6	ABC 7
$\bar{D}\bar{E}\bar{F}$ 00	RTGS DEPLOY	ANT FEED NORMAL (RQNT) H	SIGNAL PRESENT OVERRIDE	EXP C POWER OFF S	RESET RTG INV RELAYS	C30 ACS REAL TIME ΔV PULSE	C36 ACS CLOCK RESET	SPARE EXP
$\bar{D}\bar{E}F$ 10	ANT FEED OFFSET H	BOOM DEPLOY	COHERENT MODE INHIB RCVR 2 S	C15 ACS 125 MS PULSE	EXP A POWER OFF S		C31 ACS REAL TIME ΔV SCT	
$\bar{D}E\bar{F}$ 20	EXP G POWER OFF S		C10 ACS SEQUENCE EXECUTE	XMTR 1 ON S	C23 ACS SPIN RATE INCREASE	EXP B POWER OFF S	COH MODE NON INHIB RCVR 2 S	C32 ACS REAL TIME SPIN PULSE
$\bar{D}EF$ 30	C25 ACS 0° PREC PULSE	EXP D POWER OFF OFF S	XMTR 1 OFF S	C7 ACS ARM REG 1	CONSCAN THRESHOLD HIGH	RCVR 1 TO HG ANTENNA HP	EXP E POWER OFF S	
$D\bar{E}\bar{F}$ 40		C26 ACS 90° PREC PULSE	EXP SPARE POWER OFF S	ORDNANCE ARM HP	C8 ACS ARM REG 2	C40 ACS STAR DELAY UP	UNDER VOLTAGE INHIBIT	EXP J POWER OF OFF S
$D\bar{E}F$ 50	Sc	Sc	C27 ACS 180° PREC PULSE	EXP F POWER OFF S	EXECUTE STORED SEQUENCE	C9 ACS ARM REG 3	C42 ACS STAR GATE ENABLE	EXP K POWER OFF S
$\bar{D}EF$ 60			SELECT CDU +5V BUS A HP	C26 ACS 270° PREC PULSE	EXP H POWER OFF S		C19 ACS PREC PAIR 1	RESET SLOT SELECT REGISTERS
DEF 70					C29 ACS REAL TIME PREC PULSE	EXP I POWER OFF S		C21 ACS ΔV PAIR 1

Sc: Science Spare
H: High Level State
HP: High Level Pulse
S: Low Level State

CDU COMMAND MATRIX (Continued)

GH ⇒ 300

DEF	$\bar{A}\bar{B}\bar{C}$ 0	$\bar{A}\bar{B}C$ 1	$\bar{A}B\bar{C}$ 2	$\bar{A}BC$ 3	$A\bar{B}\bar{C}$ 4	$A\bar{B}C$ 5	$AB\bar{C}$ 6	ABC 7
$\bar{D}\bar{E}\bar{F}$ 00	RTGS DEPLOY REDNT	C17 ACS 1.0 SEC PULSE LENGTH	EXP A POWER ON S	C37 ACS SELECT SUN A				
$\bar{D}\bar{E}F$ 10	XMTR 2 ON S	BOOM DEPLOY REDNT	C24 ACS SPIN RATE DECREASE	EXP B POWER ON S	C36 ACS SELECT SUN B	SELECT CDU +5V BUS B HP		
$\bar{D}E\bar{F}$ 20	EXP F POWER ON S	CONSCAN THRESHOLD LOW	C20 ACS PREC PAIR 2	C41 ACS STAR DELAY DOWN LOWER 1/2	EXP E POWER ON S	C39 ACS SELECT STAR		
$\bar{D}EF$ 30	C35 ACS DESPIN START	XMTR 2 OFF S	STOP CDU STORED SEQUENCE	C22 ACS ΔV PAIR 2	C43 ACS STAR GATE DISABLE	EXP C POWER ON S	C44 ACS TIME FREQUENCY SELECT A	C50 ACS STAR DELAY DOWN UPPER 1/2
$D\bar{E}\bar{F}$ 40	C46 ACS STAR TIME GATE ENABLE		EXP I POWER ON S	RCVR 1 COHO MODE NONINHIBIT S	ALL EXP POWER OFF S	C11 ACS SEQUENCE INHIBIT	EXP D POWER ON S	C45 ACS TIME FREQUENCY SELECT B
$D\bar{E}F$ 50		C47 ACS STAR TIME GATE UP	C33 ACS CLOSED LOOP START	EXP J POWER ON S	RCVR 1 COHO MODE INHIBIT S	RESET CDU MEMORY REGISTERS	ANTENNA FEED NORMAL H	EXP POWER ON SPARE Sc
$DE\bar{F}$ 60		EXP G POWER ON S	C48 ACS STAR TIME GATE DOWN	C34 ACS CLOSED LOOP STOP	ANTENNA FEED OFFSET REDNT H	ORDNANCE SAFE HP	C6 ACS SPARE	
DEF 70			EXP H POWER ON S	C49 ACS STAR DELAY UP UPPER 1/2		EXP K POWER ON S		

Sc: Science Spare
H: High Level State
HP: High Level Pulse
S: Low Level State

APPENDIX B

PIONNEER TELEMETRY

TELEMETRY DESIGNATORS

The following mnemonics are to be used as the designator for the given telemetry measurement. Reference to, display of or printout of these measurements should be abbreviated to use these mnemonics. The enclosed list is by spacecraft subsystem, not alphabet.

References:
A. TRW Doc. No. C311292 "Pioneer Telemetry Requirements" dated July 29, 1970.
B. TRW Doc. 8220.9-378 "Pioneer F/G Engineering Programming Requirements Document" dated June 5, 1970.
C. PPO Memo PAS-0-336

1.0 ELECTRICAL DISTRIBUTION SUBSYSTEM

1.1 Boom Release Status (C-120) - MAGB

1.2 RTG Deploy Status (C-120) - R#

1.3 Ordnance Safe/Arm (C-128) - ORD S/A or ORD #

1.4 Command Memory Status (C-324) - MCMD

1.5 Command Execute (C-324) - CEX

1.6 Sequencer Status (C-132) - SQE

1.7 Sequencer Power Status (C-332) - SQP

1.8 Command Processor Memory Status (C-332) - MCP

1.9 Overload Protection Status (C-332) - OP

1.10 Separation Status (C-120) - SEP

1.11 Storage Memory Contents Command (C-306) - SMCCMD

1.12 Storage Memory Contents Time (C-305) - SMCTIME

1.13 Storage Memory Contents Register ID (C-307) - SMCRID

1.14 Storage Memory Contents DTU ID (C-332) SMCDID

1.15 CDU + 5 DC Bus Status (C-332) - CDU5A/CDU5B

1.16 RTG 1/2 (3/4) Ordnance Status (C-132) - ORD 1/2(3/4)

1.17 RTG 1/2 (3/4) Redundant Ordnance Status (C-202) - ORD 1/2(3/4)R

1.18 Receiver Reverse Inhibit Status (C-332) - RCVIN

2.0 DATA HANDLING SUBSYSTEM

2.1 A/D Calibration Voltage (C-101, 102, 103) - Low - CAL1
Med - CAL2
High - CAL3

2.2 Decoder Status (C-120) - DDA(DDB)

2.3 Convoluntional Coder Generator Status (C-132) - CCG

2.4 Roll Attitude (C-112, 116) - RAT

2.5 Extended Subcomm ID (C-104) - ESCID

2.6 Spin Period (C-405, 406, 407) - SPPD

2.7 Phase Error (C-408) - PHER

2.8 Roll Reference (C-417) - RLRF

2.9 Spin Averaging Mode (C-417) - SPAVE

2.10 ACS Operation (C-417) - ACS

3.0 THERMAL SUBSYSTEM

3.1 -X Axis, Platform Temp - Equip. (C-301) - PLT 1

3.2 -X, +Y, Platform Temp - Equip. (C-302) - PLT 2

3.3 +X, -Y, Platform Temp - Equip. (C-304) - PLT 3

3.4 +X Axis Platform Temp - Equip. (C-318) - PLT 4

3.5 +X Axis Platform Temp - Equip. (C-319) - PLT 5

3.6 -Y Axis Platform Temp - Equip. (C-320) - PLT 6

3.7 RTG No.# Fin Root Temp (C-201, 202, 203, 204) - RTG #FT

3.8 RTG No. # Hot Junction Temp (C-220, 219, 218, 217) - RTG #JT

4.0 ATTITUDE CONTROL SUBSYSTEM

4.1 SRA Temperature (C-303) - SRAT

4.2 SSA Temperature (C-317) - SSAT

4.3 Precession Pair (C-403) - PRECPR

4.4 Pulse Length (C-403) - PULSLNG

4.5 Delta V - Pair (C-403) - D/VPR

4.6 Spin Control Direction (C-403) - SPDIR

4.7 Despin Enable Status (C-410) - DSP

4.8 Conscan Enable Status (C-410) - CNS

4.9 Clock Select (C-410) - CLK A/B

4.10 Angle Gate Enable Status (C-410) - STRGT

4.11 Star Level (C-410) - STRLVL

4.12 Star Delay (C-419, 420) - STRDLY

4.13 Star Count (C-421) - STRCNT

4.14 Power Status (C-421) - PWR

4.15 Star Location (C-432) - STRLOC

4.16 Inhibit State (C-432) - INH

4.17 Arm Status (C-432) - ARM

4.18 Sequence Status (C-431) - SEQ

4.19 Time Gate Enable Status (C-431) - TGEN

4.20 Reference Select (C-431) - REFSL

4.21 No. 1(2) Precession Magnitude Register (C-411, 412(C-415, 416)) - PMAG 1(2)

4.22 Delta V - Magnitude Register (C-412, 413, 414) - D/V MAG

4.23 Star Coincidence Indication (C-416) - STRCNC

4.24 Time Gate Register (C-404) - TGREG

4.25 No. 1(2) Precession Redundant Magnitude Register (C-428, 429 (C-422, 423, 424)) - PMAG 1(2)R

4.26 No. 1(2) Precession Angle Register (C-429, 430(C-424, 425)) - PRANG 1(2)

4.27 Delta V - Redundant Magnitude Register (C-425, 426) - D/V MAGR

4.28 Delay Register (C-427, 428) - DLYREG

4.29 Delta V/Spin Thruster Mode Status (C-404) - D/V SCT

5.0 POWER SUBSYSTEM

5.1 RTG # Current (C-127, 105, 114, 123) - RTG #I

5.2 Shunt Bus Current (C-122, 209) - SHNT I

5.3 DC Bus Current (C-129) - BUS I

5.4 Battery Charge Current (C-109) - BAT CHG I

5.5 Battery Discharge Current (C-126) - BAT DCH I

5.6 RTG Voltage - RTG # (C-110, 125, 131, 113) - RTG #V

5.7 Battery Voltage (C-106) - BAT V

5.8 DC Bus Voltage - Expanded (C-107, 119) - BUS V

5.9 Battery Temp. (C-115) - BAT T

5.10 Battery Charge Status (C-128) - BCH

5.11 Battery Discharge Status (C-128) - BDCH

5.12 TRF +5V CDU Output (C-117, 118) - CD5A V(CD5B V)

6.0 COMMUNICATIONS SUBSYSTEM

6.1 Loop Stress, Receiver A (B) - (C-212(C-229)) - LOOPA(LOOPB)

6.2 Signal Strength, Receiver A(B) - (C-232(C-213)) - STRA(STRB)

6.3 Oscillator Temp, Receiver A(B) - (C-222(C-227)) - RCVAT(RCVBT)

6.4 Auxiliary Oscillator Temp, Driver A(B) - (C-206(C-223)) - DRVAT(DRVBT)

6.5 Output Power Monitor, TWT A(B) - (C-231(C-214)) - PWRA W(PWRB W)

6.6 Helix Current, TWTA(B) - (C-211(C-216)) - HLZA I(HLXB I)

6.7 Cathode Current, TWT A/B - (C-208(C-215)) - CTHA I(CTHB I)

6.8 Reference Voltage, TWT A/B (C-224(C-230)) - REFA V(REFB V)

6.9 Converter Temp. TWT A/B (C-207(C-221)) - TWTACT(TWTBCT)

6.10 TWT A/B Temp. - (C-206(C-228)) - TWTAT(TWTBT)

6.11 Signal Present, Receiver A/B, (C-308) - SIGA(SIGB)

6.12 Oscillator Enable/Disable, Receiver A/B, (C-308) - RCVAO(RCVBO)

6.13 Conscan Threshold Status (C-316) - CNSTH

6.14 Conscan Power Status (C316) - CNSPW

6.15 Conscan Threshold Mode Status (C-313) - CSNTM

6.16 Conscan AGC, Receiver A/B, (C 111(C-121)) - CAGCA (CAGCB)

6.17 Conscan Phase Output Status (C-313) - CNSPO

6.18 A Sin θ (C-313, 314) - ASIN

6.19 A Cos θ (C-314, 315) - ACOS

7.0 ANTENNA SUBSYSTEM

7.1 Receiver Switch Position (C-316) - RCVX

7.2 Transmitter Switch Position (C-108) - XMTX

7.3 Antenna Feed Switch Position (C-316) - ANTX

8.0 PROPULSION SUBSYSTEM

8.1 Propellant Supply Pressure (C-210) - TNKP

8.2 Propellant Supply Temp. (C-327) - TNKT

8.3 Velocity Thruster Cluster 1/2 Temp. (C-309(C-326)) - VTC1T(VTC2T)

8.4 Spin Thruster Cluster Temp. (C-310) - SP CT

8.5 Velocity Thruster 1A Pulse Count (C-329) - VT 1A CNT

8.6 Velocity Thruster 1B Pulse Count (C-321) - VT 1B CNT

8.7 Velocity Thruster 2A Pulse Count (C-322) - VT 2A CNT

8.8 Velocity Thruster 2B Pulse Count (C-330) - VT 2B CNT

8.9 Spin Control Thruster A Pulse Count (C-308) - SPTA CNT

8.10 Spin Control Thruster B Pulse Count (C-308) - SPTB CNT

8.11 Thruster Initiation Status (C-409) - VPT 1A
- VPT 2A
- VPT 1B
- VPT 2B
- SCT 1
- SCT 2

8.12 Velocity Thruster Temp. (*) - VT 1AT
(*) - VT 2AT
(*) - VT 1BT
(*) - VT 2BT

* To Be Determined

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
		ELECTRICAL DISTRIBUTION SUBSYSTEM						
C-008	1	H V MAGNETOMETER EXP POWER STATUS	B	OFF/ON		03 01	T067	
C-008	2	PLASMA ANALYZER EXP POWER STATUS	B	OFF/ON		03 01	T068	
C-008	3	CHARGED PARTICLE EXP POWER STATUS	B	OFF/ON		03 01	T070	
C-008	4	GEIGER TUBE TEL EXP POWER STATUS	B	OFF/ON		03 01	T071	
C-008	5	COSMIC RAY EXP POWER STATUS	B	OFF/ON		03 01	T069	
C-020	1	MAGNETOMETER BOOM STATUS	B	RELEASE/STOW		01 79	T047	
C-020	2	RTG 1-2 DEPLOY STATUS	B	DEPLOY/STOW		01 80	T048	
C-020	3	RTG 3-4 DEPLOY STATUS	B	DEPLOY/STOW		01 81	T049	
C-020	4	S/C SEPARATION STATUS	B	SEP/ATTACH		03 01	T054	
C-024	1	TRAPPED RADIATION EXP POWER STATUS	B	OFF/ON		03 01	T072	
C-024	2	ULTRAVIOLET PHOTO EXP POWER STATUS	B	OFF/ON		03 01	T073	
C-024	3	IMAGING PHOTO POLAR EXP POWER STATUS	B	OFF/ON		03 01	T074	
C-024	4	INFRARED RADIOMETER EXP POWER STATUS	B	OFF/ON		03 01	T075	
C-024	5	ASTEROID/METEROID EXP POWER STATUS	B	OFF/ON		03 01	T076	
C-024	6	METEOROID DETECTOR EXP POWER STATUS	B	OFF/ON		03 01	T077	
C-028	5	ARM/SAFE RELAY STATUS PRIME	B	SAFE/ARM		03 01	T130	
C-028	6	ARM/SAFE RELAY STATUS REDNT	B	SAFE/ARM		03 01	T131	
C-032	2	ORDNANCE SAFE/ARM	B	SAFE/ARM		03 01	T050	
C-032	3	ORDNANCE SAFE/ARM	B	SAFE/ARM		03 01	T051	
C-032	4	ORDNANCE SAFE/ARM	B	SAFE/ARM		03 01	T052	
C-032	5	ORDNANCE SAFE/ARM	B	SAFE/ARM		03 01	T053	
C-032	6	SEQUENCER STATUS	B	DIS/ENABLE		03 01	T089	
C-069		STORAGE MEM CONTENTS TIME 7 BITS	D			03 01	T100	C
C-070		STORAGE MEM CONTENTS CMD 8 BITS	D			03 01	T100	C
C-071		STORAGE MEM CONTENTS REGISTER ID (3)	D			03 01	T100	C
C-088	1	COMMAND MEMORY STATUS	B	PROC/STANDBY		03 01	T109	
C-088	6	COMMAND EXECUTE	B			03 01	T110	
C-096	1	SEQUENCER POWER STATUS	B	OFF/ON		03 01	T113	
C-096	2	OVERLOAD PROTECTION STATUS	B	ON/OFF		03 01	T114	C
C-096	3	RCVR REVERSE STATUS	B	REV/INHIBIT		03 01	T115	
C-096	4	CMD PROCESSOR MEMORY STATUS	B	B ON/A ON		03 01	T116	
C-096	5	STORAGE MEM CONTENTS DTU ID	B	NO/YES		03 01	T117	C
C-096	6	CDU +5DC BUS STATUS	B	CH B/CH A		03 01	T118	C

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
		DHS SUBSYSTEM						
C-001		A/D CAL VOLTAGE LOW	A	168 MV		06 03		C
C-002		A/D CAL VOLTAGE MED	A	1.512VOLTS		06 03		C
C-003		A/D CAL VOLTAGE HIGH	A	2.424 VOLTS		06 03		C
C-004		EXT S.C. ID	D			06 03		
C-012		ROLL ATTITUDE	D			06 03		
C-016		ROLL ATTITUDE	D			06 03		
C-020	5	DECODER A STATUS	B	OFF/ON		06 17 A	T128	
C-020	6	DECODER B STATUS	B	OFF/ON		06 17 B	T129	
C-032	1	CONV CODE GEN STATUS	B	OFF/ON		06 03		
C-101		SPIN PERIOD	D			06 03		C
C-102		SPIN PERIOD	D			06 03		C
C-103		SPIN PERIOD	D			06 03		C
C-104		PHASE ERROR	D			06 03		C
C-113	1	ROLL REFERENCE	B	0/180 DEG		06 03		
C-113	2	SPIN AVERAGING MODE	B	NO/YES		06 03		
C-113	3	ACS OPERATION	B	NO/YES		06 03		

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
		THERMAL						
C-033		RTG NO. 1 FIN ROOT TEMPERATURE	A	160 TO 360 DEG F		04 05	T059	
C-034		RTG NO. 2 FIN ROOT TEMPERATURE	A	160 TO 360 DEG F		04 05	T060	
C-035		RTG NO. 3 FIN ROOT TEMPERATURE	A	160 TO 360 DEG F		04 06	T058	
C-036		RTG NO. 4 FIN ROOT TEMPERATURE	A	160 TO 360 DEG F		04 06	T055	
C-049		RTG NO. 4 HOT JUNCTION TEMPERATURE	A	880 TO 1200 DEG F		04 06	T057	
C-050		RTG NO. 3 HOT JUNCTION TEMPERATURE	A	880 TO 1200 DEG F		04 06	T056	
C-051		RTG NO. 2 HOT JUNCTION TEMPERATURE	A	880 TO 1200 DEG F		04 05	T062	
C-052		RTG NO. 1 HOT JUNCTION TEMPERATURE	A	880 TO 1200 DEG F		04 05	T061	
C-065		TEMP HEX PLATFORM -X AXIS	A	0 TO 140 DEG F		03 01	T078	
C-066		TEMP HEX PLATFORM -X,+Y	A	0 TO 140 DEG F		03 01	T079	
C-068		TEMP HEX PLATFORM +X,-Y	A	0 TO 140 DEG F		03 01	T080	
C-082		TEMP HEX PLATFORM +X AXIS	A	0 TO 140 DEG F		03 01	T081	
C-083		TEMP EXP PLATFORM +X AXIS	A	-20 TO 110 DEG F		03 01	T082	
C-084		TEMP EXP PLATFORM -Y AXIS	A	-20 TO 110 DEG F		03 01	T083	

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
		ATTITUDE CONTROL SUBSYSTEM						
C-067		SRA TEMPERATURE	A			02 30	T064	
C-081		SSA TEMPERATURE	A			02 30	T065	
C-099	1	PRECESSION PAIR	D			02 30	T119	
C-099	2	PULSE LENGTH BIT NO. 1	D			02 30	T119	C
C-099	3	PULSE LENGTH BIT NO. 2	D			02 30	T119	C
C-099	4	PULSE LENGTH BIT NO. 3	D			02 30	T119	C
C-099	5	DELTA V-PAIR	D			02 30	T119	C
C-099	6	SPIN CONTROL DIRECTION	D			02 30	T119	
C-100	1	TIME GATE REGISTER BIT NO. 1	D			02 30	T119	
C-100	2	TIME GATE REGISTER BIT NO. 2	D			02 30	T119	
C-100	3	TIME GATE REGISTER BIT NO. 3	D			02 30	T119	
C-100	4	TIME GATE REGISTER BIT NO. 4	D			02 30	T119	
C-100	5	TIME GATE REGISTER BIT NO. 5	D			02 30	T119	
C-100	6	DELTA V SCT MODE	D			02 30	T119	
C-106	1	DESPIN ON	D			02 30	T119	
C-106	2	CONSCAN ON	D			02 30	T119	
C-106	3	CLOCK SELECT BIT NO. 1	D			02 30	T119	
C-106	4	CLOCK SELECT BIT NO. 2	D			02 30	T119	
C-106	5	STAR ANGLE GATE STATUS	D	45 DEG/360 DEG		02 30	T119	C
C-106	6	STAR LEVEL 180 DEG CANOPUS	D	YES/NO		02 30	T119	C
C-107	1	NO. 1 PRECESSION MAGNITUDE REG BIT 1	D			02 30	T119	C
C-107	2	NO. 1 PRECESSION MAGNITUDE REG BIT 2	D			02 30	T119	C
C-107	3	NO. 1 PRECESSION MAGNITUDE REG BIT 3	D			02 30	T119	C
C-107	4	NO. 1 PRECESSION MAGNITUDE REG BIT 4	D			02 30	T119	C
C-107	5	NO. 1 PRECESSION MAGNITUDE REG BIT 5	D			02 30	T119	C
C-107	6	NO. 1 PRECESSION MAGNITUDE REG BIT 6	D			02 30	T119	C
C-108	1	NO. 1 PRECESSION MAGNITUDE REG BIT 7	D			02 30	T119	C
C-108	2	NO. 1 PRECESSION MAGNITUDE REG BIT 8	D			02 30	T119	C
C-108	3	NO. 1 PRECESSION MAGNITUDE REG BIT 9	D			02 30	T119	C
C-108	4	NO 1 PRECESSION MAGNITUDE REG BIT 10	D			02 30	T119	C
C-108	5	NO 1 PRECESSION MAGNITUDE REG BIT 11	D			02 30	T119	C
C-108	6	DELTA V-MAGNITUDE REG BIT NO. 1	D			02 30	T119	C
C-109	1	DELTA V-MAGNITUDE REG BIT NO. 2	D			02 30	T119	C
C-109	2	DELTA V-MAGNITUDE REG BIT NO. 3	D			02 30	T119	C
C-109	3	DELTA V-MAGNITUDE REG BIT NO. 4	D			02 30	T119	C

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
		ATTITUDE CONTROL SUBSYSTEM CONT						
C-109	4	DELTA V-MAGNITUDE REG BIT NO. 5	D			02 30	T119	C
C-109	5	DELTA V-MAGNITUDE REG BIT NO. 6	D			02 30	T119	C
C-109	6	DELTA V-MAGNITUDE REG BIT NO. 7	D			02 30	T119	C
C-110	1	DELTA V-MAGNITUDE REG BIT NO. 8	D			02 30	T119	C
C-110	2	DELTA V-MAGNITUDE REG BIT NO. 9	D			02 30	T119	C
C-110	3	DELTA V-MAGNITUDE REG BIT NO. 10	D			02 30	T119	C
C-110	4	DELTA V-MAGNITUDE REG BIT NO. 11	D			02 30	T119	C
C-110	5	DELTA V-MAGNITUDE REG BIT NO. 12	D			02 30	T119	C
C-110	6	DELTA V-MAGNITUDE REG BIT NO. 13	D			02 30	T119	C
C-111	1	NO. 2 PRECESSION MAGNITUDE REG BIT 1	D			02 30	T119	C
C-111	2	NO. 2 PRECESSION MAGNITUDE REG BIT 2	D			02 30	T119	C
C-111	3	NO. 2 PRECESSION MAGNITUDE REG BIT 3	D			02 30	T119	C
C-111	4	NO. 2 PRECESSION MAGNITUDE REG BIT 4	D			02 30	T119	C
C-111	5	NO. 2 PRECESSION MAGNITUDE REG BIT 5	D			02 30	T119	C
C-111	6	NO. 2 PRECESSION MAGNITUDE REG BIT 6	D			02 30	T119	C
C-112	1	NO. 2 PRECESSION MAGNITUDE REG BIT 7	D			02 30	T119	C
C-112	2	NO. 2 PRECESSION MAGNITUDE REG BIT 8	D			02 30	T119	C
C-112	3	NO. 2 PRECESSION MAGNITUDE REG BIT 9	D			02 30	T119	C
C-112	4	NO 2 PRECESSION MAGNITUDE REG BIT 10	D			02 30	T119	C
C-112	5	NO 2 PRECESSION MAGNITUDE REG BIT 11	D			02 30	T119	C
C-112	6	STAR COINCIDENCE INDICATION	D	NO/YES		02 30	T119	
C-115	1	STAR DELAY BIT NO. 1	D			02 30	T119	C
C-115	2	STAR DELAY BIT NO. 2	D			02 30	T119	C
C-115	3	STAR DELAY BIT NO. 3	D			02 30	T119	C
C-115	4	STAR DELAY BIT NO. 4	D			02 30	T119	C
C-115	5	STAR DELAY BIT NO. 5	D			02 30	T119	C
C-115	6	STAR DELAY BIT NO. 6	D			02 30	T119	C
C-116	1	STAR DELAY BIT NO. 7	D			02 30	T119	C
C-116	2	STAR DELAY BIT NO. 8	D			02 30	T119	C
C-116	3	STAR DELAY BIT NO. 9	D			02 30	T119	C
C-116	4	STAR DELAY BIT NO. 10	D			02 30	T119	C
C-116	5	STAR DELAY BIT NO. 11	D			02 30	T119	C
C-116	6	STAR DELAY BIT NO. 12	D			02 30	T119	C
C-117	1	STAR COUNT BIT NO. 1	D			02 30	T119	C
C-117	2	STAR COUNT BIT NO. 2	D			02 30	T119	C
C-117	3	STAR COUNT BIT NO. 3	D			02 30	T119	C

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
		ATTITUDE CONTROL SUBSYSTEM CONT						
C-117	4	POWER STATUS BIT NO. 1	D			02 30	T119	
C-117	5	POWER STATUS BIT NO. 2	D			02 30	T119	
C-117	6	POWER STATUS BIT NO. 3	D			02 30	T119	
C-118	1	SPACING BITS	D			02 30	T119	
C-118	2	SPACING BITS	D			02 30	T119	
C-118	3	SPACING BITS	D			02 30	T119	
C-118	4	SPACING BITS	D			02 30	T119	
C-118	5	SPACING BITS	D			02 30	T119	
C-118	6	NO 2 PREC RDNT MAG REG BIT NO 1	D			02 30	T119	
C-119	1	NO 2 PREC RDNT MAG REG BIT NO 2	D			02 30	T119	
C-119	2	NO 2 PREC RDNT MAG REG BIT NO 3	D			02 30	T119	
C-119	3	NO 2 PREC RDNT MAG REG BIT NO 4	D			02 30	T119	
C-119	4	NO 2 PREC RDNT MAG REG BIT NO 5	D			02 30	T119	
C-119	5	NO 2 PREC RDNT MAG REG BIT NO 6	D			02 30	T119	
C-119	6	NO 2 PREC RDNT MAG REG BIT NO 7	D			02 30	T119	
C-120	1	NO 2 PREC RDNT MAG REG BIT NO 8	D			02 30	T119	
C-120	2	NO. 2 PREC ANGLE REGISTER BIT NO. 1	D			02 30	T119	
C-120	3	NO. 2 PREC ANGLE REGISTER BIT NO. 2	D			02 30	T119	
C-120	4	NO. 2 PREC ANGLE REGISTER BIT NO. 3	D			02 30	T119	
C-120	5	NO. 2 PREC ANGLE REGISTER BIT NO. 4	D			02 30	T119	
C-120	6	NO. 2 PREC ANGLE REGISTER BIT NO. 5	D			02 30	T119	
C-121	1	NO. 2 PREC ANGLE REGISTER BIT NO. 6	D			02 30	T119	
C-121	2	NO. 2 PREC ANGLE REGISTER BIT NO. 7	D			02 30	T119	
C-121	3	NO. 2 PREC ANGLE REGISTER BIT NO. 8	D			02 30	T119	
C-121	4	NO. 2 PREC ANGLE REGISTER BIT NO. 9	D			02 30	T119	
C-121	5	DELTA V-REDUNDANT MAG REG BIT NO. 1	D			02 30	T119	
C-121	6	DELTA V-REDUNDANT MAG REG BIT NO. 2	D			02 30	T119	
C-122	1	DELTA V-REDUNDANT MAG REG BIT NO. 3	D			02 30	T119	
C-122	2	DELTA V-REDUNDANT MAG REG BIT NO. 4	D			02 30	T119	
C-122	3	DELTA V-REDUNDANT MAG REG BIT NO. 5	D			02 30	T119	
C-122	4	DELTA V-REDUNDANT MAG REG BIT NO. 6	D			02 30	T119	
C-122	5	DELTA V-REDUNDANT MAG REG BIT NO. 7	D			02 30	T119	
C-122	6	DELTA V-REDUNDANT MAG REG BIT NO. 8	D			02 30	T119	
C-123	1	DELAY REGISTER BIT NO. 1	D			02 30	T119	
C-123	2	DELAY REGISTER BIT NO. 2	D			02 30	T119	
C-123	3	DELAY REGISTER BIT NO. 3	D			02 30	T119	

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
		ATTITUDE CONTROL SUBSYSTEM CONT						
C-123	4	DELAY REGISTER BIT NO. 4	D			02 30	T119	
C-123	5	DELAY REGISTER BIT NO. 5	D			02 30	T119	
C-123	6	DELAY REGISTER BIT NO. 6	D			02 30	T119	
C-124	1	DELAY REGISTER BIT NO. 7	D			02 30	T119	
C-124	2	NO. 1 PREC RDNT MAG REG BIT NO. 1	D			02 30	T119	
C-124	3	NO. 1 PREC RDNT MAG REG BIT NO. 2	D			02 30	T119	
C-124	4	NO. 1 PREC RDNT MAG REG BIT NO. 3	D			02 30	T119	
C-124	5	NO. 1 PREC RDNT MAG REG BIT NO. 4	D			02 30	T119	
C-124	6	NO. 1 PREC RDNT MAG REG BIT NO. 5	D			02 30	T119	
C-125	1	NO. 1 PREC RDNT MAG REG BIT NO. 6	D			02 30	T119	
C-125	2	NO. 1 PREC RDNT MAG REG BIT NO. 7	D			02 30	T119	
C-125	3	NO. 1 PREC RDNT MAG REG BIT NO. 8	D			02 30	T119	
C-125	4	NO. 1 PREC ANGLE REGISTER BIT NO. 1	D			02 30	T119	
C-125	5	NO. 1 PREC ANGLE REGISTER BIT NO. 2	D			02 30	T119	
C-125	6	NO. 1 PREC ANGLE REGISTER BIT NO. 3	D			02 30	T119	
C-126	1	NO. 1 PREC ANGLE REGISTER BIT NO. 4	D			02 30	T119	
C-126	2	NO. 1 PREC ANGLE REGISTER BIT NO. 5	D			02 30	T119	
C-126	3	NO. 1 PREC ANGLE REGISTER BIT NO. 6	D			02 30	T119	
C-126	4	NO. 1 PREC ANGLE REGISTER BIT NO. 7	D			02 30	T119	
C-126	5	NO. 1 PREC ANGLE REGISTER BIT NO. 8	D			02 30	T119	
C-126	6	NO. 1 PREC ANGLE REGISTER BIT NO. 9	D			02 30	T119	
C-127	1	SEQUENCE STATUS BIT NO. 1	D			02 30	T119	C
C-127	2	SEQUENCE STATUS BIT NO. 2	D			02 03	T119	C
C-127	3	SEQUENCE STATUS BIT NO. 3	D			02 03	T119	C
C-127	4	STAR TIME GATE ENABLE STATUS	D	ENABLE/DISABLE		02 30	T119	
C-127	5	REFERENCE SELECT BIT NO. 1	D			02 30	T119	
C-127	6	REFERENCE SELECT BIT NO. 2	D			02 30	T119	
C-128	1	STAR LOCATION BIT NO. 1	D			02 30	T119	
C-128	2	STAR LOCATION BIT NO. 2	D			02 30	T119	
C-128	3	ACS REG INHIBIT STATE	D	ENABLE/DISABLE		02 30	T119	
C-128	4	ARM STATUS BIT NO. 1	D			02 30	T119	
C-128	5	ARM STATUS BIT NO. 2	D			02 30	T119	
C-128	6	ARM STATUS BIT NO. 3	D			02 30	T119	

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
		POWER SUBSYSTEM						
C-005		RTG 2 CURRENT	A	0 TO 10 AMPS		04 05	T037	
C-006		BATTERY VOLTAGE	A	0-15 VDC		04 02	T046	
C-007		DC BUS VOLTAGE - EXPANDED	A	26-30 VDC		04 02	T029	
C-009		BATTERY CHARGE CURRENT	A	0-0.3 AMPS		04 02	T040	
C-010		RTG VOLTAGE - RTG 1	A	0-6 VDC		04 05	T042	
C-013		RTG VOLTAGE - RTG 4	A	0-6 VDC		04 06	T045	
C-014		RTG 3 CURRENT	A	0 TO 10 AMPS		04 06	T034	
C-015		BATTERY TEMPERATURE	A	-20 TO +120 DEG		04 02	T031	
C-017		TRF +5V CDU OUTPUT CH A	A	0-6V		04 14	T087	
C-018		TRF +5V CDU OUTPUT CH B	A	0-6V		04 14	T088	
C-019		DC BUS VOLTAGE	A	0-30 VDC		04 02	T030	
C-022		SHUNT BUS CURRENT	A	0-3 AMPS		04 02	T038	
C-023		RTG 4 CURRENT	A	0 TO 10 AMPS		04 06	T035	
C-025		RTG VOLTAGE - RTG 2	A	0-6 VDC		04 05	T043	
C-026		BATTERY DISCHARGE CURRENT	A	0-10 AMPS		04 02	T041	
C-027		RTG 1 CURRENT	A	0 TO 10 AMPS		04 05	T036	
C-028	1	BATTERY CHARGE STATUS	B	AUTO/CMD		04 02	T032	
C-028	2	BATTERY DISCHARGE REG STATUS	B	ENABLE/DISABLE		04 02	T033	
C-029		DC BUS CURRENT	A	0-6 AMPS		04 02	T039	
C-031		RTG VOLTAGE - RTG 3	A	0-6 VDC		04 06	T044	
C-041		SHUNT BUS CURRENT	A	0-3 AMPS		04 02	T038	

WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
		COMMUNICATIONS SUBSYSTEM						
C-011		RCV 1 AGC CONSCAN	A	-150 TO -60 DBM		05 33	T024	
C-021		RCV 2 AGC CONSCAN	A	-150 TO -60 DBM		05 41	T127	
C-037		TWTA NO 1 TEMPERATURE	A	68-194 DEG F		05 36	T007	
C-038		AUX OSC TEMP, DRIVER 1	A	30-110 DEG F		03 01	T090	
C-039		CONVERTER TEMPERATURE TWTA 1	A	56-150 DEG F		05 36	T005	
C-040		CATHODE CURRENT TWTA 1	A	24.5-33.7 MA		05 36	T001	
C-043		HELIX CURRENT TWTA 1	A	1.5-5.2 MA		05 36	T019	
C-044		LOOP STRESS, RECEIVER 1	A	-100 TO +100 KHZ		05 40	T011	
C-045		SIGNAL STRENGTH, RECEIVER 2	A	-149 TO -60 DBM		05 41	T014	
C-046		OUTPUT POWER MONITOR TWTA 2	A	6.5-9.54 DBW		05 37	T018	
C-047		CATHODE CURRENT TWTA 2	A	24.5-33.7 MA		05 37	T002	
C-048		HELIX CURRENT TWTA 2	A	1.5-5.2 MA		05 37	T020	
C-053		CONVERTER TEMPERATURE TWTA 2	A	56-150 DEG F		05 37	T006	
C-054		OSCILLATOR TEMP, RECEIVER 1	A	30-110 DEG F		05 40	T015	
C-055		AUX OSC TEMP, DRIVER 2	A	30-110 DEG F		03 01	T091	
C-056		REFERENCE VOLTAGE TWTA 1	A	0-10 VDC		05 36	T003	
C-059		OSCILLATOR TEMP, RECEIVER 2	A	30-110 DEG F		05 41	T016	
C-060		TWTA NO 2 TEMPERATURE	A	68-194 DEG F		05 37	T008	
C-061		LOOP STRESS, RECEIVER 2	A	-100 TO +100 KHZ		05 41	T012	
C-062		REFERENCE VOLTAGE TWTA 2	A	0-10 VDC		05 37	T004	
C-063		OUTPUT POWER MONITOR TWTA 1	A	6.5-9.54 DBW		05 36	T017	
C-064		SIGNAL STRENGTH, RECEIVER 1	A	-149 TO -60 DBM		05 40	T013	
C-072	1	RCVR 1 SIGNAL PRESENT	B	ABSENT/PRES		05 40	T009	
C-072	2	RCVR 2 SIGNAL PRESENT	B	ABSENT/PRES		05 41	T010	
C-072	3	OSC ENABLE/DISABLE, RCVR 1	B	DISABLE/ENABLE		05 40	T021	
C-072	4	OSC ENABLE/DISABLE, RCVR 2	B	DISABLE/ENABLE		05 41	T022	
C-077	1	PHASE OUTPUT STATUS	D	0 DEG / 180 DEG		05 33	T027	
C-077	2	THRESHOLD MODE STATUS	D	MED/HIGH		05 33	T027	
C-077	3	A SIN BIT 1 (LSB)	D			05 33	T027	
C-077	4	A SIN BIT 2	D			05 33	T027	
C-077	5	A SIN BIT 3	D			05 33	T027	
C-077	6	A SIN BIT 4	D			05 33	T027	

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
		COMMUNICATIONS SUBSYSTEM CONT						
C-078	1	A SIN BIT 5	D			05 33	T027	
C-078	2	A SIN BIT 6	D			05 33	T027	
C-078	3	A SIN BIT 7	D			05 33	T027	
C-078	4	A SIN BIT 8 (MSB)	D			05 33	T027	
C-078	5	A COS BIT 1 (LSB)	D			05 33	T027	
C-078	6	A COS BIT 2	D			05 33	T027	
C-079	1	A COS BIT 3	D			05 33	T027	
C-079	2	A COS BIT 4	D			05 33	T027	
C-079	3	A COS BIT 5	D			05 33	T027	
C-079	4	A COS BIT 6	D			05 33	T027	
C-079	5	A COS BIT 7	D			05 33	T027	
C-079	6	A COS BIT 8 (MSB)	D			05 33	T027	
C-080	1	CONSCAN STATUS	B	OFF/ON		05 33	T024	
C-080	2	CONSCAN THRESHOLD	B	BELOW/ABOVE		05 33	T025	

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
		ANTENNA SUBSYSTEM						
C-ORO	4	RECEIVER SWITCH POSITION	B	MED/HIGH		05 90	T105	
C-ORO	5	TRANSMITTER SWITCH POSITION	B	MED/HIGH		05 89	T106	
C-ORO	6	ANTENNA FEED SWITCH POSITION	B	NORM/OFFSET		05 44	T023	

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
		PROPULSION SUBSYSTEM						
C-042		PROPELLANT SUPPLY PRESSURE	A	0-600 PSIA		09 29	T028	C
C-072	5	SPIN CONTROL THRUSTER 2 PULSE CT	B			03 01	T103	C
C-072	6	SPIN CONTROL THRUSTER 1 PULSE CT	B			03 01	T104	C
C-073		VELOCITY THRUSTER ASSY 1 TEMP	A	40-200 DEG F		03 01	T084	C
C-074		SPIN THRUSTER ASSY TEMPERATURE	A	40-200 DEG F		03 01	T086	C
C-075		VPT-1 THRUSTER TEMP	A	400-1800 DEG F		03 01	T132	
C-076		VPT-2 THRUSTER TEMP	A	400-1800 DEG F		03 01	T133	
C-085		VELOCITY THRUSTER NO 2 PULSE COUNT	D	0 TO 63		03 01	T107	C
C-086		VELOCITY THRUSTER NO 4 PULSE COUNT	D	0 TO 63		03 01	T108	C
C-089		VPT-4 THRUSTER TEMP	A	400-1800 DEG F		03 01	T134	
C-090		VELOCITY THRUSTER ASSY 2 TEMP	A	40-200 DEG F		03 01	T085	C
C-091		PROPELLANT SUPPLY TEMPERATURE	A	40-140 DEG F		03 01	T066	C
C-092		VPT-3 THRUSTER TEMP	A	400-1800 DEG F		03 01	T135	
C-093		VELOCITY THRUSTER NO 1 PULSE COUNT	D	0 TO 63		03 01	T111	C
C-094		VELOCITY THRUSTER NO 3 PULSE COUNT	D	0 TO 63		03 01	T112	C
C-105	1	THRUSTER INITIATION STATUS VPT 1	B	NO/YES		03 01	T120	C
C-105	2	THRUSTER INITIATION STATUS VPT 2	B	NO/YES		03 01	T121	C
C-105	3	THRUSTER INITIATION STATUS VPT 4	B	NO/YES		03 01	T122	C
C-105	4	THRUSTER INITIATION STATUS VPT 3	B	NO/YES		03 01	T123	C
C-105	5	THRUSTER INITIATION STATUS SCT 1	B	NO/YES		03 01	T124	C
C-105	6	THRUSTER INITIATION STATUS SCT 2	B	NO/YES		03 01	T125	C

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
A-000		MAIN FRAME FORMAT A						
A-001		MODE ID	D			06 03		
A-002		BIT RATE ID	D			06 03		
A-003		FRAME SYNC	D			06 03		
A-004		FRAME SYNC	D			06 03		
A-005		FRAME SYNC	D			06 03		
A-006		FRAME SYNC	D			06 03		
A-007		FRAME SYNC	D			06 03		
A-008		FRAME SYNC	D			06 03		
A-009		GSFC MAIN FRAME DATA	D			08 54	T201	
A-010		GSFC MAIN FRAME DATA	D			08 54	T201	
A-011		GSFC MAIN FRAME DATA	D			08 54	T201	
A-012		GSFC MAIN FRAME DATA	D			08 54	T201	
A-013		UCSD MAIN FRAME DATA	D			08 55	T202	
A-014		GSFC MAIN FRAME DATA	D			08 54	T201	
A-015		GSFC MAIN FRAME DATA	D			08 54	T201	
A-016		GSFC MAIN FRAME DATA	D			08 54	T201	
A-017		GSFC MAIN FRAME DATA	D			08 54	T201	
A-018		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
A-019		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
A-020		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
A-021		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
A-022		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
A-023		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
A-024		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
A-025		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
A-026		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
A-027		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
A-028		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
A-029		U/IOWA MAIN FRAME DATA	D			08 53	T205	
A-030		U/IOWA MAIN FRAME DATA	D			08 53	T205	
A-031		U/IOWA MAIN FRAME DATA	D			08 53	T205	
A-032		U/IOWA MAIN FRAME DATA	D			08 53	T205	
A-033		FORMAT ID	D			06 03		
A-034		FORMAT ID AND SUBCOM ID	D			06 03		
A-035		SUBCOM ID	D			06 03		
A-036		SUBCOM ID	D			06 03		

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
A-037		ENGINEERING SUBCOM	D			06 03		
A-038		ENGINEERING SUBCOM	D			06 03		
A-039		SCIENCE SUBCOM	D			06 03		
A-040		SCIENCE SUBCOM	D			06 03		
A-041		GSFC MAIN FRAME DATA	D			08 54	T201	
A-042		GSFC MAIN FRAME DATA	D			08 54	T201	
A-043		GSFC MAIN FRAME DATA	D			08 54	T201	
A-044		GSFC MAIN FRAME DATA	D			08 54	T201	
A-045		JPL MAIN FRAME DATA	D			08 50	T207	
A-046		JPL MAIN FRAME DATA	D			08 50	T207	
A-047		JPL MAIN FRAME DATA	D			08 50	T207	
A-048		JPL MAIN FRAME DATA	D			08 50	T207	
A-049		JPL MAIN FRAME DATA	D			08 50	T207	
A-050		JPL MAIN FRAME DATA	D			08 50	T207	
A-051		JPL MAIN FRAME DATA	D			08 50	T207	
A-052		JPL MAIN FRAME DATA	D			08 50	T207	
A-053		JPL MAIN FRAME DATA	D			08 50	T207	
A-054		JPL MAIN FRAME DATA	D			08 50	T207	
A-055		USC MAIN FRAME DATA	D			08 56	T208	
A-056		USC MAIN FRAME DATA	D			08 56	T208	
A-057		USC MAIN FRAME DATA	D			08 56	T208	
A-058		ARC MAIN FRAME DATA	D			08 51	T209	
A-059		ARC MAIN FRAME DATA	D			08 51	T209	
A-060		ARC MAIN FRAME DATA	D			08 51	T209	
A-061		ARC MAIN FRAME DATA	D			08 51	T209	
A-062		ARC MAIN FRAME DATA	D			08 51	T209	
A-063		ARC MAIN FRAME DATA	D			08 51	T209	
A-064		ARC MAIN FRAME DATA	D			08 51	T209	

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
B-000		MAIN FRAME FORMAT B						
B-001		MODE ID	D			06 03		
B-002		BIT RATE ID	D			06 03		
B-003		FRAME SYNC	D			06 03		
B-004		FRAME SYNC	D			06 03		
B-005		FRAME SYNC	D			06 03		
B-006		FRAME SYNC	D			06 03		
B-007		FRAME SYNC	D			06 03		
B-008		FRAME SYNC	D			06 03		
B-009		UCSD MAIN FRAME DATA	D			08 55	T202	
B-010		UCSD MAIN FRAME DATA	D			08 55	T202	
B-011		UCSD MAIN FRAME DATA	D			08 55	T202	
B-012		UCSD MAIN FRAME DATA	D			08 55	T202	
B-013		UCSD MAIN FRAME DATA	D			08 55	T202	
B-014		GSFC MAIN FRAME DATA	D			08 54	T201	
B-015		GSFC MAIN FRAME DATA	D			08 54	T201	
B-016		GSFC MAIN FRAME DATA	D			08 54	T201	
B-017		GSFC MAIN FRAME DATA	D			08 54	T201	
B-018		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
B-019		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
B-020		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
B-021		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
B-022		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
B-023		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
B-024		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
B-025		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
B-026		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
B-027		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
B-028		U/CHICAGO MAIN FRAME DATA	D			08 52	T204	
B-029		U/IOWA MAIN FRAME DATA	D			08 53	T205	
B-030		U/IOWA MAIN FRAME DATA	D			08 53	T205	
B-031		U/IOWA MAIN FRAME DATA	D			08 53	T205	
B-032		U/IOWA MAIN FRAME DATA	D			08 53	T205	
B-033		FORMAT ID	D			06 03		
B-034		FORMAT ID AND SUBCOM ID	D			06 03		
B-035		SUBCOM ID	D			06 03		
B-036		SUBCOM ID	D			06 03		

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
B-037		ENGINEERING SUBCOM	D			06 03		
B-038		ENGINEERING SUBCOM	D			06 03		
B-039		SCIENCE SUBCOM	D			06 03		
B-040		SCIENCE SUBCOM	D			06 03		
B-041		UCSD MAIN FRAME DATA	D			08 55	T202	
B-042		UCSD MAIN FRAME DATA	D			08 55	T202	
B-043		UCSD MAIN FRAME DATA	D			08 55	T202	
B-044		UCSD MAIN FRAME DATA	D			08 55	T202	
B-045		JPL MAIN FRAME DATA	D			08 50	T207	
B-046		JPL MAIN FRAME DATA	D			08 50	T207	
B-047		JPL MAIN FRAME DATA	D			08 50	T207	
B-048		JPL MAIN FRAME DATA	D			08 50	T207	
B-049		JPL MAIN FRAME DATA	D			08 50	T207	
B-050		JPL MAIN FRAME DATA	D			08 50	T207	
B-051		JPL MAIN FRAME DATA	D			08 50	T207	
B-052		JPL MAIN FRAME DATA	D			08 50	T207	
B-053		JPL MAIN FRAME DATA	D			08 50	T207	
B-054		JPL MAIN FRAME DATA	D			08 50	T207	
B-055		USC MAIN FRAME DATA	D			08 56	T208	
B-056		USC MAIN FRAME DATA	D			08 56	T208	
B-057		USC MAIN FRAME DATA	D			08 56	T208	
B-058		ARC MAIN FRAME DATA	D			08 51	T209	
B-059		ARC MAIN FRAME DATA	D			08 51	T209	
B-060		ARC MAIN FRAME DATA	D			08 51	T209	
B-061		ARC MAIN FRAME DATA	D			08 51	T209	
B-062		ARC MAIN FRAME DATA	D			08 51	T209	
B-063		ARC MAIN FRAME DATA	D			08 51	T209	
B-064		ARC MAIN FRAME DATA	D			08 51	T209	

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
E1-00		SCIENCE SUBCOM FORMAT E1						
E1-01		DETECTORS TEMPERATURE	A			08 51	T301	
E1-02		ELECTRONICS TEMPERATURE	A			08 50	T302	
E1-03		SPECTRUM ANALYZER/X-AXIS OUTPUT	A			08 50	T303	
E1-04		SPECTRUM ANALYZER/Y-AXIS OUTPUT	A			08 50	T304	
E1-05		SPECTRUM ANALYZER/Z-AXIS OUTPUT	A			08 50	T305	
E1-06		STATUS	D			08 50	T306	
E1-07		EVENT COUNT	D			08 60	T307	
E1-08	1	DETECTOR D1 STATUS	B			08 52	T308	
E1-08	2	DETECTOR D2 STATUS	B			08 52	T309	
E1-08	3	DETECTOR D7 STATUS	B			08 52	T310	
E1-08	4	PRIORITY MODE STATUS	B			08 52	T312	
E1-08	5	CALIBRATE STATUS	B			08 52	T312	
E1-08	6	CALIBRATE STATUS	B			08 52	T313	
E1-09		ELECTRONICS TEMPERATURE	A			08 56	T314	
E1-10		ELECTRONICS TEMPERATURE	A			08 52	T315	
E1-11		EGG CURRENT RANGE 1	A			08 52	T316	
E1-12		EGG CURRENT RANGE 2	A			08 52	T317	
E1-13		EGG CURRENT RANGE 3	A			08 52	T318	
E1-14		FISSION DETECTOR	D			08 52	T319	
E1-15		FISSION DETECTOR	D			08 52	T319	
E1-16		FISSION DETECTOR	D			08 52	T319	
E1-17		LOW RANGE TEMPERATURE	A			08 58	T319	
E1-18		PREAMP TEMPERATURE	A			08 59	T323	
E1-19		SECONDARY VOLTAGE	A			08 59	T324	
E1-20		SPARE	A					
E1-21		SPARE	A					
E1-22		EVENT DATA	D			08 59	T327	
E1-23		EVENT DATA	D			08 59	T327	
E1-24	1	LOGIC STATUS	B			08 53	T329	
E1-24	2	STATUS	B			08 54	T330	
E1-24	3	STAR EXCLUSION STATUS	B			08 59	T331	
E1-24	4	DATA READOUT STATUS	B			08 59	T332	
E1-24	5	CHANNEL STATUS	B			08 56	T333	
E1-24	6	ROLL STATUS	B			08 56	T334	
E1-25		ELECTRONICS TEMPERATURE	A			08 54	T335	
E1-26		ANALOG DATA D1	A			08 54	T336	

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
E1-27		ANALOG DATA D2	A			08 54	T337	
E1-28		DETECTOR TEMPERATURE	A			08 54	T338	
E1-29		SECONDARY VOLTAGE	A			08 54	T339	
E1-30		IDENTIFICATION DATA	D			08 54	T340	
E1-31		COMMAND REGISTER - PART 1	D			08 58	T341	
E1-32		COMMAND REGISTER - PART 2	D			08 58	T342	

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
E2-00		SCIENCE SUBCOM FORMAT E2						
E2-01		HIGH RANGE TEMPERATURE	A			08 58	T351	
E2-02		COMMUTATED HOUSEKEEPING DATA	A			08 50	T302	
E2-03		SPECTRUM ANALYER/X-AXIS OUTPUT	A			08 50	T303	
E2-04		SPECTRUM ANALYER/Y-AXIS OUTPUT	A			08 50	T304	
E2-05		SPECTRUM ANALYER/Z-AXIS OUTPUT	A			08 50	T305	
E2-06		L1 PHA	D			08 52	T356	
E2-07		EVENT COUNT	D			08 60	T357	
E2-08	1	THRESHOLD LEVEL STATUS	B			08 59	T358	
E2-08	2	BANDWIDTH STATUS	B			08 59	T359	
E2-08	3	SPARE	B					
E2-08	4	SPARE	B					
E2-08	5	SPARE	B					
E2-08	6	SPARE	B					
E2-09		ELECTRONICS TEMPERATURE	A			08 55	T364	
E2-10		HIGH VOLTAGE MONITOR	A			08 59	T365	
E2-11		PCU MONITOR	A			08 59	T366	
E2-12		D7 COUNT RATE METER	A			08 52	T367	
E2-13		EGG TEMPERATURE	A			08 52	T368	
E2-14		L1,L2 COINCIDENCE COUNT RATE	D			08 52	T369	
E2-15		D1,D2,S D3,D7 COUNT RATE	D			08 52	T369	
E2-16		D2,D4,D5,D6,D7 COUNT RATE	D			08 52	T369	
E2-17		STANDBY VERIFICATION	A			08 57	T372	
E2-18		SPARE	A					
E2-19		SPARE	A					
E2-20		7.75 VDC MONITOR	A			08 53	T375	
E2-21		ELECTRONICS TEMPERATURE	A			08 53	T376	
E2-22		EVENT DATA	D			08 59	T327	
E2-23		EVENT DATA	D			08 59	T327	
E2-24	1	SPARE	B					
E2-24	2	TIMING STATUS	B			08 55	T380	
E2-24	3	HIGH VOLTAGE STATUS	B			08 55	T381	
E2-24	4	GSE STIMULUS STATUS	B			08 52	T382	
E2-24	5	CHANNEL STATUS	B			08 56	T333	
E2-24	6	ROLL STATUS	B			08 56	T334	
E2-25		SPARE	A					
E2-26		SPARE	A					

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WORD	BIT	MEASUREMENT	TYPE	RANGE	ACC. (%)	CONDITIONED IN	MEASUREMENT IDENTIFIER	
E2-27		SPARE	A					
E2-28		SPARE	A					
E2-29		SPARE	A					
E2-30		SPARE	D					
E2-31		SPARE	D					
E2-32		SPARE	D					

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APPENDIX C

COMMANDS
VS
TELEMETRY

COMMAND DESIGNATOR	COMMAND NUMBER	DESCRIPTION	TELEMETRY WORD/BIT	
CDS 1	102	Select Register No. 1	71/4	CDU Slot 1 = 000
CDS 2	113	Select Register No. 2	5	Indication 2 = 001
CDS 3	124	Select Register No. 3	6	Bits 3 = 010
CDS 4	135	Select Register No. 4		4 = 011
CDS 5	146	Select Register No. 5		5 = 100
CDR 1	355	Reset Memory Registers	69 70 71	
CDR 2	267	Reset Slot Select Register		
CDX 1	254	Execute Stored Sequence	-----	The command distribution unit stores up to five commands. Telemetry verification of the execution of a stored command is indicated as shown in this listing for the applicable command.
CDX 2	332	Stop stored sequence	69 70 71	CDU registers indicate status of stored commands as previously described. Those commands in the stored sequence which were executed would not be shown in the command memory readout, while those not executed would still be shown in storage.
DEP 1	200	RTG's deploy	20/2 20/3	Completed deployment indicated by change from high to low state of these two bits. Word 31, bits 2, 3, 4 and 5 will also change from high to low when the ordnance firing capacitor banks discharge.
DEP 2	500	RTG's deploy (redundant)		
DEP 3	211	Deploy Magnetometer Boom	20/1	Boom release is indicated by change from high to low state of this bit. Word 31, bits 3 and 5 will also change from high to low when boom ordnance capacitor banks discharge.
DEP 4	311	Deploy Magnetometer Boom (Redundant)		

COMMAND DESIGNATOR	COMMAND NUMBER	DESCRIPTION	TELEMETRY WORD/BIT	
ORD 2 ORD 1	243 365	Ordnance Arm Ordnance Safe	28/5 28/6	Safe/Arm relay status prime. Safe/Arm relay status redundant. Telemetry "high" indicates relay closure which connects ordnance capacitors to the AC bus.
SEQ 1	112	Start Sequencer	32/6	Command backup for the despin sensor signal is provided by command 112. Telemetry indication of sequencer operation is provided by 32/6. In addition, the four ordnance capacitor bits in word 31 will indicate high (charged) in approximately three minutes.
ANT 6 ANT 5	202 134	Receiver Signal Present Override Receiver Signal Present Reset	96/3	Receiver reverse inhibit status. High state of this bit indicates that the automatic CDU receiver reverse function is inhibited, while the low state indicates the automatic function is enabled.
PSP 2 PSP 3	246 123	Undervoltage Inhibit Undervoltage Normal	96/2	Overload Protection Status. High state of this bit indicates that the CDU circuit which removes loads upon a low bus voltage condition is disabled. The low state of the bit indicates that the circuit is enabled.
CDU 1 CDU 2	012 023	Command Processor 1 Select Command Processor 2 Select	96/4	Command Processor Memory Status. The CDU command memory registers output to either of a redundant pair of command processors. The high state of the status bit indicates the A or prime processor; the low state indicates the B or backup processor is processing the memory output.

COMMAND DESIGNATOR	COMMAND NUMBER	DESCRIPTION	TELEMETRY WORD/BIT	
SEQ 2 SEQ 3	061 072	Sequencer Inhibit Sequencer Enable	32/6	Sequencer Status. The status bit indicates high when the sequencer is enabled. It should be noted that if the inhibit command is sent, the sequencer outputs are inhibited, but the clock continues to run. If the enable command is subsequently sent before this clock completes the cycle and turns off the sequencer, an undetermined sequencer operational state exists. Command 072 should not be sent if the sequencer has been inhibited by command 061.
CDU 4 CDU 3	315 262	CDU Select 5V Bus B CDU Select 5V Bus A	96/6	CDU +5V DC Bus Status. The CDU is powered by either of a redundant pair of TRF output buses. The high state of the status bit indicates Bus A is providing CDU power.
ANT 1 ANT 2	011 122	TWTA 1 to HG Antenna TWTA 1 to MG Antenna	80/5	The high state of this bit indicates TWT #1 is connected to the medium gain antenna, and TWT #2 is connected to the high gain antenna. The low state indicates the reverse condition.
ANT 3 ANT 4	001 235	Receiver 1 to HG Antenna Receiver 1 to MG Antenna	80/4	The high state indicates receiver #1 is connected to the high gain antenna and receiver #2 to the medium gain antenna. The low state indicates the reverse condition.
MFA 9 MFA 0 MFB 9 MFB 0	210 356 364 201	Main Feed Offset Main Feed Normal Main Feed Offset Redundant Main Feed Normal Redundant	80/6	Antenna Feed Switch Position The high state indicates offset feed, the low state indicates normal feed. (Will also be verified by gain change.)
CNS 1 CNS 2	234 321	Conscan Threshold High Conscan Threshold Low	80/2	Conscan Threshold High High state indicates high threshold, low state indicates operation below threshold.

COMMAND DESIGNATOR	COMMAND NUMBER	DESCRIPTION	TELEMETRY WORD/BIT	
CNS 9	026	Conscan ON	80/1	Conscan status on = high.
CNS 0	133	Conscan OFF	29	DC Bus Current will also increase.
CNS 3	007	Conscan Phase 0 Deg.	77/1	Phase Output Status. .
CNS 4	144	Conscan Phase 180 Deg.		0 = 0° 1 = 180°
TRA 9	223	Transmitter 1 ON	63	Output Power Monitor, TWTA 1
TRA 0	232	Transmitter 1 OFF	40	Cathode Current, TWTA 1
			43	Helix Current, TWTA 1
				Tube and Converter temperatures are also telemetered.
TRB 9	310	Transmitter 2 ON	46	Output Power Monitor, TWTA 1
TRB 0	331	Transmitter 2 OFF	47	Cathode Current, TWTA 1
			48	Helix Current, TWTA 1
				Tube and Converter temperatures are also telemetered.
		NOTE: No interlock between the two TWTA ON/OFF commands exists, thus it is possible to turn both TWTA's either on or off independently.		
COA 2	343	Coherent Mode Non-Inhibit Receiver 1	72/3	Oscillator Enable/Disable, Receiver 1
				Oscillator enabled = high state
COA 1	354	Coherent Mode Inhibit Receiver 1		Also verified by presence/absence of coherency of received signal.
COB 2	226	Coherent Mode Non-Inhibit Receiver 2	72/4	Oscillator Enable/Disable, Receiver 2
				Same as Receiver 1 above.
COB 1	212	Coherent Mode Inhibit Receiver 2		
RTM 1	X60	Real Time Mode	Main Frame	Real Time 000 or 001
MRM 1	X61	Memory Readout	Word 1	Memory R/O 100 or 101
TSM 1	X62	Telemetry Store		Tlm Store 010 or 011
DTA 1	044	Redundant A		Note that the third bit above indicates
DTB 1	155	" B		either DTU-A (0) or DTU-B (1) on.

COMMAND DESIGNATOR	COMMAND NUMBER	DESCRIPTION	TELEMETRY WORD/BIT		
CCM 1	X70	Coded Data	32/1	Code Generator On/Off.	
VCM 1	X71	Uncoded Data		On is indicated by the high state.	
BIT 8	X27/X57	2048 Bit Rate	Main Frame Word 2	2048	111
BIT 7	X46/X56	1024 Bit Rate		1024	110
BIT 6	X45/X55	512 Bit Rate		512	101
BIT 5	X44/X54	256 Bit Rate		256	100
BIT 4	X43/X53	128 Bit Rate		128	011
BIT 3	X42/X52	64 Bit Rate		64	010
BIT 2	X41/X51	32 Bit Rate		32	001
BIT 1	X40/X50	16 Bit Rate		16	000
		Format Commands TBS			
MHM 1	X64	Memory Halt		Format and mode return to last commanded state.	
MCM 1	X65	Memory Continue		Memory readout continues from last memory location.	
RIP 1	X66	Roll Reference 0 Deg.	113/1	Roll Reference.	
RIP 2	X67	Roll Reference 180 Deg.		0 = 0° 180° = 1	
RIP 3	X77	Averaging Mode	113/2	Spin Averaging Mode	
RIP 4	X76	Non-Averaging Mode		No = 0, Yes = 1	
RIP 5	X74	ACS Mode	113/3	ACS Mode	
				No = 0, Yes = 1	

COMMAND DESIGNATOR	COMMAND NUMBER	DESCRIPTION	TELEMETRY WORD/BIT	
BAT 1 BAT 2	043 164	Battery Auto Charge Battery Float Charge	28/1	Battery Charge Status. Low state indicates automatic control, high state indicates commanded float charging. Further battery status is also provided by analog words 9 and 27, battery charge and discharge current monitors.
BAT 3 BAT 4	004 115	Discharger Enable Discharger Disable	28/2	Battery Discharge Regulator Status. Low state indicates enable. High state indicates disabled.
BAT 9 BAT 0	015 126	Battery Heater ON Battery Heater OFF	15	Battery Temperature.
PSP 1	204	Relay Reset		An RTG voltage and current monitor Indicating zero would indicate the relay may require resetting.

COMMAND DESIGNATOR	COMMAND NUMBER	DESCRIPTION	TELEMETRY WORD/BIT		
PSE 9	166	ACS Power Standby (PSE On & DSL's Off)	117/4 5 6	Power Status Bits	001
SLA 9*	036	DSL-1 ON - Reset PSE			10X
SLB 9*	047	DSL-2 ON - Reset PSE			01X
PSE 0	025	PSE-OFF			XX0
ACS 1	233	Arm Register 1	128/4 5	Arm Status Bits	100
ACS 2	244	Arm Register 2	6		010
ACS 3	255	Arm Register 3			001
ACX 1	222	Sequence Execute	128/3	ACS Reg. Inhibit	1
ACX 2**	345	Sequence Inhibit	128/3	ACS Reg. Inhibit	0
ACX 3	010	Sequence Override	128/3	ACS Reg. Inhibit	1
ACX 4	101	Sequence Reset	127/1 2 3		000
ACX 5	042	Sequence Step	127/1 2 3	Monitor change in code	XXX

* Should be followed by ACR 1 -- clock reset

** Should be followed by either sequence override ACS 3 (to continue) or sequence reset ACS (to start over)

COMMAND DESIGNATOR	COMMAND NUMBER	DESCRIPTION	TELEMETRY WORD/BIT		
PUL 1	022	Pulse length 1 -0.03125 sec	099/2	Pulse Length	001
PUL 2	213	Pulse length 2 -0.125	3	Codes	011
PUL 3	143	Pulse length 3 -0.5	4		111
PUL 4	301	Pulse length 4 -1.0			110
PUL 5	055	Pulse length 5 -2.0			100
PRE 1	266	Precession Pair 1 (2 & 3)	099/1	Precession Pair	0
PRE 2	322	Precession Pair 2 (1 & 4)			1
PRE 3	230	0° Precession pulse	Verified by CDU pulse count data of the selected thrusters (Words 85, 86, 93, and 94) and by thruster temperature data.		
PRE 4	241	90° Precession pulse			
PRE 5	252	180° Precession pulse			
PRE 6	263	270° Precession pulse			
PRE 7	274	Real time precession pulse			
VEL 3	205	Real time ΔV pulse			
VEL 4	216	ΔV SCT Mode Enable			

COMMAND DESIGNATOR	COMMAND NUMBER	DESCRIPTION	TELEMETRY WORD/BIT			
VEL 1	277	V pair 1 (2 & 4)	099/5	Delta V Pair State		0
VEL 2	333	V pair 2 (1 & 3)				1
SPN 1	330	Despin Start	106/1	Despin on State		1
SPN 2	227	Real time spin pulse	72/5 72/6	Spin thruster 2 pulse count Spin thruster 1 pulse count Also monitor spin period indication		
SPN 3	224	Spin direction up	99/6	Spin control direction - state 0		
SPN 4	312	Spin direction down				state 1
SEN 1	303	Sensor select, Sun A	127/5 & 6	Sun A		11
SEN 2	314	Sensor select, Sun B		Sun B	Code	10
SEN 3	325	Sensor select, Star		Star		01

COMMAND DESIGNATOR	COMMAND NUMBER	DESCRIPTION	TELEMETRY WORD/BIT	
SAS 1	256	Star Angle Gate Enable	106/5	Angle Gate Enable Status - Enable 1
SAS 2	334	Star Angle Gate Disable		Disable 0
SAS 3	245	Star Delay up Lower Half	115/1-6	Star Delay (Lower half 12 bit register)
SAS 4	373	Star Delay up Upper Half	116/1-6	Star Delay (Upper half 12 bir register)
SAS 6	337	Star Delay down Upper Half		
SAT 1	340	Star Time Gate Enable	127/4	Star Time Gate Enable Status - State 1
SAT 2	351	Star Time Gate up	100/1-5	Time Gate Register
SAT 3	362	Star Time Gate down		

COMMAND DESIGNATOR	COMMAND NUMBER	DESCRIPTION	TELEMETRY WORD/BIT	
ACA 1	336	Time Frequency Select 1	106/3	Clock Select 1 - State 1
ACB 1	347	Time Frequency Select 2	106/4	Clock Select 2 - State 1
ACR 1	206	Clock Reset	106/3 & 4	Clock Select 1 and 2 - State 0
CNS 5	352	Closed loop start	106/2	Conscan On - State 1
CNS 6	363	Closed loop stop		State 0

APPENDIX D

POWER ALLOCATIONS

POWER ALLOCATION DOCUMENT
ELECTRICAL INTERFACE CONTROL
PIONEER F/G PROJECT

1.0 Scope

This Electrical Interface Control document established the power allocations for the subsystems and equipments which comprise the Pioneer F/G Spacecraft.

2.0 Applicable Documents

2.1 The following documents, of the exact issue shown form a part of this Electrical Interface Control Document to the extent specified herein. In the event of the conflict between documents referred herein and other detailed contents of Section 3, the detailed requirements of Section 3 shall be considered a superceding requirement, except for PC-210 which shall supercede in all cases.

Specifications

PC-210, Revision 4 10 March 1970	Spacecraft and Relatcd Requirements
PC-220, Revision 10 8 May 1970	Spacecraft/Scientific Instrument

3.0 Power Requirements

3.1 The Power Allocations for each subsystem and equipment unit are given in the Pioneer Power Requirements Table, pages 1 through 12. The description of the column headings used in the Table and explanations of the various coded entries are given in Section 4. Notes.

4.0 Notes

4.1 Description of Column Headings

4.1.1 Equipment

This column lists the name of the unit for which the power allocation is given. The first line entry, in general, identifies the subsystem associated with the units listed below it. The number of redundant units operating is indicated in parentheses.

4.1.2 Input Voltage

This column lists the voltage required by the equipment. If the voltage is 16 volts or less, it is supplied as a secondary voltage from the Central TRF unit. If the voltage listed is 28 VDC it is supplied from the main dc bus in the Power Control Unit. If the voltage is 30.5 volts a-c, it is supplied from the cross-strapped inverter assemblies forming the main a-c bus.

4.1.3 TOL + %

This column lists the tolerance associated with the users input voltage. It defines the voltage tolerance the equipment can tolerate, measured at the unit's input connector.

4.1.4 Mission Phase

The mission phases are defined as: launch, cruise, and commanding.

4.1.4.1 Launch

The Launch Phase spans the time from pre-launch and liftoff up to about six hours, prior to the first "conscan operation" and prior to experiment turn-on. During this time the Radioisotope Thermoelectric Generators are supplying their maximum output power.

4.1.4.2 Cruise

The Cruise Phase is defined as that time when the spacecraft is fully operational and is cruising with all experiments turned on. A point in time is assumed where the RTG's output power has degraded to minimum operating power (approximately 900 days from liftoff).

4.1.4.3 Peak Power Commanding

The Commanding Phase is defined as the maximum spacecraft power mode. The spacecraft is in the cruise mode with the following additional functions simultaneously taking place:

a) A command is being decoded.

b) The Command Distribution Unit is processing this command.

c) Propellant line thermostats have turned on both line heaters.

d) The conscan signal processor has been turned on.

e) The spacecraft is preparing for a conscan maneuver. Thrusters are not yet firing.

f) The DSU is in the 16KBS rate mode. (2KBS for normal commanding mode.)

g) The battery heater is assumed off.

4.1.5 Min - Nom - Max

The column headings MIN - NOM - MAX are defined as follows:

Column Heading	Power	Input Voltage	Voltage Tolerance	Design	Temp.
MIN	Minimum	Lowest	Minus Limit	Worst Case	Extreme
NOM	Nominal	Nominal	Zero	Nominal	70°F
MAX	Maximum	Highest	Plus Limit	Worst Case	Extreme

4.1.6 Notes

4.1.6.1 The last column contains explanatory notes.

4.1.6.2 Experiment data was taken from PC 220 Rev 14.

4.1.6.3 Commanding mode was derived from PC 210.03 Section 3.1.5.9 and NAS2-5600, App F, Section 3.2.2.

4.1.7 Abbreviations

Abbreviations used in this document are listed below:

ACS	Attitude Control System
CEA	Control Electronics Assembly
CDU	Command Distribution Unit
CMD	Command
Conscan	Conscan Signal Processor
CTRF	Central Transformer/Rectifier Filter
DDU	Digital Decoder Unit
Driver	Transmitter Driver
DSU	Data Storage Unit
DTU	Digital Telemetry Unit
MAX	Maximum
MIN	Minimum
NOM	Nominal
PCU	Power Control Unit
Prop	Propellant Line Heaters
RCVR	Receivers
RTG	Radioisotope Thermoelectric Generator
SSA	Sun Sensor Assembly
SRA	Stellar Reference Assembly
Transducer	Propellant Pressure Transducer
TWTA	Traveling Wave Tube Amplifier

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EQUIPMENT	MISSION PHASES		
	LIFTOFF TO T+6 HOURS	3 RTG'S NEAR ENCOUNTER	
	Launch Nominal	Cruise Nominal	Commanding Nominal
DTU	3.725	3.725	3.725
DDU (2)	0.290	0.290	1.264
DSU	0.425	0.542	1.163
RCVRS (2)	3.400	3.400	3.400
DRIVER	1.352	1.352	1.352
CEA	2.210	0.910	2.685
SSA	0.170	0.170	0.170
SRA	0.294	0.294	0.294
CONSCAN	0.0	0.0	1.200
CDU	1.855	1.855	2.332
CENTRAL TRF OUTPUT	13.721	12.538	17.585
CENTRAL TRF LOSS	8.966	8.579	9.853
CTRF EFFICIENCY (%)	60.4	59.4	64.1
CENTRAL TRF INPUT	22.687	21.117	27.438
BATTERY ELECTRONICS LOSS	0.1	0.1	0.1
TRANSDUCER	0.238	0.238	0.238
PROP HEATERS (2)	2.0	2.0	2.0
ACS	0.269	0.269	0.454
CDU	0.215	0.215	0.215
TWTA	27.8	27.8	27.8
CABLE LOSS (S/C)	0.6	0.6	0.6
PCU LOSS	14.42	7.228	7.296
INVERTER LOSS (13% - 11%)	21.715	10.329	11.141
RTG CABLE LOSS	6.96	3.912	4.220
SPACECRAFT SUB TOTAL (Less Shunt Loss)	97.012	73.808	81.502
EXPERIMENT POWER	0.0	24.0	24.0
VEHICLE TOTAL POWER (Less Shunt Dissipation)			
SHUNT RADIATOR	38.988	0.0	0.0
SHUNT REGULATOR	38.000	0.0	0.0
RTG OUTPUT POWER	174.000	97.808	105.502

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EQUIPMENT	INPUT VOLTAGE	TOL ± %	MISSION PHASE: LAUNCH MIN	LAUNCH NOM	LAUNCH MAX	CRUISE MIN	CRUISE NOM	CRUISE MAX	COMMANDING MIN	COMMANDING NOM	COMMANDING MAX	MIN	NOM	MAX	
DTU (2 UNITS)															
	+5VDC	5		254	2814		254	2814		254	2814				SWITCHED
	+5VDC	5		0565	0620		0565	0620		0565	0620				UNSWITCHED
	+12VDC	5		02	0234		02	0234		02	0234				A ON B STBY
	-12VDC	5		015	0156		015	0156		015	0156				
	-16VDC	5		027	0312		027	0312		027	0312				
TOTAL				3725	4136		3725	4136		3725	4136				TOTAL FOR BOTH UNITS
DDU (2 UNITS)															
	+5.3VDC	5		0040	0044		0040	0044		0848	0944				POWER DIVIDED
	+16VDC	5		0122	0135		0122	0135		0128	0142				EQUALLY BETWEEN UNITS
	-16VDC	5		0128	0143		0128	0143		0288	0320				
TOTAL				0290	0322		0290	0322		1264	1406				
DSU	+5VDC	5		0411	0457		0424	0471		0516	0573				CRUISE = 2 KHZ
	+12VDC	5		0006	0007		0039	0043		0130	0144				MEMORY STORE MODE
	-12VDC	5		0008	0009		0079	0087		0517	0575				CMDG = IPP 16 KHZ
TOTAL				0425	0473		0542	0601		1163	1292				DATA STORE MODE
RCVRS (2 UNITS)															
	+12.0VDC	2		148	1799		148	1799		148	1799				POWER DIVIDED EQUALLY
	-12.0VDC	2		192	2200		192	2200		192	2200				BETWEEN RECEIVERS
TOTAL				340	3999		340	3999		340	3999				

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EQUIPMENT	INPUT VOLTAGE	TOL ± %	MISSION PHASE: LAUNCH MIN	LAUNCH NOM	LAUNCH MAX	CRUISE MIN	CRUISE NOM	CRUISE MAX	COMMANDING MIN	COMMANDING NOM	COMMANDING MAX	MIN	NOM	MAX	
DRIVER	-16VDC	3		1352	1499		1352	1499		1352	1499				
CONSCAN (DIGITAL)															
	+5VDC	5			0		0	0	0950	105	1171				
	+12.0VDC	3			0		0	0	0021	0022	0022				
	-12.0VDC	3			0		0	0	0119	0124	0130				
TOTAL					0		0	0	109	1196	1325				
ACS	+5VDC	8		222	266		92	108		278	336				COMMANDING
	+12VDC	1.5		27	296		27	296		27	296				NO VALVES
	-12VDC	1.5		184	196		184	196		184	196				
	+28VDC	2		269	328		269	328		454	568				
TOTAL				294	349		164	191		369	443				
CDU	+5.3VDC	3	1748	1855	1965	1748	1855	1965	2159	2332	2511				
	28VDC	2	0215	0215	0215	0215	0215	0215	0215	0215	0215				
TOTAL			1963	207	218	1963	207	218	2374	2547	2726				

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EQUIPMENT	INPUT VOLTAGE	TOL ± %	MISSION PHASE											
			LAUNCH			CRUISE			COMMANDING					
			MIN	NOM	MAX	MIN	NOM	MAX	MIN	NOM	MAX	MIN	NOM	MAX
POWER SUMMARY														
DATA HANDLING SUBSYSTEM														
DTU				3719	4137		3719	4137		3719	4137	DTU A ON B STBY		
DDU (2)				0289	0321		0289	0321		1264	1406			
DSU				0425	0473		0542	0601		1163	1292	DSU, LAUNCH-STBY		
TOTAL				4433	4931		455	5059		5171	5750	DSU, CRUISE-2KHZ (MS/MRO) DSU, CMD - 16KHZ (IPP-DS)		
COMMUNICATIONS SUBSYSTEM														
CONSCAN			00	00	00	00	00	00	109	120	1323	CONSCAN LAUNCH AND CRUISE MODE IS STBY		
RECEIVERS	(2)			34	40		34	40		34	40			
DRIVER				135	15		135	15		135	15			
TWTA				278	283		278	283		278	283			
TOTAL				3255	338		3255	338		3375	35123			

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EQUIPMENT	INPUT VOLTAGE	TOL ±	MISSION PHASE: LAUNCH MIN	LAUNCH NOM	LAUNCH MAX	CRUISE MIN	CRUISE NOM	CRUISE MAX	COMMANDING MIN	COMMANDING NOM	COMMANDING MAX	MIN	NOM	MAX
POWER SUMMARY														
ATTITUDE CONTROL (ACS) SUBSYSTEM														
CEA				248	297		118	139		314	379			
SUN SENSOR				017	02		017	02		017	02			
STELLAR REFERENCE				0294	0298		0294	0298		0294	0298			
TOTAL				294	347		164	189		360	429			
ELECTRICAL POWER SUBSYSTEM LOSS														
PCU				14420			7228			7296				
BATTERY ELECTRONICS				01	01		01	01		01	01			
CENTRAL TRF				8966			8579			9853				
INVERTERS				21715			10329			11141				
SHUNT REGULATOR				3898R			00	00		00	00			
SHUNT RADIATOR				38000			00	00		00	00			
TOTAL				122189			26236			28390				
COMMAND/ELECTRICAL DISTRIBUTION SUBSYSTEM														
CDU				197			208			266				
SPACECRAFT CABLES				06			06			06				
RTG CABLE				696			3912			4220				
TOTAL				9530			6592			7480				

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PIONEER POWER REQUIREMENTS (WATTS)

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EQUIPMENT	INPUT VOLTAGE	TOL ± %	MISSION PHASE											
			MIN	NOM	MAX	MIN	NOM	MAX	MIN	NOM	MAX	MIN	NOM	MAX
PULSED LOAD	+28VDC													
EXPERIMENTS (11)														50W-MINUTE 13W MAX 2 PER CENT DUTY CYCLE
TRANSFER SWITCH (1)				112										30 TO 70 MILLISECOND PULSE POWER
FEED MOVEMENT MECHANISM				40										12 - 26 MIN. CONT. PLUS 162 MIN AT 50 PER CENT DUTY CYCLE
THRUSTER DURATION			.032	.125	.500	1.0	2.0	SEC.	OR	CONT	FOR			90 MINUTES (MAX)
1 VALVE				56										
2 VALVE				112										2 VALVES ARE NORMAL OPERATION
ORDNANCE	30.5VAC			05	15									NOM = 3 MIN. AVG., MAX = PEAK PRIOR TO SQUIB FIRING.
(1) SQUIB				45	250									POWER SUPPLIED FROM
(6) SQUIBS				270	1500									CAPACITOR BANK IN CDU
PROPULSION LINE HEATERS 1	ON			10	10						50			PER CENT DUTY CYCLE
2	ON			20	20									2 ON ARE NORMAL OPERATION
BATTERY HEATER				16										
CDU LOSSES														
o FEED MOVEMENT				025	03									
o ORDNANCE FIRING				30	36						70			MS PULSE POWER

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PIONEER POWER REQUIREMENTS (WATTS)

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EQUIPMENT	INPUT VOLTAGE	TOL ± %	MISSION PHASE												
			MIN	NOM	MAX	MIN	NOM	MAX	MIN	NOM	MAX	MIN	NOM	MAX	
EXPERIMENT PC220, REV 13	LOADS +28VDC		MIN.	AVG.	PEAK										
SMITH	JPL			308	410										AVG DURING CRUISE PEAK AT ENCOUNTER
WOLFE	ARC		40	45	50										
SIMPSON	CHG			239	239										CONT
VAN ALLEN	IOWA			080	080										CONT
MC DONALD	GSFC		24	28											CONT
FILLIUS	UCSD		20	21											CONT
JUDGE	USC			100	700										7W PEAK FOR 10 MS AT L + 24 HOURS
GEHRELS	ARZ			350	410										2.75 SEC PEAK EVERY 4 REVS IN IMAGING MODE
MUNCH	CIT			120	170										.1 SEC PEAK EACH REV AT ENCOUNTER
SOBERMAN	GE			170	280										.2 SEC PEAK (MAX) EACH HOUR
KINARD	LARC			075	100										PEAK, TBS MILL SEC TO .5HR 1 EACH DAY MAX.
TOTAL				2382	3379										

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EQUIPMENT	INPUT VOLTAGE	TOL ± %	MISSION PHASE: STAND BY MIN	STAND BY NOM	STAND BY MAX	PROCESSING MIN	PROCESSING NOM	PROCESSING MAX	MIN	NOM	MAX	MIN	NOM	MAX	
VALUES IN CURRENT (MILLIAMPS)															
DTU (A)	+5VDC(SW)	5		508	536		508	536							SWITCHED
ON	+5VDC	5		59	62		59	62							UNSWITCHED
	+12VDC	5		83	93		83	93							
	-12VDC	5		62	62		62	62							
	-16VDC	5		83	93		83	93							
DTU (B)	+5VDC(SW)	5		0	0		0	0							
STANDBY	+5VDC	5		54	56		54	56							
	+12VDC	5		83	93		83	93							
	-12VDC	5		62	62		62	62							
	-16VDC	5		83	93		83	93							
DDU (A)	+5.3VDC	5		37	392		80	848							
	+16VDC	5		38	402		40	424							
	-16VDC	5		40	424		9	954							
DDU (B)	+5.3VDC	5		37	392		80	848							
	+16VDC	5		38	402		40	424							
	-16VDC	5		40	424		9	954							
DSU	+5VDC	5		822	871										
(STANDBY)	+12VDC	5		05	053										
	-12VDC	5		07	074										
DSU-2KHZ	+5VDC	5					848	898							
(STORE)	+12VDC	5					326	34							
	-12VDC	5					66	69							
DSU-16KHZ	+5VDC	5					1031	1092							
(STORE)	+12VDC	5					108	114							
	-12VDC	5					431	456							

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EQUIPMENT	INPUT VOLTAGE	TOL ± -	MISSION PHASE: STANDBY MIN	STANDBY NOM	STANDBY MAX	PROCESSING MIN	PROCESSING NOM	PROCESSING MAX	MIN	NOM	MAX	MIN	NOM	MAX
VALUES IN CURRENT (MILLIAMPS)														
RECEIVER (1)	+12.0VDC	2		616	735		616	735						
	-12.0VDC	2		80	899		80	899						
TRANSMITTER DRIVER (1)	-16VDC	3			00		845	910						
CONSCAN LOGIC	+5.0 VDC	5			00	200	210	223						
	+12.0VDC	3			00	18	18	18						
	-12.0VDC	3			00	102	103	105						
TRAVELING WAVE TUBE ASSEMBLY	28VDC	2												
		+2		2				9999						
		0		2				9930						
		-2		2				9750						
COMMAND DISTRIBUTION UNIT	+5.3VDC	3	340 0	3500	3600	420 0	4400	4600						
	28VDC	2		8	8		8	8						
	30.5VRMS	3		00	00									
PRESSURE TRANSDUCER	+28VDC	1		85	150		85	150						
PROPELLANT LINE HEATERS	+28VDC	1												
1 HEATER ON							35	36						
2 HEATERS ON							70	72						

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EQUIPMENT	INPUT VOLTAGE	TOL ±	MISSION PHASE: STORAGE MIN	STORAGE NOM	STORAGE MAX	STAND BY MIN	STAND BY NOM	STAND BY MAX	COMMANDING MIN	COMMANDING NOM	COMMANDING MAX	MIN	NOM	MAX	
VALUES IN CURRENT (MILLIAMPS)															
ACS SYSTEM															
CONTROL ELECTRONICS ASSEMBLY (CEA)															
	+5VDC	8		442	490		182	197		537	597				NO THRUSTERS FIRING
	+28VDC	2		96	115		96	115		162	199				
SUN SENSOR ASSEMBLY (SSA)															
	+12VDC	1.5		83	98		83	98		83	98				
	-12VDC	1.5		53	61		53	61		53	61				
	+5VDC	8		18	25		18	25		18	25				
STELLAR REFERENCE ASSEMBLY (SRA)															
	+12VDC	1.5		145			145			145					
	-12VDC	1.5		10			10			10					

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EQUIPMENT	INPUT VOLTAGE	TOL ± %	MISSION PHASE: LAUNCH MIN	LAUNCH NOM	LAUNCH MAX	ENCOUNTER MIN	ENCOUNTER NOM	ENCOUNTER MAX	MIN	NOM	MAX	MIN	NOM	MAX
POWER CONTROL UNIT (PCU) LINE-TO-CENTER TAP	30.5VRMS	3												
CHARGE REGULATOR				07			07							
DISCHARGE REGULATOR				03			03							
SHUNT REGULATOR				09			09							
TELEMETRY				08			08							
EXPERIMENT FUSES				00			01							
POWER CONDITIONING				11720			4596					VARIES WITH LOAD EQUALS (1.8VDC) (I_{DC}) + (0.2ohm) $(I_{DC})^2$		
TOTAL				14420			7296							

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EQUIPMENT	INPUT VOLTAGE	TOL ± %	MISSION PHASE									
			LAUNCH			ENCOUNTER						
			MIN	NOM	MAX	MIN	NOM	MAX	MIN	NOM	MAX	MIN	NOM	MAX
INVERTER ASSEMBLY			87% EFFICIENT			89% EFFICIENT								
				21715			11141							

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