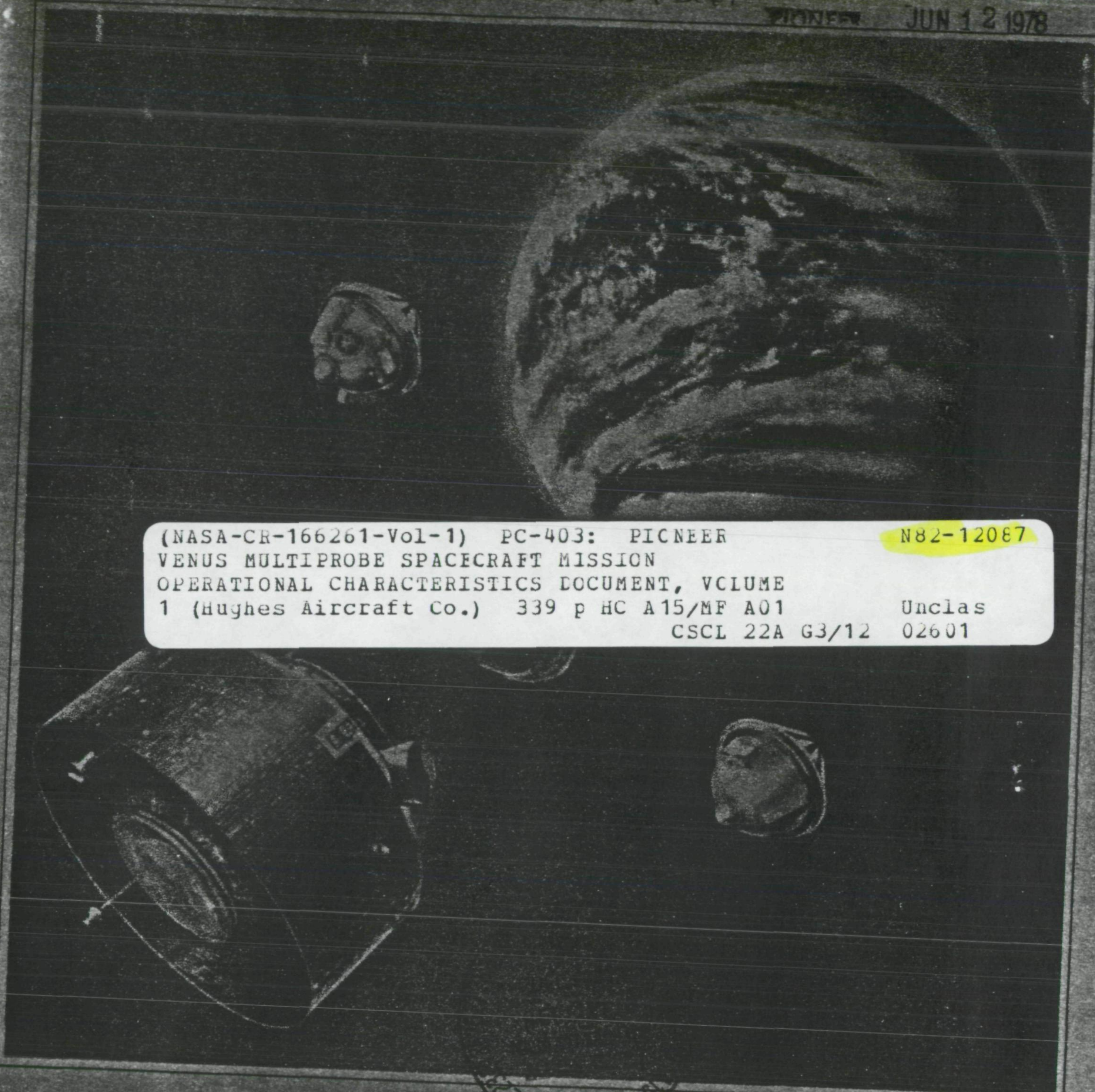
CR 166261

JUN 12 1978



(NASA-CR-166261-Vol-1) PC-403: PIONEER VENUS MULTIPROBE SPACECRAFT MISSION OPERATIONAL CHARACTERISTICS DOCUMENT, VOLUME 1 (Hughes Aircraft Co.) 339 p HC A15/MF A01 CSCL 22A G3/12 N82-12087 Unclas 02601

PC-403 — VOLUME I OF III
PIONEER VENUS MULTIPROBE SPACECRAFT MISSION OPERATIONAL CHARACTERISTICS DOCUMENT

MAY 1978

CONTRACT NO. NAS 2-9366 • HS 507-7517 • HUGHES REF NO. D8821

HUGHES

HUGHES AIRCRAFT COMPANY
SPACE AND COMMUNICATIONS GROUP

MAY 1978

PC-403 — VOLUME I OF III
PIONEER VENUS MULTIPROBE SPACECRAFT MISSION OPERATIONAL CHARACTERISTICS DOCUMENT

CONTRACT NO. NAS 2-9366 • HS 507-7517 • HUGHES REF NO. D8821

HUGHES

HUGHES AIRCRAFT COMPANY
SPACE AND COMMUNICATIONS GROUP
EL SEGUNDO, CALIFORNIA

Acknowledgement

This document was prepared by the Hughes Aircraft Company Pioneer-Venus System Engineering Organization and their consultants, that included the following major contributors:

- F.C. Barker
- L.W. Butterworth
- R.E. Daniel
- R.J. Drean
- K.A. Filetti
- J.N. Fisher
- L.A. Nowak
- J. Porzucki
- F.L. Richardson
- J.O. Salvatore
- J. Sikola
- G.A. Tadler

TABLE OF CONTENTS/REVISIONS

PRECEDING PAGE BLANK NOT FILMED

TABLE OF CONTENTS/REVISIONS (Continued)

ORIGINAL PAGE IS
OF POOR QUALITY

Revision

TABLE OF CONTENTS/REVISIONS (Continued)

TABLE OF CONTENTS/REVISIONS (Continued)

Revision

TABLE OF CONTENTS/REVISIONS (Continued)

Revision

TABLE OF CONTENTS/REVISIONS (Continued)

Revision

TABLE OF CONTENTS/REVISIONS (Continued)

TABLE OF CONTENTS/REVISIONS (Continued)

Revision

TABLE OF CONTENTS/REVISIONS (Continued)

TABLE OF CONTENTS/REVISIONS (Continued)

Revision

TABLE OF CONTENTS/REVISIONS (Continued)

TABLE OF CONTENTS/REVISIONS (Continued)

TABLE OF CONTENTS/REVISIONS (Continued)

TABLE OF CONTENTS/REVISIONS (Continued)

Revision

LIST OF TABLES

LIST OF TABLES (Continued)

Revision

LIST OF TABLES (Continued)

Revision

LIST OF TABLES (Continued)

LIST OF TABLES (Continued)

LIST OF TABLES (Continued)

Revision

LIST OF TABLES (Continued)

LIST OF TABLES (Continued)

Revision

LIST OF TABLES (Continued)

Revision

LIST OF TABLES (Continued)

LIST OF FIGURES

LIST OF FIGURES (Continued)

LIST OF FIGURES (Continued)

LIST OF FIGURES (Continued)

LIST OF FIGURES (Continued)

ORIGINAL PAGE IS
OF POOR QUALITY

LIST OF FIGURES (Continued)

LIST OF FIGURES (Continued)

LIST OF FIGURES (Continued)

LIST OF FIGURES (Continued)

LIST OF FIGURES (Continued)

LIST OF FIGURES (Continued)

LIST OF FIGURES (Continued)

LIST OF FIGURES (Continued)

LIST OF FIGURES (Continued)

LIST OF FIGURES (Continued)

LIST OF FIGURES (Continued)

1.0 INTRODUCTION

The purpose and organization of this document, applicable documents, and the Multiprobe Mission Objective and basic description, are presented below.

1.1 DOCUMENT PURPOSE

The Pioneer-Venus Multiprobe Operational Characteristics document was generated by Hughes Aircraft Company between September 1976 and May 1978 for the National Aeronautics and Space Administration, Ames Research Center, under Contract Number NAS2-9366. The purpose of this document is to describe the operational characteristics of the Multiprobe System and its subsystems in extensive detail to the NASA/ARC personnel having the responsibility for conducting the Pioneer-Venus Multiprobe Flight Mission operations.

1.2 DOCUMENT ORGANIZATION

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

Section 3 presents Bus Spacecraft functional and operational descriptions at the subsystem and unit level. The subtleties of nominal operation as well as detailed capabilities and limitations beyond nominal performance are discussed. A Command and Telemetry Logic flow diagram for each subsystem is included. Each diagram identifies in symbolic logic all signal conditioning encountered along each command signal path into, and each telemetry signal path out of the subsystem.

Section 4 describes how the Multiprobe Spacecraft performs in normal operating modes that correspond to the performance of specific functions at the time of specific events in the mission. Principal backup means of performing the normal Multiprobe operating modes are included.

Section 5 presents descriptions, normal operating modes, and backup operating modes for the Large Probe and the Small Probes. The level of detail is analogous to Sections 3 and 4, but is limited in scope to that which can be controlled or initiated by ground operations during the mission.

Appendix A is a listing of all telemetry and essential pertinent operational characteristics, including source derivation and power source. (Source derivation is the generic signal source, prior to any signal conditioning, of the telemetry parameter). Power Source is a description of the regulated or semi-regulated voltage applied to the generic signal source, starting with the Power Bus Source and tracing the power path to the generic signal source.

Appendix B is a listing of all telemetry in alphabetical order for the mnemonics. It excludes power source and source derivation, repeats some information from Appendix A; and includes some additional information such as Format assignment and TM Word (and BIT) assignment.

Appendix C is the collection of all foldout drawings and foldout diagrams referenced throughout the document.

1.3 MULTIPROBE MISSION OBJECTIVES

The basic spacecraft objective of the Multiprobe Spacecraft is to deliver scientific instruments that are on-board each of four probes and a bus into Venus' atmosphere during the 1978 opportunity.

These instruments will provide data that will be used to satisfy the scientific objectives of the mission, that are to determine (1) Nature and composition of the clouds, (2) composition and structure of the atmosphere, and (3) general circulation pattern of the atmosphere.

In particular, the Large Probe is intended to (1) measure the structure, composition, and clouds of the atmosphere through altitude extremes, and (2) assess the energy balance interaction between the atmosphere, the sun, and thermal radiation. All four probes are intended to provide space-diversified measurements for three-dimensional modeling of lower atmosphere dynamics. The bus is intended to measure the upper atmosphere composition below the orbiter periapsis altitude.

1.4 MISSION SYSTEMS DESCRIPTION

The top-level elements of the Multiprobe Mission consist of the following systems: (1) The Multiprobe Spacecraft that will journey to Venus' atmosphere to obtain scientific data until final impact with the surface. It consists of a bus that carries one separable Large Probe and three separable Small Probes. (2) The Atlas SLV-3D/Centaur D-1AR Launch Vehicle that will place the Multiprobe Spacecraft on the desired Type I interplanetary trajectory. (3) The Deep-Space Network (DSN) of Earth Stations that will provide the command and data links between Mission Control and the Multiprobe Spacecraft. 64-Meter ground stations will be used for maneuvers; probes checkouts, releases, and entries; and Bus entry. 26-Meter ground stations will be used on a one-station-pass-per-day basis, otherwise. (4) The Pioneer Mission Operations Center (PMOC), located at NASA-Ames Research Center, that includes real-time data processing and display equipment, off-line processing equipment for attitude determination and prediction, and the mission operations team. (5) The Mission Flight Sequence Plan and all ancillary reference documents and procedures. The Plan is segmented into nine Mission time phases for the purpose of detailed communication ahead in this document. The nine Mission time phases, in chronological order are: (1) Prelaunch through spinup to 15 rpm, (2) Attitude determination, star survey, and bus science checkout, (3) Jet Calibration and TCM (Trajectory Correction Maneuver) #1, (4) TCM#2, (5) Probes checkout; transmitter power and bit rate changes, (6) TCM #3 through horn attitude operations, (7) Precess to Release Attitude through Large Probe Separation, (8) Post-Large Probe Separation through Small Probes Separation, (9) Post-Small Probes Separation through Bus Impact.

These nine Mission Phases correlate in time to the NASA-Ames' defined set of Mission Phase that are listed in Table 1.4-1 and are illustrated in Figure 1.4-1. This information was obtained from Reference: Paragraph 1.5.32.

1.5 APPLICABLE DOCUMENTS

1.5.1 PC-455 - Pioneer Venus Project Command and Control Data Processing System Software Specification (CMDS).

1.5.2 PC-454 - Pioneer Venus Project Real-Time Data Processing System Software Specification (TM).

1.5.3 Multiprobe/Orbiter Data Book - Hughes Document No. HS507-5163 - June, 1976.

1.5.4 PC-410 - Multiprobe and Orbiter Spacecraft and related characteristics, Rev. 9 - Aug. 1976.

1.5.5 System Final Design Review - Vu-Graph Data Package - June, 1976 - Hughes Documents No. HS507-5224 through HS507-5228 (5 volumes).

1.5.6 PT-403 - Pioneer Venus Multiprobe Mission Trajectory Characteristics - October, 1976.

1.5.7 PC-430: Bus Spacecraft/Scientific Instruments Interface, Rev. 10, 23 April 1976.

1.5.8 Pioneer Venus Command Assignments List - Hughes Specification No. SS31639-008.

1.5.9 Centaur D-1A Systems Summary - Report No. GDCA-BNZ72-026 September, 1972 - General Dynamics.

1.5.10 Atlas SLV-3D/Centaur D-1A Configuration, Performance and Weight Status Report, dated March 1978.

1.5.11 Centaur D-1A Systems Orientation, 21 April 1976 - GD, Convair.

Revision

1.5.12 Rate, Clock and Status signals required by Pioneer Venus Subsystems and Scientific Instruments, Hughes IDC No. HS507-3738, Rev. A, dated 18 November 1975.

1.5.13 Large and Small Probe Data Book - Hughes Document No. HS507-5164 - June, 1976.

1.5.14 Pioneer Venus Quarterly Review - 8 February 1977.

1.5.15 Mass Properties Report - Hughes Document No. HS507-2002-35, dated 31 January 1978.

1.5.16 P.V. Magnetic Model Update - Hughes Document No. HS507-5803, dated 24 November 1976.

1.5.17 Pioneer-Venus Multiprobe Telemetry Data Conversion Handbook - Hughes Specification No. SS31639-901.

1.5.18 Preliminary Mission Constraints and Spacecraft Signature Logs - Hughes Memo No. HS507-7044 dated 26 August 1977.

1.5.19 Data Rate Capabilities of the P.V. Multiprobe Bus - NASA-Ames Memo #SVS-5-143:244-8 from R. Ramos, dated 18 December 1975.

1.5.20 Pioneer-Venus Technical Proposal - Hughes - Vol. 1 - August 1973.

1.5.21 Pioneer-Venus Criteria of "Hot RF" Switching - Hughes Memo No. HS507-6774, dated 30 June 1977.

1.5.22 P.V. Bus, Orbiter and Probes Electrical Loads Status Report - 11 July 1977, Hughes Memo No. HS507-4725.

1.5.23 Star Sensor Flight Unit #1 Test Report, BBRC S/N 3, Hughes S/N 2, Ball Brothers Test Report dated 22 June 1977.

1.5.24 Multiprobe Star Sensor Final Alignment Data Report, part of Hughes TP31639-001: Multiprobe Performance Assessment Document, dated 7 April 1978.

1.5.25 Pioneer Venus Telemetry Assignments List - Hughes Specification No. SS31639-009.

1.5.26 P-V Sun Sensor Large Sun Test Report - Hughes Memo No. HS507-4865 dated 17 March 1976.

1.5.27 P-V Sun Sensor Output Pulse Delay - Hughes Memo No. HS507-4161, dated 12 November 1975.

1.5.28 Pioneer Venus Spacecraft Time Code - Hughes Memo No. HS507-7155, dated 21 September 1977.

1.5.29 Pioneer Venus Command Timing - Hughes Memo No. 4141.12/42, dated 14 October 1977.

1.5.30 Delay Time in Starting the Command Memory From the 4096 Signal - NASA-Ames Memo #SVS-6-81:244-8, dated 24 August 1976.

1.5.31 Data Rate Capabilities of the Pioneer Venus Multiprobe Bus - NASA-Ames Memo SVS-5-143:244-8, dated 18 December 1975.

1.5.32 PC-470 - Multiprobe Mission Operations Plan Document.

1.5.33 Pioneer Venus Multiprobe STV Test Report - Bus Thermal Control - Hughes Memo No. HS507-7618, dated May 1978.

1.5.34 Recommendations for Stable Oscillator Operation - Hughes Memo No. HS507-0082-2887, dated 17 February 1978.

TABLE 1.4-1

NOMINAL MULTIPROBE MISSION PHASE/PERIOD DEFINITIONS
(NASA-AMES)

NOMINAL PHASE/PERIOD	TIME INTERVAL
1. Test Phase	Composed of Discrete, Separate Periods
1.1 Orbiter Pre-Launch Test Period	Start of Ground Data System (GDS) Acceptance Tests to Start of Orbiter Countdown
1.2 Multiprobe Pre-Launch Test Period	End of Orbiter TCM2 to Start of Multiprobe Countdown
1.3 Post Launch Test Period	End of Multiprobe TCM2 to Start of Orbiter TCM3
1.4 Approach Test Period	End of Bus Maneuver (E-18 Days) to VOI-48 Hours
2. Multiprobe Launch Phase	Start of Countdown Activities to Launch +36 Hours
2.1 Multiprobe Countdown Period	Start of Countdown Activities to Lift-Off
2.2 Multiprobe Post Launch Period	Lift-Off to Launch +36 Hours
3. Multiprobe Cruise Phase	Multiprobe L+36 Hours to Multiprobe Encounter -30 Days
3.1 Multiprobe Spacecraft Calibration Periods	As Required
3.2 Multiprobe TCM1 Period	Launch +5 Days
3.3 Multiprobe TCM2 Period	Launch +20 Days

TABLE 1.4-1. (Continued)

NOMINAL PHASE/PERIOD	TIME INTERVAL
3.4 Bus Instruments Checkout Period	Launch +14 Days
3.5 All Probes and Bus Instruments Checkout Period	Launch +60 Days
4. Multiprobe Encounter Phase	Encounter -30 Days to EOM (≈E+2 Hours)
4.1 Multiprobe TCM3 Period	Encounter -30 Days
4.2 Multiprobe Maneuver Period for Large Probe Deployment	Encounter -28 Days
4.3 Large Probe and Bus Instruments Checkout Period	Encounter -27 Days
4.4 Large Probe Release Period	Encounter -24 Days
4.5 Multiprobe Maneuver Period for Small Probes Targeting	Encounter -23 Days
4.6 Small Probes Checkout Period	Encounter -22 Days
4.7 Small Probes Release Period	Encounter -20 Days
4.8 Bus Maneuver Period for Encounter Targeting	Encounter -18 Days
4.9 Bus Final Maneuver Period	Encounter -48 Hours

TABLE 1.4-1. (Continued)

NOMINAL PHASE/PERIOD	TIME INTERVAL
4.10 Bus BNMS Instrument Calibration and Cap Ejection Period	Encounter -42 Hours
4.11 Probes Encounter and Descent Period*	Encounter -22 Minutes to Impact ($\approx$E+60 Minutes)
4.12 Bus Encounter and Burnup Period*	Encounter -2 Hours to Burnup ($\approx E_{BUS}$+90 Seconds)

*E=0 is defined as the time the individual vehicle is at 200 km altitude above the surface of Venus; hence each probe and the bus will have a different universal time for E=0. For operational planning, the nominal E=0 is for the Large Probe.

86112-9

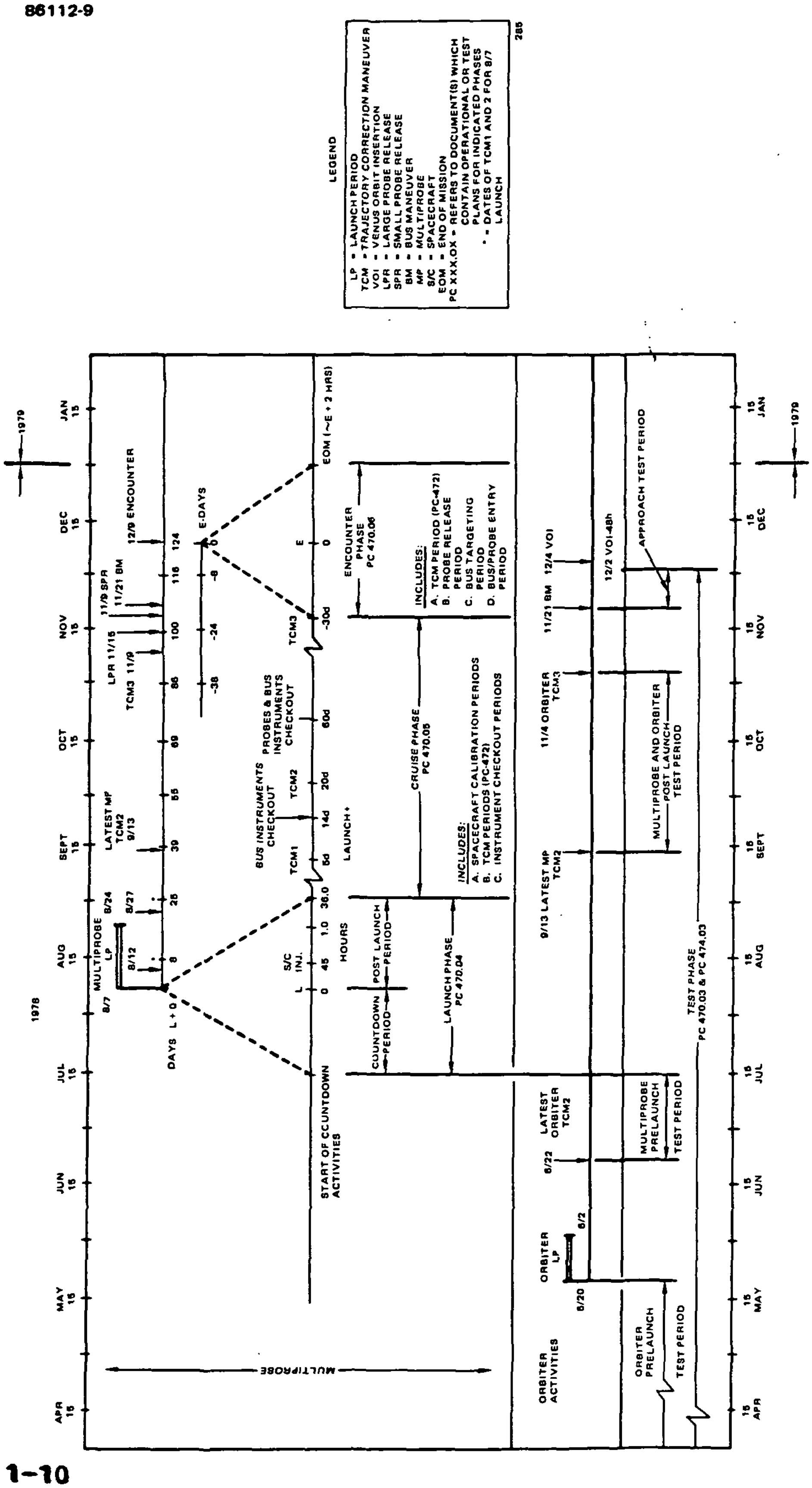


FIGURE 1.4-1. MULTIPROBE OPERATIONAL MISSION PHASES AND PERIODS (NASA-AMES)

FIGURE 1.4-1. MULTIPROBE OPERATIONAL MISSION PHASES AND PERIODS (NASA-AMES)

2.0 SYSTEM DESCRIPTION

System level (overview) descriptions of the nominal mission phases, Multiprobe system interfaces, and the capabilities and limitations of system level performance, are presented ahead.

2.1 GENERAL DESCRIPTION

Each top-level element of the mission (Multiprobe, Launch Vehicle, Ground System, and nominal Mission Plan) is described on a system-level below.

2.1.1 Multiprobe Description. The Multiprobe System (Figures 2.1.1-1 and 2.1.1-2) consists of the Bus, Large Probe, and three identical Small Probes, each of which includes a payload of scientific instruments.

The Multiprobe System travels intact, as shown in Figure 2.1.1-1, from launch until 24 days before Large Probe Entry (≡E), a nominal cruise period of 100 days. The predominate orientation during the cruise period consists of the spin axis (Bus +Z-axis) pointing normal to the ecliptic plane and in a southerly direction. During this period, the Bus Communications Subsystem is used exclusively for uplink and downlink contact with the probes as well as the bus, although all probes have full downlink capability and the Large Probe has limited uplink capability. (These probes' links capabilities are used only after probes' separation.)

2.1.1.1 Bus Description. The Bus consists of the following subsystems and functions: Mechanical function (including the spacecraft structure), Thermal function (accomplished by the Structure/Harness Subsystem), Controls S/S, Propulsion S/S, Data Handling S/S, Command S/S, Communication S/S, and Power S/S.

The Mechanical features of the Bus Spacecraft can be described by five basic assemblies, as seen in Figure 2.1.1-2: Large Probe support structure, Small Probe support structure, equipment shelf, substrate (solar array), and thrust tube. Shape and equipment layout conform to the basic mechanical requirements of a spin stabilized vehicle. The solar cells on the cylindrical solar panel, antennas orientations, and thrust vector orientations provide efficient power, communications, and maneuverability while the Bus is spinning in its cruise attitude.

The thermal design is based on isolating the equipment from the external solar extremes experienced during the mission. (Solar Intensity increases by a factor of 1.98 from Earth to Venus.) Commandable heaters are provided to prevent possible freezing of hydrazine monopropellant, and to make up heat balance should there occur an inadvertent trip of non-essential spacecraft loads. Eleven thermostatically-controlled thermal louvers are mounted on the aft side of the equipment shelf beneath units having high dissipations.

The controls subsystem provides the sensing logic and actuators to accomplish the following stabilization, control, and reference functions:

(a) Spin axis attitude determination (via use of slit field-of-view type sun sensors and star sensors), science roll reference signals generation, and spin period measurement.
(b) Control of thrusters for spin axis attitude maneuvers, spin speed control, and spacecraft velocity maneuvers.
(c) Nutational damping, via use of a partially filled tube of liquid freon E3.

The propulsion subsystem provides the hydrazine monopropellant storage, pressurization, distribution lines, isolation valves, filtering, and thruster assemblies used to accomplish Multiprobe maneuvers throughout the mission.

The Data Handling Subsystem conditions and integrates into command-selectable (choice of four fixed and one programmable) formats all analog and digital telemetry data (197 assigned channels) originating in the bus subsystems and bus science instruments, as well as a few parameters originating in each of the four probes. (Each probe also generates, conditions and integrates an extensive set of parameters which are modulated onto a subcarrier that bypasses the Bus Data Handling Subsystem and instead passes directly to the Bus Communications Subsystem during attached Probe checkouts.) The selected format of the all-digitized data modulates a 16,384 Hz subcarrier at a command selectable (choice of thirteen rates between 8 and 4096 bps) bit rate. The resulting information is routed to the Communications Subsystem for modulation of the downlink S-band carrier.

The Command Subsystem decodes all commands received via the Communications Subsystem at the fixed rate of 4 bps, and either stores the command for later execution or routes the command in real-time to the addressed destination.

In particular, 43 assigned commands dedicated to the Large Probe are partially decoded by the Command Subsystem, and routed to the attached Large Probe, where final decoding takes place.

Each of the remaining 366 assigned commands is either completely decoded (discrete-type command) by the Command Subsystem and the execution command generated; or is partially decoded (quantitative type command) by the Command Subsystem and the command is routed to the addressed destination for final decoding. ("QUANTITATIVE" IS USED THROUGHOUT THIS DOCUMENT TO DESCRIBE BOTH "MULTIFUNCTION" TYPE AND PURELY "QUANTITATIVE" TYPE COMMANDS.)

The Communications Subsystem provides radiation reception and transmission capabilities for the command and telemetry information that modulate S-band frequency carriers (assigned in the 2.115 and 2.295 GHz bands).

There are two redundant reception channels; each includes a hemispherically omni-directional antenna (aft or fwd) that spatially supplements the other to produce total spatial coverage.

The downlink is assignable by command to any one of the aft or forward omni-directional antennas, or to the Medium Gain Horn (Directional) antenna. Its frequency may be slaved to the uplink frequency (phase-locked); or, according to circumstances, it may be a multiple of a crystal oscillator located in the channel receiver. The downlink may also be transmitted via any one of, or some pairs of, four 10-watt power amplifiers.

The power subsystem provides semi-regulated 28-volts ±10 percent to all spacecraft loads (including science instruments).

The primary source of power is the main solar array. When the solar panel output cannot provide adequate power for all spacecraft loads (at low sun angles and during eclipses), the two batteries (each rated at 7.5 amperes full capacity) come on line automatically through the discharge regulators. Battery energy is replenished through a small boost charge array. The power interface unit provides power switching for the propulsion heaters, probe heaters and probe checkout buses; and relay drivers for internal/external power relays in each of the probes. It also contains fuses for these circuits and the bus science instruments input power lines.

Power is distributed on four separate power buses. If a spacecraft over-current condition or under-voltage on either battery occurs, loads are removed to protect the spacecraft from potential catastrophic failure by tripping off buses in the following sequence: Science, switched loads and transmitter. This leaves only those loads that are absolutely essential to spacecraft survival in a continuously powered ON mode.

The RF transmitters and exciters are on the transmitter bus. Controls and data handling units are on the switched loads bus. Instruments and probe checkout power are on the science bus.

Command units, probe and propulsion heaters, power conditioning units, and spacecraft receivers are on the essential bus.

Excitation for the pyro bus is derived from a battery tap located 16 cells (of a total of 24) from the ground reference level.

The bus voltage is limited to 30.0 volts by five shunt limiters that dissipate all excess solar panel capacity in load resistors mounted on the solar panel substrate and equipment shelves.

2.1.1.2 Large Probe Description. The Large Probe travels attached to the Bus, as shown in Figure 2.1.1-1, for most of its journey to Venus. Twenty-four days before its scheduled entry into the Venus atmosphere, the Large Probe is released from the Bus on a planned trajectory to impact. The atmosphere entry sequence is depicted in Figure 2.1.1.2-1. The pressure vessel continues to transmit information about the atmosphere via seven scientific instruments until surface impact, occuring about one hour after entry.

The Large Probe contains: (1) A communications subsystem via which an uplink carrier (no commands) may be received for phase-locking the Probe downlink - to provide precise tracking information; (2) a power subsystem that derives energy solely from a 40-amp-hr. silver zinc battery, once the Probe is released; (3) a command/data subsystem that accepts uplink commands only via the attached Bus; (43 assigned via Large Probe COM; three others additionally assigned); and transmits 49 assigned (via the C/DU) telemetry channels of information (24 analog, 9 serial digital, and 16 bilevel). Additionally, a data storage capability exists for scheduled use during an expected communications blackout upon atmospheric entry. Also, a set of 53 assigned discrete commands exists that are hardwired (PROM) into fixed sequences for probe-internal use from entry until impact; (4) a separable aft cover and a separable deceleration module (including the parachute subsystem depicted in Figure 2.1.1.2-1) for protection and deceleration during atmospheric entry, and (5) seven scientific instruments housed in the pressure vessel, and intended to measure atmospheric properties until surface impact.

2.1.1.3 Small Probes Description. The three small Probes travel attached to the Bus, as shown in Figure 2.1.1-1, for most of their journey to Venus. Twenty days before their scheduled entry into the Venus atmosphere, the three small probes are nearly simultaneously released from the Bus on separate planned trajectories to impact as seen in Figure 2.1.1.3-1.

The three small probes are identical in design. Each contains:

(a) A communications subsystem that provides downlink information only - it contains no uplink receiver,
(b) A power subsystem that derives energy solely from an 11 amp-hr. silver zinc battery, once the probe is released,
(c) A Command/Data subsystem that accepts uplink commands only via the attached Bus, and transmits 36 assigned (via the C/DU) telemetry channels of information (17 analog, 5 serial digital, and 14 bilevel). Additionally, a data storage capability exists for scheduled use during an expected communications blackout upon atmospheric entry, and during an expected loss of downlink signal during a scheduled bit rate change when each Small Probe is at an altitude of approximately 30 km. Also, a set of 40 assigned discrete commands exists that are hardwired (PROM) into fixed sequences for Probe-internal use from entry until impact,
(d) A non-separable deceleration module, a yo-yo despin system, and scientific instrument deployment mechanisms for protection and despin during atmospheric entry, and

(e) Three scientific instruments housed in the pressure vessel, intended to measure atmospheric properties until surface impact.

2.1.2 Launch Vehicle Description. The space booster vehicle for the Pioneer-Venus Mission is the two-stage Atlas SLV-3D/Centaur D-1AR. As shown in Figure 2.1.2-1, the vehicle is comprised of the Atlas booster that includes the interstage adapter, the restartable Centaur, and a nose fairing which encloses the Pioneer-Venus spacecraft. The interstage adapter supports the Centaur atop Atlas and remains with Atlas at separation. The spacecraft is mounted on a payload adapter that mates with the Centaur.

2.1.2.1 Atlas Booster. The Atlas stage-and-a-half concept requires that all five of its engines be ignited before liftoff, while it is held at the launcher, as a reliability measure. This permits an assessment of proper propulsion system operation before commitment to actual flight.

The rated thrust (431,040 lb. total) at sea level consists of 185,000 lb. for each of the two booster engines, 60,000 lb. for the sustainer engine, and 502 lb. for each of the two vernier engines. All five engines use in common the sustainer section tankage that provides RP-1 (liquid hydrocarbon) fuel and liquid oxygen as the propellant components.

From liftoff until separation from Centaur, Atlas is steered by commands from Centaur. SLV-3D electronics are extensively integrated with the Centaur D-1AR astrionics hardware and software systems. The electronics aspects of the ATlas guidance, flight control, sequencing, and telemetry systems are all provided by Centaur.

The engine angle commands are generated by the Centaur Digital Computer Unit based upon the guidance equations, derived roll rates, and Atlas Rate Gyro Unit pitch rate and yaw rate inputs. The commands are sent to the Centaur Servo

Inverter Unit and then to the Atlas Servo Inverter Unit.

These steering commands are executed by gimballing the Atlas engines. The booster engines gimbal for pitch, yaw, and roll control. After booster jettison, the sustainer engine gimbals for pitch and yaw control, and the vernier engines swivel for roll control.

The Booster section is composed of two nearly identical thrust chambers and a power package consisting of two dual turbo pumps that deliver the propellants from the sustainer section tankage. After developing a rated I_{sp} of 257 seconds (sea level), the booster engines are cut off (BECO - at 5.7 g), approximately 139 seconds after the vehicle was released from the launch pad. Three seconds later, the booster section is jettisoned. The booster section is attached to a thrust ring near the aft end of the tank assembly by 10 pneumatically operated separation latches. Separation is activated by a command from the Centaur guidance system via the Centaur flight control subsystem, and backed up by a staging accelerometer.

The sustainer section houses, within an average 10 foot diameter stainless steel frame, the propellants and tankage. The sustainer engine is gimbal mounted at the aft end of the fuel tank. The vernier engines are gimbal mounted on opposite sides of the tank structure. Liquid oxygen feed lines, tank pressurization lines, and electrical cable fairings are attached to the outside of the tank structure. Equipment pods containing electrical and electronic units are also attached to the outside of the tank structure..

The engines burn from liftoff until propellant depletion occurs, approximately 252 seconds after liftoff. The sustainer engine delivers a rated I_{sp} of 312 seconds (vacuum). The Centaur insulation panels are jettisoned 45 seconds into the sustainer phase (period between Booster engine cutoff and vernier engine cutoff (VECO)), followed by nose fairing jettison.

About two seconds after sustainer engine cutoff (SECO), the pyrotechnic system releases the Centaur from the interstage adapter. Separation is achieved by firing the retrorockets mounted on the aft end of the Atlas sustainer tank.

2.1.2.2 Centaur. The Centaur D-1AR vehicle has some basic redundancy features that the previous D-1A vehicle class did not have.

The main feature of Centaur is the ability to deliver two or more continuous firings of its engines separated by automatically-controlled coast periods.

The Centaur vehicle has a nominal diameter of 10 feet and is 31 feet in length. This lightweight structure, combined with the high specific impulse (I_{sp} = 439 seconds minimum) of the liquid hydrogen/liquid oxygen main engines, makes it a very efficient vehicle for high energy missions.

Primary thrust is provided by two Pratt and Whitney RL10A-3-3 engines that develop 29,500 lb. total thrust.

Two identical and separate hydraulic power supply systems provide the force to gimbal the Centaur main engines, one system for each engine.

During coast, separation, and retromaneuvers, attitude control and propellant settling are provided by a hydrogen peroxide-fueled thrust system. This system consists of small engines (6 to 24 lb. thrust) attached to the aft bulkhead.

The Centaur D-1AR astrionics system integrates many former hardware functions into the airborne computer software. Digital autopilot, maneuvering attitude control, sequencing, telemetry formatting, propellant management, plus guidance and navigation are all within the flexible system software scope.

Telemetry data from information sensors is converted to digital words for transmission to the ground station via an S-band transmitter.

The C-band tracking system provides data to determine position and velocity information for use by range safety at the eastern test range.

The Centaur airborne transponder returns an amplified radio-frequency signal when it detects a tracking radar's interrogation.

2.1.2.3 Atlas/Centaur Flight Sequence. The sequence described in Sections 2.1.2.1 and 2.1.2.2 are illustrated in Figure 2.1.2.3-1 and summarized chronologically in Table 2.1.2.3-1. The times for the listed events are estimated as of June 1976, taken from Reference Paragraph 1.5.10. The actual times for most of the events after launch will vary slightly from those shown, based on Atlas/Centaur weight, engines performance, propellant loading, payload weight, launch day, launch time of day and launch azimuth.

2.1.3 Ground System Description. The Ground System used during the mission for Command, Telemetry and Tracking is described ahead.

2.1.3.1 General. The Pioneer Ground Data System (PGDS) is used to receive, process, record and display spacecraft telemetry and tracking information; and to transmit commands to the spacecraft. Its basic elements are shown in the overview diagram of Figure 2.1.3-1. The PGDS is comprised of the NASA Deep Space Network whose stations are located around the world (refer to Figure 2.1.3-1), the Network Control System (NCS) located at JPL-Pasadena, and the Pioneer Mission Operations Center (PMOC) located at the NASA-Ames Research Center - Moffett Field, California.

The Deep Space Stations (DSS) are responsible for the RF Communications link with the Pioneer Spacecraft.

Command and telemetry data are block formatted via standard DSS ground Transmission formatting and routed directly between the DSS and the PMOC via Network Control at JPL. Communications is accomplished through the Ground Communications Facility using the High Speed Data lines (HSD) and voice/data lines.

Tracking data is processed at JPL to accurately determine spacecraft present location and predict future flight path, whereas spacecraft attitude determination and prediction is accomplished at PMOC.

Two Sigma 5 Computers are dedicated for on-line real time data processing and display at PMOC. A third Sigma 5 is used for offline specialized processing of data.

The PDP-11 Computer systems at PMOC encode spacecraft commands, messages, and format them for the HSDL.

2.1.3.2 PGDS Telemetry System. RF signals received at a DSS are detected, and the telemetry subcarrier demodulated. The serial bit stream is synchronized, and the detected bits formatted for HSD lines. When the spacecraft data are convolutionally encoded, the segmential decoding process is employed. Display of selected engineering data for station operations takes place at some sites.

All telemetry data, together with associated DSN status data, are transmitted to PMOC via HSD. The PMOC has the capability to process and display data in real time from one or more DSS and spacecraft simultaneously without interference.

2.1.3.3 PGDS Command System. Command, mode control, and recall data are generated at the PMOC and formatted for transmission via HSD lines to a DSS. Timed commands are transmitted at the time defined, whereas non-timed commands are transmitted in the order defined in the HSD command messages. The DSS generates and formats acknowledge, alarm, configuration, abort, and recall response messages for transmission via HSD to the PMOC.

Initialization of the Command Systems takes place from the NCS, although backup initialization can be accomplished at the DSS in the event of an NCS failure. In order for the NCS to have access to the Command System at the DSS over the same single HSD line which interfaces with the PMOC, a special piece of hardware is required which replaces filler blocks coming from the PMOC with out-going messages from the NCS. There are three filler multiplexers (FMXR), two on-line and a spare. When station handovers require a third HSDL into the PMOC, the spare FMXR is utilized if available. If the spare FMXR is not available, then coordination is necessary between the PMOC and DSN operations control during time periods when both Multiprobe and Orbiter spacecraft are being tracked and a station handover occurs because of the communications switching that is required.

The command message construction, verification, and HSD block formatting functions are performed by the PDP-11 computer systems located in the PMOC. Response message blocks returning from the DSS are routed to the PDP-11 for verification, and to the Sigma-5 system for post-transmission processing. Mode change and recall request messages are generated by the PDP-11.

2.1.3.4 PGDS Tracking System. The tracking system provides and processes precision radiometric data for determination of the present and future flight path of the spacecraft. This information is used for specific DSS tracking and specific mission planning.

The operating DSS provides measurements to the NCS at JPL of:

(a) Integrated Doppler
(b) Spacecraft range
(c) Differenced range versus integrated Doppler
(d) Antenna pointing control and angle data

(e) Frequency and timing information
(f) Ground weather data measurements, and
(g) Various station status information such as receiver lock status.

The Network Control System at JPL selects, calibrates, and then uses the radiometric data for generation of present corrected spacecraft flight path, predicted spacecraft flight path, and predicted DSS tracking requirements. This information is then routed to cognizant DSS, and mission control at PMOC. The predicts are also compared with incoming data for use in an iterative-correction cycle to increase the accuracy of the iterative-correction cycle to increase the accuracy of the predicts.

2.1.4 Mission Description. An overview Mission description and a description of significant events are given below.

2.1.4.1 Overview Mission Description. The Multiprobe Mission will last approximately 110 to 120 days, from launch (circa August 1978) thru impact of the 4 probes and the bus upon the surface of Venus. All will travel intact until 24 days before impact, when the Large Probe will be released. Four days later, the three Small Probes will be released. All five devices will enter the atmosphere of Venus at scattered points across the Venus' hemisphere that is centered on the sub-earth point (refer to Figure 2.1.1.3-1).

While all are intact, all two-way communication will be via the Multiprobe Bus. From each Probe Release until Probe's entry into the atmosphere, two-way communication is available only with the Multiprobe Bus. From approximately E-22 minutes (E ≡ Large Probe entry into the Venusian Atmosphere = +200 km Altitude) until Bus vehicle burnup in this atmosphere (E+96 minutes to E+112 minutes, depending on launch date and communications look angle) one to five of the devices are telemetering information directly to the earth simultaneously (i.e., as many as five downlinks simultaneously).

2.1.4.2 Mission Events Description. The significant mission events within each of the nine adopted Mission time phases are described below.

2.1.4.2.1 Prelaunch Through Spinup to 15 RPM (L-5 Minutes to L+5 Hours). Approximately five minutes before launch, the Multiprobe Spacecraft electrical loads will be switched to internal battery power.

The Multiprobe spacecraft will be launched with an Atlas SLV-3D/Centaur D-1AR launch vehicle (AC-52) from CCAFS, Florida. The launch will take place during a 40-minute window on one of 28 successive days in the period from 7 August through 3 September 1978. The launch vehicle will place the Multiprobe on the desired Type I interplanetary trajectory.

During the launch phase, spacecraft engineering telemetry will be transmitted via low transmitter power from the forward omni antenna at nominally 256 bps rate. Immediately prior to separation, the Centaur will orient the spacecraft in a normal-to-the-ecliptic attitude with the positive spin axis in the direction of the South ecliptic pole. The separation switches will initiate redundant command sequences stored in the two command processor memories that results in spacecraft spin-up to nominally 15 rpm, and switchover to the aft omni antenna, for transmission during initial ground station acquisition.

It is expected that ground station acquisition from the Honeysuckle DSS will occur nominally 1 hour after launch, at which time the spacecraft bit rate will be increased to 2048 bps nominally, by ground command. Shortly after the rapidly-changing communications look angle becomes favorable for transmission via the forward omni-antenna, switchover via ground command will take place.

The shelf heater on the Large Probe will then be turned ON to operate continuously until nominally L+87 days, when the increased solar intensity will be sufficient to maintain the Large Probe at a thermally safe level.

2.1.4.2.2 Attitude Determination and Star Survey (≈L+5 Hours to L+1 Day). The scheme for attitude determination at this position will employ the use of at least two detectable stars. The star survey portion of the attitude determination effort may be repeated several times during the transit to Venus for sensor bias calibration.

2.1.4.2.3 Jet Calibration, TCM #1, Bus Science Checkout, and Cruise (L+1 Day to L+20 Days). As early as L+1 day, in-flight calibration of the axial and radial thrusters will be performed in preparation for the first trajectory correction maneuver (TCM #1) at L+5 days. For a known duty cycle (spin rate and pulse width selection) and number of pulsed firings for each required thruster; the impulse and alignment will be determined - for efficient thruster(s) usage during TCM #1.

TCM #1 requires a ΔV of 12.1 m/sec (worst case) to correct launch vehicle injection errors. The maneuver may be performed in the existing normal-to-the-ecliptic attitude via use of the radial thrusters in pulsed mode (and possibly in addition, the axial thrusters in pulsed or continuous mode); or, after precession around the sunline, in an attitude that requires use of the axial thrusters only (in pulsed or continuous mode). The selection will depend upon which mode minimizes propellent usage.

Spacecraft transmitter high power and a 64 meter ground station will be used during the calibration and during TCM #1, with cruise status applying before, between, and after these events.

The Bus Science will be initially checked out at L+14 days, nominally with the transmitter in high power mode, telemetry at 1024 bps, and use of a 26 meter (Dish) Ground Station.

2.1.4.2.4 TCM #2 and Cruise (L+20 Days to L+60 Days). Nominally at L+20 days, the second TCM scheduled to correct for residual velocity errors will be performed. TCM #2 will be functionally the same as TCM #1, except the nominal telemetry bit rate will be 1024 bps for the forward omni. Nominally, the ΔV will be small in magnitude compared to TCM #1, and accomplished in the cruise attitude.

Post-TCM #2 cruise status will apply until L+60 Days.

2.1.4.2.5 Probes Checkout (L+60 Days to ≈ L+87 Days (=E-40 Days, Where E=Large Probe Entry at 200 km Altitude)). All four probes are scheduled for the first post-launch checkout at L+60 days, using Multiprobe Bus power (the probes' batteries are not rechargeable in flight). A 64 meter ground station will be used, as well as transmitter high power. Most of each Probes' data will be applied to the Bus downlink carrier via a probe unique subcarrier, in addition to Bus telemetry data being carried by its own subcarrier.

Cruise status will apply after Probes' checkout until nominally L+80 days, when the Bus transmitter will be switched to high power mode to maintain adequate R.F. link margin due to increased range. Increased solar intensity will support transmitter high power operation and cruise status loads, and still have surplus solar panel power for the cruise attitude (The Bus batteries have been maintained at full-charge via a trickle charge rate since spinup, except for a small discharge during Probes' checkout).

Nominally 25 days before small probes separation (i.e., at ≈ L+82 days = E-45 days), the stable oscillator in each small probe will be turned ON and left ON continuously until just prior to small probes separation. This "bakeout" is to ensure that each oscillator will meet specified performance when required during that small probe's entry and descent (Reference: Paragraph 1.5.34).

2.1.4.2.6 Pre-TCM #3 Thru Horn Attitude Operations (E-40 Days to ≈ E-24D 6H). At ≈L+84 Days (≈E-40D), the Large Probe shelf heaters will be commanded OFF.

Nominally at E-30 days, the third TCM scheduled to correct for residual velocity errors will be performed. TCM #3 will be functionally the same as TCM #2, except the nominal telemetry bit rate will be 64 bps.

At ≈E-28 days, the Forward Axial Jet Heater is turned OFF just before the spacecraft spin axis is precessed to an attitude in the ecliptic plane that provides usage of the medium gain horn antenna. (The OMNI antenna, transmitter high power mode, and 26 meter Ground Station combination does not provide sufficient downlink received signal strength to the Earth-based ground receivers to receive even the lowest available telemetry bit rate (8 bps) after E-28 days. Therefore, the medium gain horn is employed).

The sun line-of-sight (l.o.s.) with w.r.t. (with respect to) the spacecraft +Z axis (Aspect Angle) is reduced to approximately 50°; however, there is still a substantial solar panel margin in the Horn Attitude.

The bus transmitter is returned to low power mode for approximately 3 days, then switched to High power at ≈LPR-12 Hours (12 hours prior to Large Probe release - occurring at E-24D), for the last Large Probe checkout.

After the last Large Probe checkout, the bus transmitter is returned to Low power for ≈3 hours, while the batteries are recharged at the high charge rate. (Worst case conditions for Large Probe checkout result in approximately 60 percent Batteries DOD. This is replenished nominally in a few hours of High rate charging.)

2.1.4.2.7 <u>Precess to Release Attitude Through Large Probe Separation (≈E-24D 6H to E-24D)</u>. At ≈6 Hours before Large Probe release, the bus transmitter is again switched to high power, the Large Probe is switched to internal power, and the Large Probe coast timer is loaded and initiated.

The release attitude is selected so that the Large Probe will enter the atmosphere with a zero angle of attack.

The spacecraft transmitter is switched to the aft omni, and the spacecraft is then precessed nominally $\Delta 44°$ to the Large Probe release attitude. The precession is performed at least 4 Hours before release in order to allow enough time for attitude determination (and attitude trim and spin rate trim, if required).

The Large Probe is released nominally at E-24 Days.

2.1.4.2.8 <u>Post-LP Separation Through Small Probe Separation (E-24 Days to E-20 Days)</u>. Immediately after Large Probe release, the spacecraft is precessed to the Small Probes targeting attitude that permits use of the Horn in low transmitter power. The batteries are switched to High Charge Rate.

At ≈E-23 days, the Multiprobe will be spun up to 48.5 rpm and a pulsed radial jet ΔV maneuver will be performed to effect a ΔV of 5.1 m/sec to achieve the desired Small Probe targeting. The 48.5 rpm spin rate will impart lateral momentum sufficient to achieve the target points shown in Figure 2.1.1.3-1.

The last checkout of the small probes will be performed, including turn OFF of each stable oscillator.

About 7 hours before small probes' release, the spacecraft will be switched to xmtr low power and 128 bps TM rate. Then, the spacecraft will be processed to an intermediate attitude (sun look angle = 28°; comm look angle = 30°) that permits the star sensor to cool down from it's exposure to the sunline in the previous attitude while maintaining a power balance. (The Small Probes' release attitude also would permit cooling of the star sensor, but at the expense of discharging batteries).

About 3 hours before Small Probes' release, the spacecraft will be precessed to the release attitude (sun look angle = 28°; comm look angle = 31°). The cooled star sensor will be used for verification of release attitude.

The three Small Probes are released nominally at E-20 days.

2.1.4.2.9 Post-SP Separation Thru Bus Communications Blackout (E-20 Days to ≈ E+100 Minutes). Immediately after Small Probe release, the spacecraft will be precessed back to the S.P. targeting attitude which allows use of the medium gain horn and provides a sun angle of 40 degrees. The transmitter is returned to lo power.

At ≈E-18 days, the transmitter is switched to hi power on the aft omni, and the spacecraft is precessed to the Bus Targeting attitude. (The transmitter will remain at hi power for the duration of the mission.) Then, a ΔV maneuver of 19.1 m/sec will move the trajectory aim point to that desired for Bus entry and slow the occurrence of Bus entry to be 85 ±5 minutes after entry of the last small probe.

Immediately after the Bus Targeting, the spacecraft spin axis will be precessed back to the post-SP release Horn attitude for adequate communications coverage via a 26 meter DSN ground station.

At ≈E-8 days, following a possible Bus targeting trim, the spacecraft spin axis is precessed to the Bus entry attitude.

The Bus is despun to 9.45 RPM, a science instruments requirement for the Bus Entry through impact time period, at E-2 days.

Events occurring just prior to Large Probe Entry (≡E) through Bus Entry are listed chronologically in Table 2.1.4.2.9-1, as well as their nominal times of occurrence and the rationale. This information is valid as of 24 April 1978 per NASA-Ames.

2.2 INTERFACES

Interfaces between the multiprobe bus and the launch vehicle, probes, and scientific instruments, as well as interactions between bus subsystems, are described below.

2.2.1 Multiprobe/Launch Vehicle Interface.

<u>Multiprobe/Launch Vehicle Interface</u>. The Multiprobe combined integrated system interfaces with the Atlas SLV-3D/Centaur D-1AR, two-stage launch vehicle. To the top of the Centaur mounts a one-piece spacecraft attach fitting which interfaces with the spacecraft. The spacecraft attach fitting remains with the Centaur at Spacecraft separation. Figure 2.2.1-1 shows the Multiprobe combined integrated system as encapsulated within the payload fairing (shroud).

2.2.1.1 Mechanical Interfaces

<u>Station and Coordinate System</u>. The relationship of the integrated launch system coordinate system and Station location in the region forward of the Centaur equipment module is shown on Figure 2.2.1.1-1.

<u>Payload Fairing</u>. The basic fairing is composed of a 120-inch diameter cylindrical afterbody, 208.33 inches in length, and a 194.74-inch long conical forebody. The allowable spacecraft dynamic envelope provided by the payload fairing is shown in Figure 2.2.1-1. For on-stand access to the spacecraft, two fairing doors will be provided for access to the Multiprobe umbilical connector area, battery and pyrotechnic flight plugs and star sensor. Except for the fairing split-line latches, the fairing provides rf transparency forward of Centaur station 137.7 because of its fiberglass construction. Extending aft of this Centaur station, a 56-inch long aluminum cylindrical section imposes some rf opacity.

<u>Spacecraft Attach Fitting</u>. The spacecraft attach fitting shown in Figure 2.2.1-2 is the intermediate structure between the launch vehicle and the Multiprobe. It is 32-inches in height

with a top diameter of 44 inches and a bottom diameter of 63 inches. The fitting interfaces the Centaur equipment module forward attach ring at Centaur station C163.90 or Spacecraft station 8.0. The spacecraft attach fitting will be vented during ascent through the atmosphere to provide essentially a zero pressure differential. The spacecraft is secured to the attach fitting forward ring before launch by a two-piece Marman type separation clamp located at Spacecraft station 40.0. Two pyrotechnic separation bolts located 180 degrees apart which hold the clamp assembly in tension are actuated by signals from the Centaur. Upon actuation of either or both of the explosive bolts, band tension is released permitting the clamp assembly to be pulled down onto the attach fitting. Separation of the spacecraft from the attach fitting is effected by the relaxation of four springs equally spaced around the perimeter of the forward ring. There are four separation switches located 90 degrees apart, adjacent to each separation spring location, two of which provide separation signals to the Centaur. The spacecraft attach fitting also contains bracketry and nut plates for installation of GFE accelerometers, pressure transducer, temperature transducer, and associated harnesses.

Aft Actuator Fitting Interface. The spacecraft attach fitting/launch vehicle mechanical interface occurs at the aft actuator fitting interface. Two actuator spring fittings are located 180 degrees apart on the attach fitting. These serve as part of the fairing separation function. Two compressive springs apply a load against these spring fittings to "kick" the lower end of the fairing clear of the launch vehicle. The helper spring fitting shown in Figure 2.2.1.1-2 provides a direct load path to the aft ring.

Electrical Interface Connector Bracket. The spacecraft attach fitting/launch vehicle mechanical interface also occurs at the airborne electrical interface connector bracket. This connector panel provides a break point for the electrical interface between the Centaur and the spacecraft attach fitting.

2.2.1.2 Electrical Interface. This interface occurs in the connectors mounted to spacecraft attach fitting electrical interface connector bracket. These connectors pass the firing currents from the Centaur used to actuate the pyrotechnic bolts in the Marman type separation clamp and causes subsequent separation of the spacecraft from the spacecraft attach fitting. Wiring from two separation sensing switches mounted on the spacecraft attach fitting near spacecraft Station 40 pass through these connectors for telemetering by the Centaur. Other circuitry routed through this interface connector pass through GPE harnessing which terminates in GPE hardware.

Unlike other spacecraft which commonly telemeter information via the Centaur transmission system, the Multiprobe Spacecraft has no such telemetry interface.

2.2.2 Bus/Probes Interface. The Bus provides the means of communication, and the necessary environment for maintenance and checkout of the attached probes as described ahead. Once detached, the probes cease to interface with the bus in any manner.

2.2.2.1 Mechanical Interface. The probes are attached mechanically to the Bus as seen in Figures 2.1.1-1 and 2.1.1-2.

The Bus Spacecraft Large Probe support structure (an inverted conical frustrum) is the primary support structure for the Large Probe and a primary load path for the Small Probe support Structure. The Large Probe support and thrust tube structure are composed of magnesium skins, aluminum rings and longerons. Three explosive bolts are used to attach the Large Probe to the Bus until Large Probe separation, at which time separation springs between the Bus and Large Probe act to produce the desired L.P.ΔV.

The Bus Small Probe spacecraft structure is an aluminum honeycomb panel (shear web) supported at the outer edge by two aluminum tubular struts which terminate at the equipment shelf. Each Small Probe is held attached to the bus via a fly-away clamp. After the In-Flight Disconnect cable on each probe has been disconnected by command, Probes' separation is accomplished with centrifugal force acting on each probe and its fly-away clamp.

2.2.2.2 Thermal Interface. Each probe has self-contained passive and active thermal control in its design, including forward and aft shelf heaters for the Large Probe and a forward shelf heater for each of the Small Probes. The shelfs heaters circuitry is designed to be powered only from the attached Bus by command to conserve the limited, non-replenishable energy in each probe battery.

2.2.2.3 Command Interface. All ground originated commands to the attached probes involve the use of the Bus Communications S/S and the Bus Command S/S.

Commands to the Large Probe may be routed either to a dedicated COM (No. 7) located within the Large Probe, or from several Bus COMs to the Large Probe C/DU for final routing. COM No. 7 has a capacity of 64 discrete commands and 4 quantitative commands. Presently, 18 discrete commands and 1 quantitative command are assigned for the Large Probe subsystems via COM No. 7 (excludes scientific instruments). Two discrete commands and one quantitative command emanate from several Bus COMs and pass to the Large Probe subsystems via the IFD. (All electrical power and signal lines from the Bus to each probe pass via the probe's IFD as seen in Figures 2.2.2.3-1 and 2.2.2.3-2.)

Commands to each Small Probe emanate from several Bus COMs to the SMALL Probe C/DU for final routing. Thirteen discrete commands and one quantitative command are presently assigned to each Small Probe in this category (excludes instruments). Probes' checkout power, heater power and pyro signals are commanded to Bus Subsystems that provide the power and signals via lines to each probe, and are not technically commands that go directly to the probes.

2.2.2.4 Data Handling and Timing Signals Interface. Allocated and presently assigned numbers of telemetry channel types for the attached probes' subsystems, via use of the Bus Communications S/S, each probe's C/DU, and each probe's subcarrier, are as follows:

SUBSYSTEM TELEMETRY (EXCLUDES INSTRUMENTS)		TELEMETRY CHANNEL TYPE		
		ANALOG	SERIAL/ DIGITAL	BILEVEL
Large Probe	Allocated	22	4	15
	Assigned	15	2	9
Small Probe (Each)	Allocated	25	4	16
	Assigned	10	2	8

Additionally, seven channels from each of the four probes are routed directly to the Bus DIMs in the Bus Data Handling Subsystem for processing. Status of each Probe heater and status of checkout power are telemetered as part of the Bus Power Subsystem. Stowed/Released status of each probe is telemetered as part of the Bus Structures/Harness "subsystem".

Timing signals provided by the Bus to the probes' subsystems (excluding probes' science instruments) are:

(a) Read clock
(b) Analog read envelope
(c) Digital read envelope

2.2.2.5 Power and Pyro Interface. By command to the Bus Power Subsystem, the Bus provides to the attached probes each of the following:

SUBSYSTEM POWER (EXCLUDING INSTRUMENTS)	LARGE PROBE	SMALL PROBE (EACH)
Checkout Power	350 ma.	400 ma.
Heater Power	780 ma.	75 ma.

Bus PCU No. 1 provides the commandable control to fire squibs that cut the Large Probe IFD and release the Large Probe. Bus PCU No. 2 provides the commandable control to fire squibs that cut each Small Probe IFD and release each Small Probe.

2.2.3 Interfaces with Scientific Instruments. Interfaces between the Bus and all scientific instruments onboard the Bus and Probes, are described below.

2.2.3.1 Bus/Bus Instruments Interface. The Bus provides the necessary operating environment and means of communication for the Neutral Mass Spectrometer (BNMS) and Ion Mass Spectrometer (BIMS) as described below.

2.2.3.1.1 Mechanical and Thermal Interface. The Bus accommodates the two scientific instruments mechanically by providing footprint areas and inserts on the shelf forward side. Both instruments are mounted on Hughes supplied aluminum brackets to provide the necessary fields of view as shown in Figures 2.2.3.1.1-1 and 2.2.3.1.1-2.

Thermal Control of both instruments is accomplished by use of passive materials.

2.2.3.1.1.1 Neutral Mass Spectrometer (BNMS). The BNMS sensor is mounted on a 4" high bracket to elevate the instrument above the equipment shelf for a clear 120-degree-conical FOV. The pointing direction of the instrument axis is 5 degrees from the +Z axis and its location on the shelf is at θ=343 degrees. The electronics unit is mounted directly to the shelf and inboard of the sensor. Both the electronics unit and the sensor are mounted over a thermal louver.

The spacecraft upper thermal blanket, along with the conducting surface, interface with the sensor at the "Z" ring provided around the sensor neck.

2.2.3.1.1.2 Ion Mass Spectrometer (BIMS). The BIMS is mounted on a 11.685 inch high tripod bracket and located on the equipment shelf at θ = 120 degrees. The bracket height along with the overall instrument height, position the viewing aperature above the probe supporting structure to obtain the required 2π steradian FOV. The pointing angle of the instrument is parallel to the spacecraft +Z axis.

Thermal control of the instrument is afforded by a multilayer blanket which extends from the spacecraft forward thermal barrier to just under the instrument aperture. The spacecraft conducting surface also interfaces with the instrument at this same point.

2.2.3.1.2 Command Interface. There are 8 discrete commands and 1 quantitative Command assigned for both instruments via use of the Bus Communications S/S and Bus Command S/S.

2.2.3.1.3 Data Handling and Timing Signals Interface. Assigned telemetry channels for both instruments, via use of the Bus Communications S/S and the Bus Data Handling S/S, consist of 4 serial digital, 2 analog, and 2 bi-level channels.

Additionally, temperature of each instrument is telemetered as part of the Bus Structures/Harness "subsystem" (SBNMST and SBIMST), and the instruments' collective load currents are monitorable via PSCICI.

Timing and status Signals provided by the bus to some of the scientific instruments consist of the following:

(a) word rate
(b) minor frame rate
(c) major frame rate
(d) read clock
(e) analog read envelope
(f) digital read envelope
(g) 32,768 Hz clock
(h) 2,048 Hz clock
(i) bit rate status
(j) Roll Index pulse (RIP)

2.2.3.1.4 Power and Pyro Interface. The Bus spacecraft provides, via a fused line to each Bus instrument from the 28 vdc science bus, measured operating power as shown in Table 2.2.3.1.4-1.

TABLE 2.2.3.1.4-1

BUS SCIENCE INSTRUMENTS MEASURED OPERATING POWER

INSTR	TURN-ON AND STEADY STATE CURRENT:(AMPS)	PEAK CURRENT (AMPS)	FUSE SIZE (AMPS)
BIMS	.05	.05	.25
BNMS	.18	0.7	1.0

Bus PCU #1 provides the commandable control to fire squibs that eject the sealing cap from the BNMS ion source entrance (releases the BNMS breakoff Hat (ORD15 and ORD16)). Bus PCU #2 provides the commandable control to fire the squib to break a glass capsule that releases the BNMS Calibration GAS (ORD24 or ORDB4).

2.2.3.2 Bus/Probes Instruments Interface. As for the probes subsystems (Section 2.2.2), the bus provides the means of communication and the necessary environment for maintenance and checkout of the attached probes and probes instruments as described ahead. Once detached, the probes and probes instruments cease to interface with the bus in any manner.

2.2.3.2.1 Mechanical Interface. The probes' instruments are rigidly fastened to the probes' structures. There is no special mechanical interface between the bus and probes' instruments beyond the general interface between the bus and probes that is described in Section 2.2.2.1.

2.2.3.2.2 Thermal Interface. The thermal interface with the attached probes' instruments is the same as that described for the attached probes in Section 2.2.2.2.

2.2.3.2.3 Command Interface. All ground originated commands to the probes' instruments, while the probes are attached to the bus, involve the use of the Bus Communications S/S and the Bus Command S/S.

The description of the Command Interface between bus and probes' instruments is the same as applies for the interface between bus and probes in Section 2.2.2.3, except the numbers of commands assigned to the probes' instruments are different. Presently, 23 discrete commands and 1 quantitative command are assigned for the Large Probe scientific instruments via COM #7. No commands via the other COMs are assigned to the Large Probe instruments.

Commands to each Small Probe instrument emanate from several Bus COMs to the Small Probe C/DU for final routing. Fourteen discrete commands are presently assigned to the instruments only of each Small Probe.

2.2.3.2.4 Data Handling and Timing Signals Interface. Assigned numbers of telemetry channel types for the attached probes' instruments, via use of the

Bus Communications S/S, each probe's C/DU, and each probe's subcarrier, are as follows:

INSTRUMENTS TELEMETRY (EXCLUDES PROBES' S/S TM):	ASSIGNED TELEMETRY CHANNEL TYPES		
	ANALOG	SERIAL DIGITAL	BILEVEL
Large Probe	9	7	7
Small Probe (each)	7	3	6

No other telemetry about the probes' instruments directly are routed to the Bus by any other means than that above.

No timing signals are routed from the Bus to the probes' instruments directly. Major and Minor frame rates, read envelope, read clock, word rate, and 2048Hz clock are routed for each probe C/DU to the probes' instruments, however.

2.2.3.2.5 Power and Pyro Interface. By command to the Bus Power subsystem, the Bus provides power to the attached probes' instruments via the same checkout power line going to each probes' subsystems.

INSTRUMENTS POWER (EXCLUDES PROBES' S/S POWER REQUIREMENTS)	LARGE PROBE (ALL INSTRUMENTS ON) NOTE (1)	SMALL PROBE (EACH) (ALL INSTRUMENTS ON)
Checkout Power	106 watts	10 watts
Note (1): All instruments ON simultaneously is a worst case condition for Checkout. Nominally, the instruments are checked out sequentially.		

There are no squibs associated directly with the Probes' instruments that are fired via the Bus.

2.2.4 Interactions Between Subsystems. The fundamental interfaces between the previously listed Bus Subsystems can be classified as mechanical, thermal, and electrical.

The mechanical interface between bus subsystems is essentially rigid and unchangeable, i.e., nothing exists in the mechanical design that is deployable, or reorientable by ground control. The four probes that are mechanically detached from the bus approximately twenty days before Venus encounter are external to the bus subsystems by definition.

The thermal interface between bus subsystems, resulting from the primarily passive thermal design that includes nearly all subsystem equipment mounted to a single shelf, tends to be distributive in nature for cold case conditions, and localized in nature for hot case conditions. The shelf-mounted equipment is enveloped in a compartment bounded by the forward side of the shelf, inner wall of the solar panel substrate, and the aft side of the forward blanket that isolates the equipment from a space environment. If the temperature sensed at two thermostats on the RF shelf half drops below 60°±5°F and there is sufficient excess of solar panel power, four heaters on the RF shelf half and four heaters on the battery shelf half are powered ON, providing a uniform heating of the equipment. If the temperature sensed at two thermostats on the Battery shelf half drops below 35°±5°F and there is sufficient excess of solar panel power, four heaters on the Battery shelf half and four heaters on the RF shelf half are powered ON.

All shelf-mounted units which dissipate more than two to three watts are located over themostatically-controlled louvered radiator areas. There are 11 thermal louvers distributed over the AFT side of the shelf, each one controlled by its own locally-mounted thermostat. When the temperature rises above 55°±5°F at a thermostat, the louver begins to open. At 80°±5°F, the louver is fully open, radiating heat away from the local hot spot. With such a locally-controlled thermal radiation design, it is seen that the thermal interface between subsystem equipments is minimal in the hot case (equipment from the same subsystem are generally mounted on the shelf in close physical proximity).

All subsystems electrical interfaces, excluding thermal, and including probes and bus Science, are summarized in the interaction matrix, Table 2.2.4-1. The power subsystem interface with other subsystems has been described in Section 2.1.1. The remaining subsystems have numerous interfaces that are primarily electrical.

The numbers in Table 2.2.4-1 refer to the following electrical interface items:

1 Power (= nominal cruise power)

2 Commands: ((D) = Discrete; (Q) = quantitative - <u>excludes redundant channels</u>).

2a These are all the unique commands that pass between the Communications Subsystem and the Command Subsystem at 48 bits per command, 4 bps, and on an FSK subcarrier ("0" tone = 100 Hz; "1" tone = 250 Hz).

2b These are all the unique commands that pass from one or more Command Subsystem COMs to the first affected subsystem.

2c These are all the commands that are processed in the first affected subsystem and pass to the final affected subsystem for terminal action. Most of the time, the first and final affected subsystems are the same.

NOTE: Some single commands can finally affect two or more subsystems, and are counted for each such subsystem, whereas, they are each counted only once in category 2b. Therefore, the total of all 2c commands from the first affected subsystem does not always equal the quantity of 2b commands for that subsystem. An example of such a command is: (ORD13 or ORDA3): ARM Large Probe IPD and BNMS B/O HAT. This command is indicated as a 2c command for each of the Large Probe and the BNMS, but it is counted only once as a 2b command in the first affected subsystem (The Command Subsystem).

The sum total of all 2b commands does equal the 2a commands. Again, redundancy in COM assignments is not included. This is a total of unique commands.

3 Telemetry (= number of parameters, either analog (AN), serial digital (SD), or Bilevel (BL). - Excludes redundant channels).

3a These are all the unique parameters that pass from the source subsystem to one or more Data Handling Subsystem DIMs, either directly, or via an intermediate subsystem. Those in the latter category are listed twice, for the source subsystem (including identification of the intermediate subsystem), and for the intermediate subsystem.

3b These are all the unique parameters that pass from the output of the Data Handling Subsystem, via a 16,384 Hz subcarrier, to the Communications Subsystem.

4 Telemetry timing signals (= number of data timing signal lines supplied directly to the subsystem). Timing signal types include

major frame rate, minor frame rate, bit rate status, analog read envelope, digital read envelope, read clock, and word rate.

5 Attitude timing signals: RIP is supplied to the Bus Scientific Instruments.

6 Clock Reference (= number of clock references supplied directly to the subsystem).

2.3 CAPABILITIES AND LIMITATIONS FOR MULTIPROBE SYSTEM AND FOR THE BUS ONLY

The nominal spacecraft equipment configurations during the nine mission phases, as well as spacecraft system-level capabilities and limitations, are described below.

2.3.1 Nominal Configurations. Spacecraft equipment configurations for each of the nine nominal mission phases described in Paragraph 2.1.4 is shown in Table 2.3.1-1. Of course, alternate equipment of a redundant pair could be used optionally. Some sequences of actions could be chronologically ordered differently and accomplish essentially the same result. However, this nominal mission sequence and corresponding nominal equipment configurations have been adopted from numerous sources as listed in Section 1.5, or, in some instances, for some net advantage. This has been done for the purpose of communicating what can be expected nominally from the equipment, act as a reference for discussing what capabilities exist beyond nominal planned usage, and what limitations exist for the equipment as designed.

2.3.2 System Capabilities and Limitations. The capabilities and limitations of system performance in the following disciplines are discussed below.

(a) Commanding
(b) Telemetering
(c) Power Control

(d) Thermal Control, and
(e) Maneuver Control

Subsystem level details of these capabilities and limitations are discussed in Sections 3 and 4.

2.3.2.1 Commanding. The maximum field-of-view, with respect to the Spacecraft Coordinate System for spacecraft reception of commands, is essentially omnidirectional because one spacecraft receiver is connected to one hemispherically directional antenna while the other receiver is simultaneously connected to another antenna that is hemispherically directional in the remaining hemisphere. The two antennas (Fwd Omni and Aft Omni) can be interchanged in their connections to the receivers by command. Both receivers are powered on at all times and cannot be commanded off. Each receiver is also factory set to receive at a fixed S-band frequency carrier.

Spacecraft-to-earth range is a major factor in determining commanding field-of-view, (calculated command threshold is -136.61 dBm at the receive antenna output port for a BER (bit error rate) of 3.5×10^{-6}) but that factor will not be discussed here. Each antenna essentially provides approximately 0 dBi gain over most of the hemisphere it sees.

The real-time command rate is fixed at 4 bits per second. Real time commanding, therefore, is limited to a maximum rate of 5 commands per minute (48 bits per command). The maximum stored-commanding rate, however, is much faster, at 8 commands per second. The two command memories onboard the spacecraft can each store 128 commands maximum (including time-delay commands). The two command memories provide not only the capability of rapid commanding, but also commanding when the spacecraft is not in view of an earth station, such as during the launch-to-first ground station acquisition period, and during bus entry.

2.3.2.2 Telemetering. The field-of-view with respect to the spacecraft coordinate system for spacecraft transmission of telemetry depends on the transmitter power amplifier/antenna configuration chosen by command. Any one of the fwd omni (provides ≥-2.5 dB over ±90° w.r.t. the spacecraft +Z axis), the aft omni (provides ≥-6 dB over ±90° w.r.t. the spacecraft -Z axis) or the medium gain horn (provides +3.0 to +12.0 dB gain over ±35° w.r.t. the spacecraft -Z axis) can be selected for telemetry use. Not only can low (1 amplifier) or hi (2 amplifiers) power transmitting mode be selected for any one of the three antennas, but both omnis can be fed simultaneously (each by one amplifier) in low power mode, or the medium gain horn and the aft omni can be fed simultaneously in low power mode. Again, spacecraft range is a major factor, but to enhance downlink signal-to-noise ratio, the downlink frequency (S-band) can be locked to the uplink frequency. Alternatively, the downlink transmits a multiple of an onboard crystal oscillator if no uplink is available.

The telemetry data rate is selectable by command (choice of thirteen rates between 8 bps and 4096 bps). The all digitized telemetry (253 available channels) is sampled at various rates based on the format selected (choice of four fixed and one programmable). The programmable format was created for ground test purposes. However, it provides the versatility to select any combination of eight analog, serial digital, or bilevel words maximum (or any single parameter selected eight times), and increase the sampling of the selected parameters to the limit of the selected bit rate.

Telemetry data rates for nominal times in the mission are shown in Figures 3.7.3-1 and 3.7.3-2. These are extracted from Reference 1.5.31.

The above discussion applies for the Bus only. The Bus telemetry bit stream phase shift keys (PSK) a 16,384 Hz subcarrier that, in turn, phase modulates (PM) the downlink S-band carrier. Telemetry from any one of the four attached probes rides on a probe-dedicated subcarrier that also phase modulates the downlink carrier.

2.3.2.3 Power Control. The voltage on all four bus lines (essential, science, switched loads, and RF transmitter buses) is limited to 30.0 volts maximum by the bus limiters. When the voltage drops to nominally 27.8V, the two batteries (nominal 15 amp hours total capacity) come on line in a load-sharing configuration to support the solar panel. When either battery drops to nominally 27.55 volts (or the spacecraft load current exceeds nominally 16.25 amps), an undervoltage trip (or an overload trip, respectively) occurs that may dump all loads except the essential bus loads (e.g., the command receivers).

Maximum batteries depth of discharge (DOD) expected for the nominal mission is 60 percent, although the batteries can take a 70 percent DOD with little risk. Most of the nominal mission is estimated to be flown with the batteries at full charge, necessitating the excess solar panel power to be dumped by the bus limiters nearly continuously into their load resistors. The excess will increase as Venus is approached and the solar intensity factor increases.

The tolerable sun aspect angle range (centered on the optimum 90° sun aspect angle w.r.t. the spacecraft +Z axis) to maintain power balance (i.e., not discharge the batteries) during the mission depends primarily on load level, solar intensity factor, solar cells temperature, and solar cells degradation. The range increases in magnitude as Venus is approached, as the solar intensity factor increases for the cruise conditions specified in Table 2.3.1-1.

2.3.2.4 Thermal Control. Thermal control has been designed to be as passive as possible; augmented by onboard thermostatic control (cannot be overridden by command) of thermal louvers, and shelf heaters (associated with two bus limiters that can be enabled or disabled by command). Onboard thermostatic control also automatically turns off battery charging when battery temperatures exceed 95°±5°F. Battery charging is automatically resumed when battery temperatures decrease by approximately 10°F. Battery charging can be resumed optionally by ground command sooner (before the battery temperature decreases by 10°F). Radial jet heaters and propulsion tank heaters are on continuously and cannot be commanded off.

The only on/off commandable heaters on the Bus (Fwd and Aft axial jet heaters) are expected to be required in an on state for most of the nominal mission (until the fwd axial jet heater is commanded off nominally after Large Probe separation). (All four probes heaters are on/off commandable. The Large Probe heater is expected to be required in an on state until ≈L+87 days.)

The only remaining possible thermal control via the ground in the face of a thermally malfunctioning spacecraft are (1) unscheduled on/off cycling of equipment or (2) unscheduled maneuvering to change the sun aspect angle. Either approach can result in a major impact on some or all of the other performance disciplines discussed in this Section 2.3.2, and must be considered carefully.

The tolerable sun aspect angle range (centered on the optimum 90° sun aspect angle, w.r.t. the spacecraft +Z axis) to maintain thermal balance during the mission depends on the same factors as for power control. However, the range decreases in magnitude as Venus is approached, unlike the trend for power balance maintenance. The thermal design will accommodate a minimum of 0.3 hour of solar illumination directly on the -Z axis spinning end, or a maximum of 4 hours directly on the +Z axis spinning end.

2.3.2.5 <u>Maneuver Control</u>. Various combinations of jets and their firing timing (128 ms pulsed, 512 ms pulsed, or continuous) provide redundant means to generate accurate precession or translation in any direction. (Sections 3.3, 3.4 and 4.1 provide extensive details.) Since the tanks will likely be completely filled at launch (total capacity: 70 lb of hydrazine), and the nominal mission is expected to consume conservatively 37.4 lb, the reserve is substantial.

Maneuver accuracy is not only dependent on resolution of jet firing timing, but also on accuracy of attitude determination. Redundant sun sensors and redundant star sensor channels, each with narrow slit fields-of-view, are used to provide better than 2.5° accuracy in attitude determination. Each star sensor channel signal threshold and bandwidth is selectable in several discrete steps by command. In the normal mode, the attitude telemetry measurement system is designed to trigger in response to only one star at a time - the first one encountered azimuthly within a selected azimuth range (11.25° wide Gate B) in each spin period from occurrence of the sun pulse that meets selected threshold and bandwidth levels.

Spin rate control nominally uses redundant combinations of radial jets that fire continuously. The nominal cruise spin rate of 15 rpm is a compromise between keeping attitude changes due to solar torque small, and fuel consumption low during precessions. The spinup to nominally 48.5 rpm following Large Probe release is to provide a desired centrifugal force for proper small probes' targetting. A despin to 10 rpm prior to Bus entry is done for designed performance of the Bus Science Instruments.

TABLE 2.1.2.3-1

PRELIMINARY ATLAS/CENTAUR D-1AR SEQUENCE OF EVENTS
(REFERENCE: PARAGRAPH 1.5.10)

EVENT	BASIS	APPROX. TIME FROM LIFTOFF (SECONDS)
Liftoff	2-Inch Vertical motion of Atlas/Centaur	0
Roll Program	Liftoff + 2 seconds	2-15
Booster Engines Cutoff (BECO)	5.7 g	139.2
Booster Package Jettison	BECO + 3.1 seconds	142.2
Jettison Insulation Panels	BECO + 45 seconds	184.1
Jettison Nose Fairing	---	224.1
Sustainer Engine Cutoff (SECO) / Vernier Engines Cutoff (VECO)	Prop. Depletion	251.8
Atlas/Centaur Separation	SECO + 1.9 sec	253.8
1st Main Engine Start (MES 1) - Centaur	SECO + 11.5 sec	263.3
1st Main Engine Cutoff (MECO 1) - Centaur	Parking Orbit (Guidance)	582.6
2nd Main Engine Start (MES 2) - Centaur	Guidance	1818.7 (Note 1)
2nd Main Engine Cutoff (MECO 2) - Centaur	Hohmann Apogee	1946.9
Centaur/Spacecraft Separation	MECO 2 + Δt (varies)	2081.9
Reorient Centaur to Retro Vector		2091.9
Start Centaur Blowdown		2401.9
End Centaur Blowdown		2651.9

NOTE 1: Coast time is variable between 12 and 18 minutes, depending on launch day, launch time of day, and launch azimuth.

TABLE 2.1.4.2.9-1

SEQUENCE OF FINAL EVENTS FOR MULTIPROBE SYSTEM

TIME (RELATIVE TO LARGE PROBE ENTRY = E) IN MINUTES:	EVENT	REMARKS
Two hours prior to Bus ≈ (E-31) to (E-21)	Turn ON of Bus Science	• Times shown here and subsequently are representative; actual times depend on launch day and communications look angle. • This is the only listed action by ground command; all others listed are automatic actions.
(E-22)	Large Probe RF ON	• Each of the four probes' RF power is automatically turned ON 22 minutes prior to it's own entry.
(E-17)	Small Probe 1 RF ON	• Order of occurrences for RF on, entry, and impact may change due to launch day and final deployment.
(E-15)	Small Probe 3 RF ON	
(E-13)	Small Probe 2 RF ON	

TABLE 2.1.4.2.9-1. (Continued)

TIME (RELATIVE TO LARGE PROBE ENTRY = E) IN MINUTES:	EVENT	REMARKS
E	Large Probe Entry	•Entry for each probe is defined as the occurrence of that probe reaching 200 km altitude. •Time spread between 1st small probe entry and last small probe entry will be between 3 and 16 minutes.
(E+5)	Small Probe 1 Entry	
(E+7)	Small Probe 3 Entry	
(E+9)	Small Probe 2 Entry	
(E+55)	Large Probe Impact	
(E+62)	Small Probe 1 Impact	•Each impacts ≈ 57 minutes after it's entry.
(E+64)	Small Probe 3 Impact	
(E+66)	Small Probe 2 Impact	
(E+89) to (E+99)	Bus Entry	•85 ±5 minutes after entry of last small probe. •Communications black-out time is expected to occur 1 minute after Bus Entry. The Bus is expected to break up substantially after blackout; assigning an impact time would be inapplicable.

TABLE 2.2.4-1

BUS SUBSYSTEMS & PROBES INTERACTION MATRIX

Affected Subsystem / Source Subsystem	Controls S/S	Propulsion S/S	Data Handling S/S	Command S/S	Communications S/S	Power S/S	Structure & Harness S/S	Large Probe	Small Probes (Each)	Bus Science BIMS	Bus Science BNMS
Controls S/S	2c 8(D); 3(Q).	2c 4(D).	3a 2(SD); 5(AN);8(BL).							2c 2(D).	5 RIP
Propulsion S/S			3a (a) Via Pwr S/S; 3(BL). (b) 17(AN); 2(BL)								
Data Handling S/S	4 5		2c 6(D)4(Q) 3a 2(AN); 4(BL). 4 5	4 1 6 4096 Hz (two lines)	3b 18(SD);78(AN); 68(BL), (Excludes LP & SP C/DU TM). 4 1	4 1	4 1	4 1	4 1	4 6	4 6 6 2048 32,768 Hz.
Command S/S	2b 14(D); 3(Q)		2b 6(D);4(Q) 3a 8(SD); 4(BL).	2b 17(D); 2(Q) 2 17(D); 2(Q).	2b 25(D) (Excludes "Test Only" cmds).	2b 48(D).		2b (a) Via COM 7: 41(D); 2(Q). (b) Via IFD: 2(D); 1(Q). 2c 4(D).	2b Via IFD: 27(D); 1(Q). 2c 3(D).	2b 3(D)	2b 5(D); 1(Q). 2c 5(D).
Communications S/S			3a 16(AN); 14(BL).	2a 188L; 14(Q). (Excludes Program Commands)	2c 23(D).			2 2(D).	2 2(D).		
Power S/S	1 0.294A.	1 0.735A 2c 6(D).	1 0.471A. 3a 17(AN); 24(BL).	1 0.328A.	1 (a) 1.946A (One Pwr Amp ON) (b) 3.546A (Two Pwr Amps ON).	1 0.588A 2c 25(D).	1 Zero A.	1 0.768A (Heater) 2c 7(D).	1 0.280 (Heater) 2c 7(D)	1 0 2c 2 1(D)	1 0 2c 1(D).

TABLE 2.2.4-1 (Continued)

Affected Subsystem / Source Subsystem	Controls S/S	Propulsion S/S	Data Handling S/S	Command S/S	Communications S/S	Power S/S	Structure & Harness S/S	Large Probe	Small Probes (Each)	Bus Science BIMS	Bus Science BNMS
Structure/Harness S/S			3a 14(AN) 4(BL)								
Large Probe			3a (a) Via Pwr S/S:2(BL) (b) Via Str.S/S 1(BL) (c) Via Bus DIMs 1(SD);3(AN); 3(BL). (d) Via Probe C/DU: 9(SD); 24(AN);16(BL)						2c (a) Via Com 7: 41(D), 2(Q). (b) Via IFD: 2(D); 1(Q) (c) Via Pgmr: 53(D).		
Small Probes (Each)			3a (a) Via Pwr S/S: 2(BL) (b) Via STR.S/S: 1(BL). (c) Via Bus DIMs 1(SD);3(AN); 3(BL). (d) Via Probe C/DU: 5(SD); 17(AN);14(BL)						2c (a) Via IFD: 27(D); 1(Q). (b) Via Pgmr: 41(D).		
Bus Science - BIMS			3a (a) Via STR S/S: 1(AN) (b) 2(SD);2(AN); 1(BL)							2c 3(D)	
Bus Science - BNMS			3a (a) Via STR S/S: 1(AN) (b) 2(SD);1(BL)							2c 5(D); 1(Q).	

TABLE 2.3.1-1

MULTIPROBE NOMINAL EQUIPMENT CONFIGURATIONS DURING THE MISSION

MULTIPROBE PHASE → / STATUS (Within, or at end of Phase)	Prelaunch Through Spinup to 15 RPM	Attitude Determination & Star Survey	Jet Calibration; TCM #1; Bus Science Check-out; & Cruise	TCM #2 & Cruise	Probes Checkout; Xmtr Pwr. & Bit Rate Change	Pre-TCM #3 Thru Horn Attitude Operations	Precess to Release Attitude Thru LP Separation	Post-LP Separation Thru SP Separation	Post-SP Separation Thru Bus Impact
Typical Start Time (Relative)	L-5 Min. (switch to S/C Battery Pwr) (L ≡ Launch ≅ 7 thru 24 August 1978).	L+5 Hr.	≅ L+1 Day: Jet Cal. ≅ L+5 Days: TCM No. 1. ≅ L+14 days: Bus Science Checkout	L+20 Days	1 L+60D for early checks 2 E-30D to E-20D. For later checks. 3 Changes depend on range; to Xmtr Hi Pwr ≅ L+80D.	E-40 Days (≅ L+87 days) (E = Large Probe Entry into the Venusian Atmosphere at 200 km Altitude).	E-24 Days 6 Hours	E-24 Days	E-20 Days
Typical Phase Duration & Remarks	Δ5 Hours	1 Δ3 Hrs. for Attitude Deter. & Star survey. (Includes Att. trim (if required)).	Δ19 Days overall (Δ10 Hrs. for jet calibration; Δ4 Hrs for TCM #1); Δ3 Hrs. for bus science check-out.	Δ40 Days overall (Δ4 Hrs for TCM #2).	1 Δ9 Hrs. for early checkout; Δ10 Hrs. total per day for later checkouts. 2 Comm. changes are range dependent.	Δ16 Days (includes pre-TCM #3; precess to horn attitude; large probe checkout; coast timer loading & initiation; & batteries recharge).	Δ6 Hrs.	Δ4 Days	≅ Δ20 Days
Nominal Earth-Multiprobe Distance	0 to 8 x 10^4 km	8 x 10^4 to 3.30 x 10^5 km	3.3 x 10^5 to 5 x 10^6 km	5 x 10^6 to ≈15 x 10^6 km	15 x 10^6 to ≈30 x 10^6 km	30 to ≈40 x 10^6 km	≈40 x 10^6 km	≈40 x 10^6 km to ≈50 x 10^6 km	50 x 10^6 to >70 x 10^6 km
Nominal Sun L.O.S. (W.R.T. Spin Axis)	90°	90°	89° to 90°	88° to 90°	88° to 90°	1 88° to 90° for TCM #3. 2 51° to 56° for Horn Att.	22° to 44° for L. P. release Attitude.	15° to 32° for S. P. release Attitude.	54° to 60° for Entry Attitude.

C-2

TABLE 2.3.1-1 (Continued)

MULTIPROBE PHASE → STATUS (Within, or at end of Phase)	Prelaunch Through Spinup to 15 RPM	Attitude Determination & Star Survey	Jet Calibration; TCM #1; Bus Science Check-out; & Cruise	TCM #2 & Cruise	Probes Checkout; Xmtr Pwr. & Bit Rate Change	Pre-TCM #3 Thru Horn Attitude Operations	Precess to Release Attitude Thru LP Separation	Post-LP Separation Thru SP Separation	Post-SP Separation Thru Bus Impact
Nominal Earth L.O.S. (W.R.T. Spin Axis)	75^o to 83^o	75^o to 83^o	71^o to 78^o	71^o to 82^o	71^o to 88^o	1 82^o to 88^o for TCM#3. 2 ≈150^o for Horn Att.	130^o to 154^o for L.P. Release Attitude.	145^o to 158^o for S.P. Release Attitude.	165^o to 179^o from E-8D to E≡0.
Earth Station(s) in use:	26 M.	26 M:	26 M. for jet calibration and for bus science checkout; 64 M for TCM#1.	64 M for TCM #2; 26 M, otherwise.	64 M for Probes checkout; 26 M, otherwise.	64 M.	64 M for Probe release 26 M, otherwise.	64 M for Probes release; 26 M otherwise.	64 M for bus entry; 26 M otherwise.
Communications S/S Mod. Index (Radians)	1.18			1 1.18 (Lo pwr; 64 M). 2 0.65 (Lo pwr; 26 M).	Probe/Bus; 1.025/0.65.	For last Large Probe checkout; Probe/Bus: 1.025/0.65.	1 Bus only; 0.65 (omni); 1.18 (Horn). 2 For Large Probe Coast Timer set: Probe/Bus: 1.025/0.65.	1 For SP Coast Timer set; Probe/Bus: 1.17/0.65. 2 For Bus only; 0.65 (omni); 1.18 (horn).	Bus Lo/Hi Pwr (Horn): 0.65/1.18.
Transmit Antenna	•Forward Omni unit separation +2 minutes; then Aft Omni until shortly after initial ground acq. (L+4 Hr); then return to forward Omni.					After TCM#3: 1 Partial precession, then switch to Aft omni. 2 Remaining precession, then sw. to medium gain horn.	Switch to Aft omni just prior to precession to release attitude.	Precess to SP targeting, then switch to horn.	1 Precess back to SP Target Att. (Horn Att). 2 Sw. to aft omni just prior to precess to Bus target. att. 3 After Bus targeting, repeat 1, above.

TABLE 2.3.1-1 (Continued)

MULTIPROBE PHASE → STATUS (Within, or at end of Phase)	Prelaunch Through Spinup to 15 RPM	Attitude Determination & Star Survey	Jet Calibration; TCM #1; Bus Science Check-out; & Cruise	TCM #2 & Cruise	Probes Checkout; Xmtr Pwr. & Bit Rate Change	Pre-TCM #3 Thru Horn Attitude Operations	Precess to Release Attitude Thru LP Separation	Post-LP Separation Thru SP Separation	Post-SP Separation Thru Bus Impact
Receive Antenna	RCVR 1 to Fwd Omni; RCVR 2 to Aft Omni.								
Exciter	1 ON; 2 OFF. Coherent Mode enabled.								
Power Amplifier(s)	1 ON; 2, 3, & 4 OFF (≅ Low Power).	Low Power	Low Power; except Hi Pwr (1 & 3 ON) for Bus Science Checkout.		High Pwr for Probes checkouts, and beginning ≅ L+80 days.	(1) Hi pwr until E-28 days; then Low Power. (2) Hi pwr at LPR-12 Hr.	Hi power starting at (LPR-5H40M); Low pwr, otherwise. (LPR = L. P. Release Time).	Precess to SP targeting, then Low pwr.	(1) Prior to E-18D: Low for horn; Hi for Aft omni. (2) After E-18D: Hi for Aft omni & Horn.
States of Transfer Switches	(1) Lo pwr to Fwd omni. (2) Exciter 1 selected. (3) Amp 1 to Low. (4) Fwd omni selected (5) RCVR normal selected.		For Hi Xmtr pwr (bus science checkout); (1) Hi pwr to fwd omni. (2) Amp 1 to Hi (3) Amp 3 to Hi			After TCM#3: (1) Partial precession, then sw. to Aft Omni. (2) Remaining precession then: a) Hi pwr to fwd/horn. b) Horn Selected.	At LPR-5H40M: a) Amp 1 to Hi. b) Hi pwr to horn.	(1) Immediatly after precession to S. P. targeting attitude: a)Amp 1 to Lo. b)Horn select.	(1) At E-18D (Bus targeting) Aft Omni; Hi (2) At E-17D 20H: a) Amp 1 to Hi. b) Hi pwr to Fwd/Horn.

(Continued)

TABLE 2.3.1-1 (Continued)

MULTIPROBE PHASE → / STATUS (Within, or at end of Phase)	Prelaunch Through Spinup to 15 RPM	Attitude Determination & Star Survey	Jet Calibration; TCM #1; Bus Science Check-out & Cruise	TCM #2 & Cruise	Probes Checkout; Xmtr Pwr. & Bit Rate Change	Pre-TCM #3 Thru Horn Attitude Operations	Precess to Release Attitude Thru LP Separation	Post-LP Separation Thru SP Separation	Post-SP Separation Thru Bus Impact
States of Transfer Switches (Continued)						3 At E-28D& LPR-9H: a) Amp 1 to Low. b) Low pwr to Fwd/Horn 4 At LPR-12H: Same as 2, above.			
Command S/S : Command Processor	1 1 & 2 ON. 2 Sep. sw. armed. 3 Spin-up sequence is stored in both, prelaunch; executed ≅ L+45M; then SCL 1 & 2 go to standby.	SCL 1 & 2 cleared.							Entry Sequence is stored in SCL 1 & 2, then executed.
Command Output Modules	1 Through 7 ON.								
Pyrotechnics	PCU 1 & PCU 2 Disarmed.						1 LP IFD Armed & Fired. 2 LP SEP Armed & Fired.	1 All SP IFD Armed & Fired. 2 All SP SEP Armed & Fired.	BNMS CAL GAS Pyro & B/O Hat Armed and Fired sequentially.

TABLE 2.3.1-1 (Continued)

MULTIPROBE PHASE ⟶ / STATUS (Within, or at end of Phase)	Prelaunch Thru Spinup to 15 RPM	Attitude Determination & Star Survey	Jet Calibration; TCM#1; Bus Science Checkout; & Cruise	TCM#2 & Cruise	Probes Checkout; Xmtr Pwr & Bit Rate Change	Pre-TCM#3 Thru Horn Attitude Operations	Precess to Release Attitude Thru LP Separation	Post-LP Separation Thru SP Separation	Post-SP Separation Thru Bus Impact
Control S/S : Star Sensors		Ψ* & Ψ_2 * both ON; Appropriate gain & band-width selected.							
Sun Sensors	Mid-range selected								
Attitude Data Processor: ON/OFF State	Both ON for auto. spinup; then only ADP 1 ON.								
Attitude Data Processor: Measurement Select	Nominal config. (for spin rate & Sun L.O.S. measurement); A: Ψ to Ψ B: Ψ to Ψ_2	1 Nominal config, then (for Star L.O.S. measurements): A: Ψ to Ψ_2 B: Ψ to gated Star B.							
Attitude Data Processor: Jet Control	1 R1 & R3 fired continuously for Spinup to 15 RPM. 2 Spin rate detector inhibited.	For Attitude Trim: 1 (R1 & R4) or (R2 & R3) or A5 or A5 & A6 or A6 selected for pulsed precession. 2 Spin rate detector enabled.	For Jet Calib: 1 Radials (pulsed), then; 2 Axials (pulsed). For TCM#1 (radials pulsed)) (followed by axial(s) in continuous mode.)	Same as TCM#1	No jets selected	Same as for TCM#1.	Attitude determination, attitude trim, spin rate trim, and precessions. Same as in previous applicable phases.	1 Same as for preceding phase; 2 Additionally, Trim Jet(s) combinations are the same choices as for TCM #1.	Same as for preceding phase.

TABLE 2.3.1-1 (Continued)

MULTIPROBE PHASE → STATUS (Within, or at end of Phase)		Prelaunch Thru Spinup to 15 RPM	Attitude Determination & Star Survey	Jet Calibration; TCM#1; Bus Science Checkout; & Cruise	TCM#2 & Cruise	Probes Checkout; Xmtr Pwr & Bit Rate Change	Pre-TCM#3 Thru Horn Attitude Operations	Precess to Release Attitude Thru LP Separation	Post-LP Separation Thru SP Separation	Post-SP Separation Thru Bus Impact
Attitude Data Processor (cont'd)	ADP Mode Select	① PLL loss of lock inhibited until after spinup to 15 rpm. ② Sun, SRR, normal all selected ③ Sun gate disabled for initial Sun acq.; then enabled. ④ SRR advance inhibited. ⑤ Star Gate B Channel 1 selected	Additionally: PLL Spin range of 8.0 - 17.7 rpm selected.					Attitude determination, attitude trim, spin rate trim, and precessions. Same as in previous applicable phases.	① Same as for preceding phase; ② Additionally, ΔV Trim Jet(s) combinations are the same choices as for TCM#1.	Same as for preceding phase.
ADP Configure Commands 3 thru 11.		JCE buffer output enabled for spinup to 15 rpm, then disabled.	For Attitude Trim: ① JCE enabled ② PLL spin period magnitude loaded. ③ JCE count down magnitude loaded ④ ACS angle magnitude loaded. ⑤ Jets fired; then JCE disabled.	① For pulsed firing: Same as for Attitude Trim. ② For continuous firing: Same as for spinup to 15 rpm.	Same as for TCM#1	JCE disabled.	Same as for TCM#1			

TABLE 2.3.1-1 (Continued)

MULTIPROBE PHASE → STATUS (Within, or at end of Phase)	Prelaunch Thru Spinup to 15 RPM	Attitude Determination & Star Survey; &	Jet Calibration; TCM#1; Bus Science Checkout & Cruise	TCM#2 & Cruise	Probes Checkout; Xmtr Pwr & Bit Rate Change	Pre-TCM#3 Thru Horn Attitude Operations	Precess to Release Attitude Thru LP Separation	Post-LP Separation Thru SP Separation	Post-SP Separation Thru Bus Impact
Spin Rate	Automatic spinup to 15 rpm.							Spinup to nominally 48.5 rpm	Despin to 9.45 rpm.
Data Handling S/S : Nominal Bit Rate (BPS)	256 until 26 M. station acquisition of downlink after initial spinup; then 2048.	2048 to 16, depending on distance.	2048 for 64M dish; 1024 to 341.33 for 26M -either with xmtr lo pwr.	1024 for 64M dish; 341.33 for 26M - with xmtr lo pwr;	LP/Bus: 256 /8 SP/Bus: 64/8	① For LP/Bus Operation: LP/Bus: 256/8 ② For Bus only: Lo/Hi pwr: 128/256 on Horn; 16 on Omni	Bus: Lo/Hi power: 128/256 on Horn; 16 on Omni.	① For SP/Bus Operation: SP/Bus 64/8 ② For Bus only: a) Lo/Hi pwr: 64/256; at SP release: 128 (All on Horn)	① Via Aft omni: Lo/Hi pwr: 16 ② Via Horn: Lo/Hi pwr: 64 to 256/64 to 128.
Data Format	Bus Engineering	ACS for att. det. & stars survey; Bus eng., otherwise.	① For Jet Calib. & TCM #1: ACS ② "Entry" for Bus science checkout. ③ Bus eng. otherwise.	① For TCM#2: ACS. ② Bus Eng. otherwise	Bus = Bus Eng; LP= Desc Blackout; SP=Up. Desc/Low Desc/Bkout.	① For att. Det. & Maneuvers: ACS; Bus eng., otherwise.	Bus = ACS for Att. Det. & Maneuvers; bus eng., otherwise. LP=Desc./Blackout.	For Maneuvers: ACS; Bus Eng. otherwise.	① For maneuvers: ACS. ② Entry format for entry ③ Bus Eng., otherwise.
PCM Encoder	1 ON; 2 OFF								
TM Processor	1 ON; 2 OFF. Subcarrier & data on: convolutional encoder OFF.								
DATA INPUT MODULES	All 8 ON.								

TABLE 2.3.1-1 (Continued)

MULTIPROBE PHASE → STATUS (Within, or at end of Phase)	Prelaunch Thru Spinup to 15 RPM	Attitude Determination & Star Survey	Jet Calibration; TCM#1; Bus Science Checkout; & Cruise	TCM#2 & Cruise	Probes Checkout; Xmtr Pwr & Bit Rate Change	Pre-TCM#3 Thru Horn Attitude Operations	Precess to Release Attitude Thru LP-Separation	Post-LP Separation Thru SP Separation	Post-SP Separation Thru Bus Impact
Power S/S : Typical Solar Panel Power:		228 W.			234 W.			80 W. for S. P. release attitude.	
Typical Max. Batteries D.O.D.					4% for probes' ckout.			48% for S. P. release att. & 50% duty cycle of xmtr Hi. Pwr.	
Batteries Charge/Discharge State (Typical)	1 Hi charge rate selected. 2 Primary discharge regulator selected.						Batteries to Hi chg. rate for several hrs; then return to low chg. rate.	Batteries to Hi charge rate	Batteries returned to Lo charge rate, if not already in that state.
Protective Circuits	1 All 5 limiters enabled. 2 UV/OL protection OFF at launch, then ON at Acquisition. 3 Precharge OFF.								
Science Bus	All loads OFF.		Bus Science Loads; then All Loads OFF		Checkout pwr ON; then left ON to support S.P. Stable Oscillator operation			Checkout power OFF after last checkout of SP	

TABLE 2.3.1-1 (Continued)

MULTIPROBE PHASE → STATUS (Within, or at end of Phase)	Prelaunch Thru Spinup to 15 RPM	Attitude Determination & Star Survey	Jet Calibration; TCM#1; Bus Science Checkout; & Cruise	TCM#2 & Cruise	Probes Checkout; Xmtr Pwr & Bit Rate Change	Pre-TCM#3 Thru Horn Attitude Operations	Precess to Release Attitude Thru LP Separation	Post-LP Separation Thru SP Separation	Post-SP Separation Thru Bus Impact
Propulsion S/S: Latch Valves	1 & 2 opened for auto. spinup to 15 rpm; closed otherwise.	For attitude trim: 1 & 2 both opened; then closed.	1 & 2 both opened; then closed.	1 & 2 both opened; then closed.		1 & 2 both opened; then closed.	1 & 2 both opened, then closed after all firings approx. adjacent in time	1 & 2 both opened before, then closed after all firings approx. adjacent in time	1 & 2 both opened before, then closed after all firings approx. adjacent in time.
Heaters	Primary tanks, fwd. & aft Jet Heaters all ON.					Fwd Axial Jet Htr OFF at ≈E-28D.			
Nominal Fuel Usage (lbs)	Δ1.64 for initial spinup to 15 rpm.		1 Δ9.93 for axial ΔV. 2 Δ11.27 for radial ΔV.	1 Δ0.57 for axial ΔV. 2 Δ0.63 for radial ΔV.		1 Δ0.08 for axial ΔV. 2 Δ0.09 for radial ΔV. 3 Δ1.30 for precess to Horn attitude	1 Δ0.64 for precess to L.P. Sep. Att.	1 Δ0.64 for precess back to Horn Att. 2 Δ2.93 for spin-up to 48.5 rpm. 3 Δ2.34 for radial ΔV. 4 Δ2.17 for precess to S.P. Sep. Att.	1 Δ0.61 for precess to Horn Att. 2 Δ0.63 for precess to Bus Target Att. 3 Δ3.90 for axial ΔV (Bus Retarget). 4 Δ0.63 for precess back to Horn Att. 5 Δ0.37 for precess to Bus Entry Att. 6 Δ1.66 for despin to 10 ±0.1 rpm.
Bus Science Instruments	BIMS & BNMS both OFF.		On sequentially; BNMS calibrate & pump ON; then all OFF.						E-4011 to E-1011: Calibrate Gas released; B/O Hat Deployed, then BIMS & BNMS Pwr. ON.

Revision

TABLE 2.3.1-1 (Continued)

MULTIPROBE PHASE → STATUS (Within, or at end of Phase)	Prelaunch Thru Spinup to 15 RPM	Attitude Determination & Star Survey	Jet Calibration; TCM#1; Bus Science Checkout; & Cruise	TCM#2 & Cruise	Probes Checkout Xmtr Pwr & Bit Rate Change	Pre-TCM#3 Thru Horn Attitude Operations	Precess to Release Attitude Thru LP Separation	Post-LP Separation Thru SP Separation	Post-SP Separation Thru Bus Impact
Large Probe :	Heater ON after spinup to 15 rpm.				(1) Checkout pwr ON; descent format science ON & com. as req'd.; then checkout pwr OFF. (2) Htr OFF @ ≈E-40D	LP htr. OFF	(1) L. P. Internal pwr ON; C/DU assessment; Coast timer set & started; then C/DU OFF. (2) L. P. released.		
Small Probe :					(1) S.P. checkout power ON and Stable Oscillator in each Small Probe turned ON at L+82 days (E-45 days); and both left ON continuously until S.P. separation (2) Stable Oscillator turned ON. (3) Sequential checkout for each S.P., as for L.P. above.			For each S. P. Stable Oscillator OFF, internal power ON; C/DU assessment; Coast Timer set & started; then C/DU OFF; all S. P. released simultaneously.	

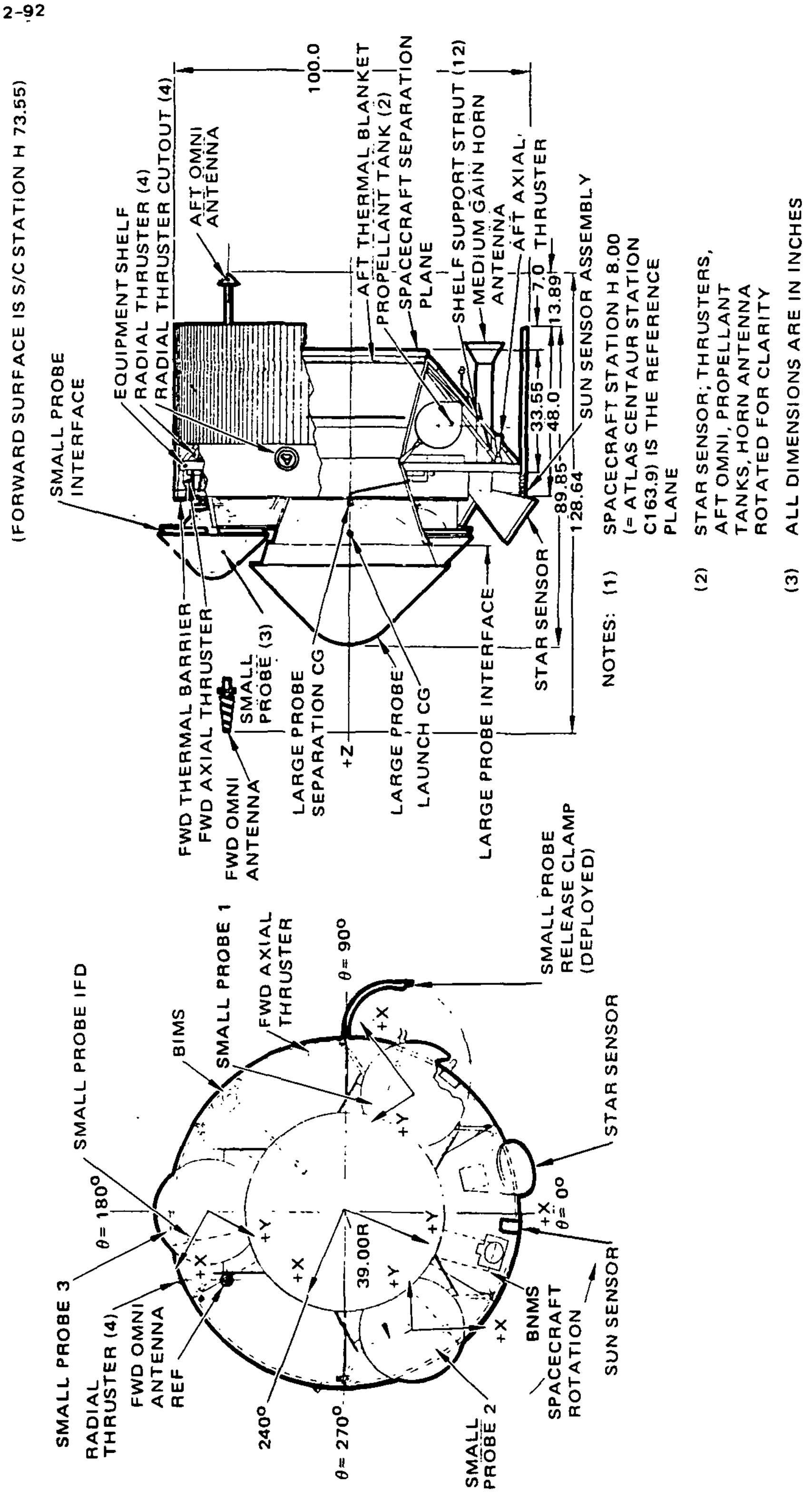


FIGURE 2.1.1-1 MULTIPROBE CONFIGURATION AND COORDINATE SYSTEM

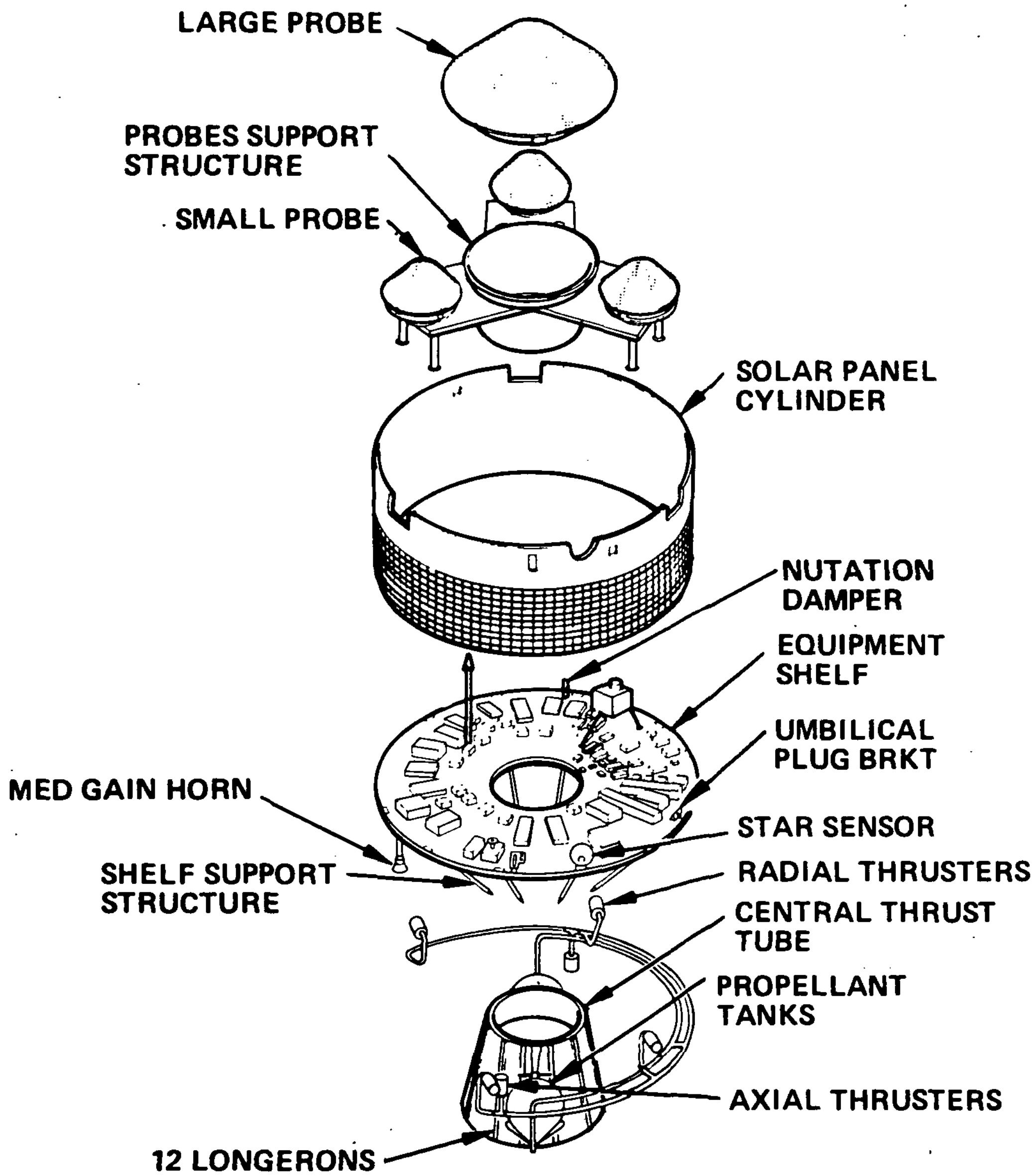


Figure 2.1.1-2. Multiprobe Exploded View

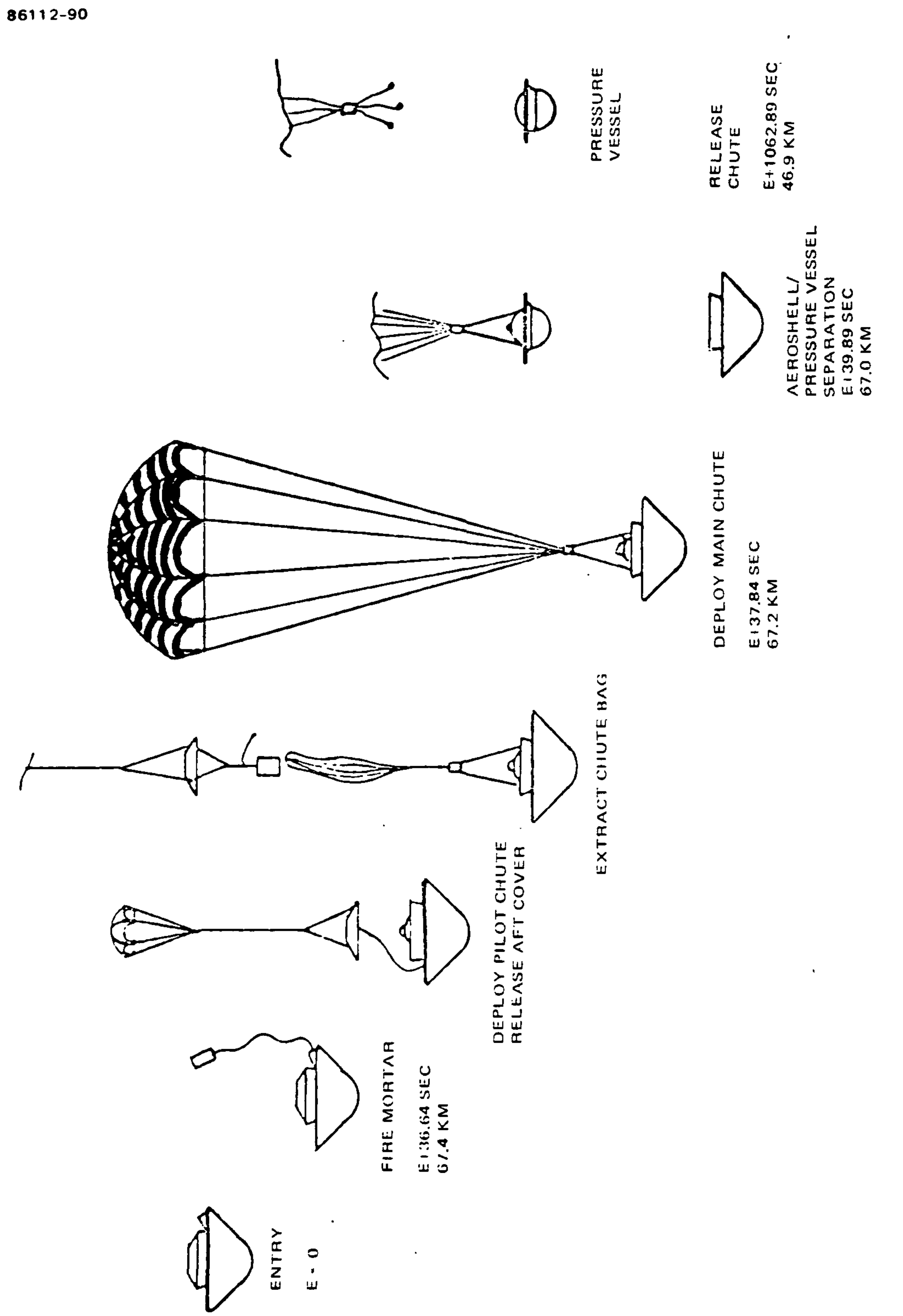


FIGURE 2.1.1.2-1 LARGE PROBE CONFIGURATION HISTORY

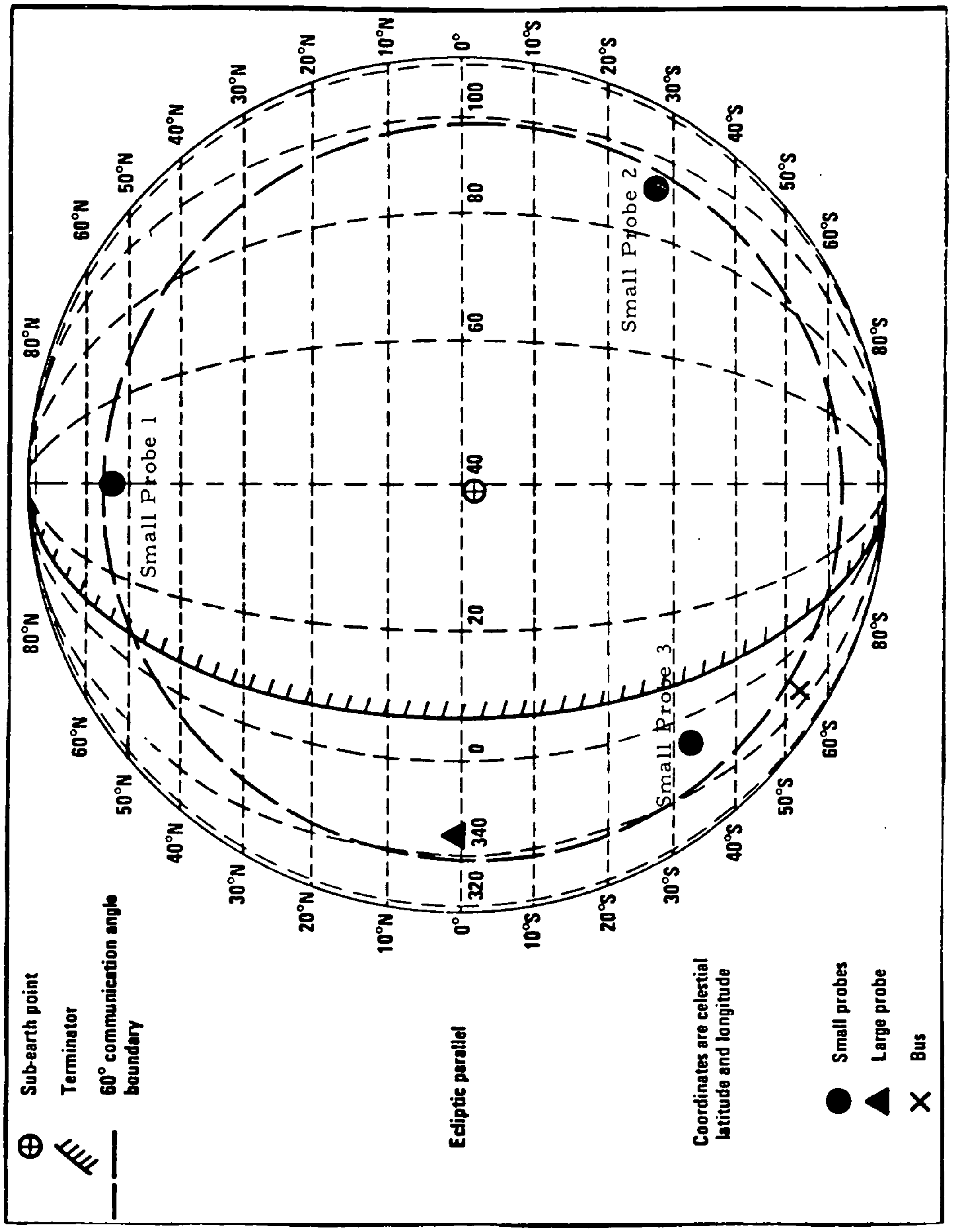


Figure 2.1.1.3-1. Probes and Bus Entry Location as Viewed from Earth

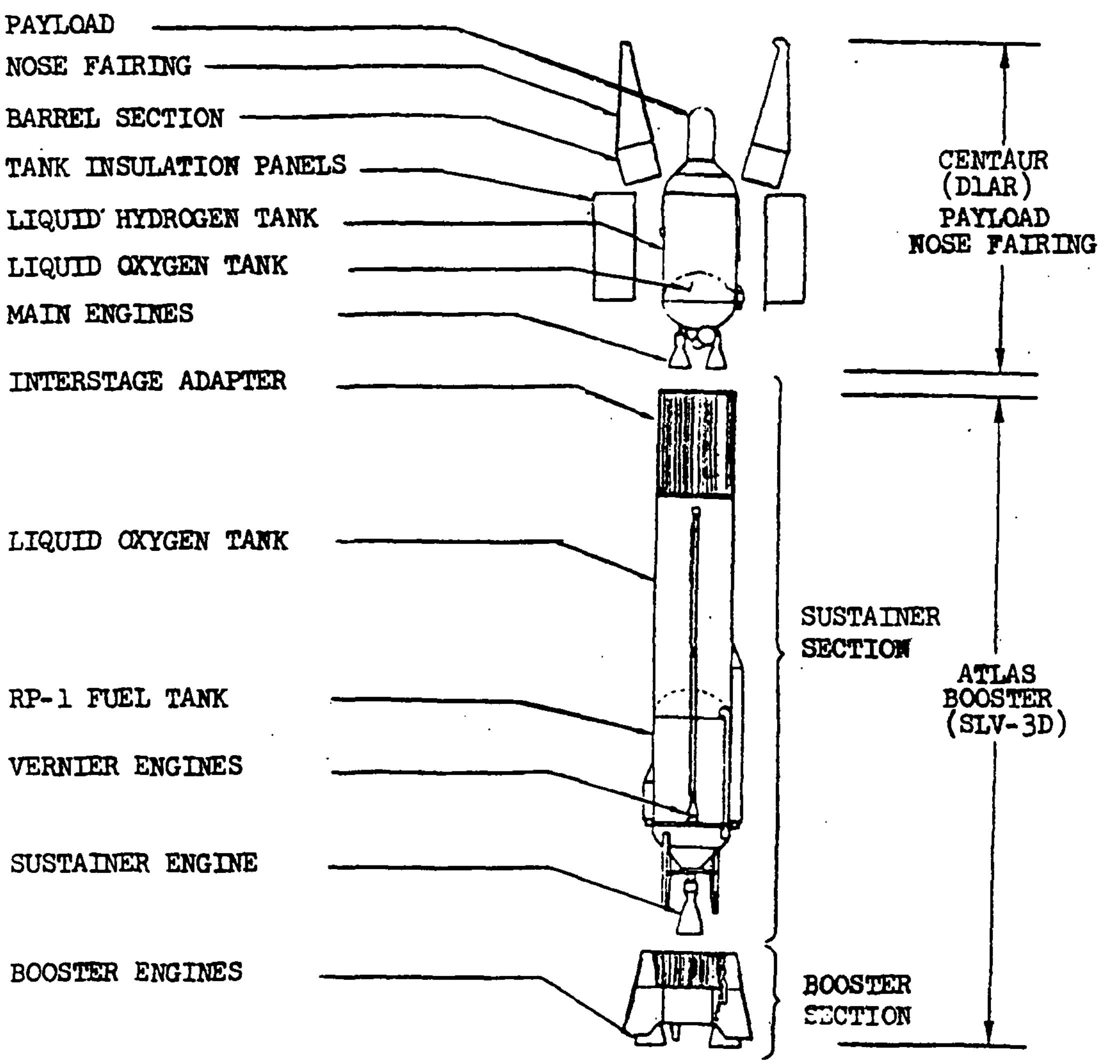


Figure 2.1.2-1. The Atlas Centaur D-1AR Vehicle

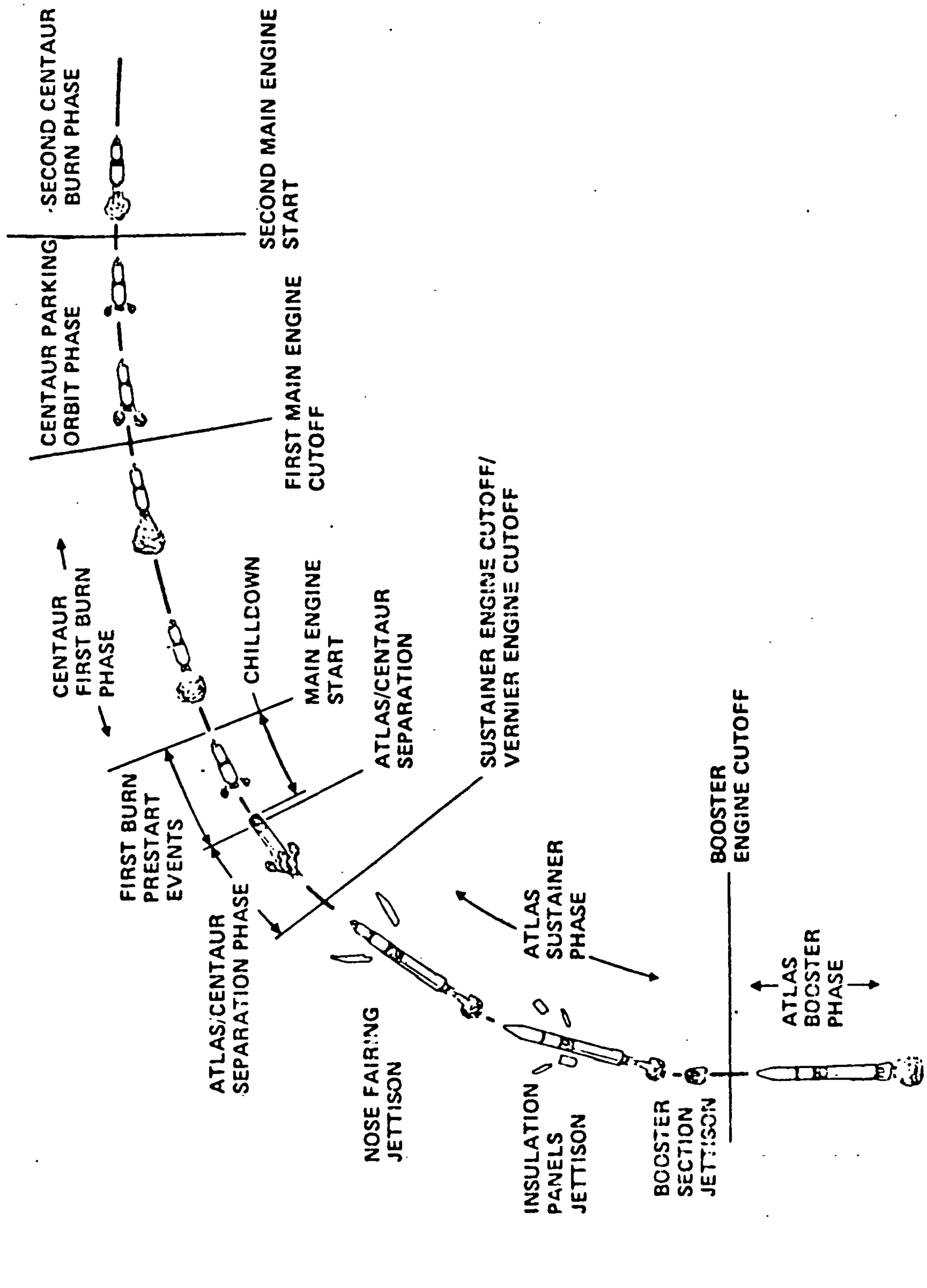


Figure 2.1.2.3-1A. Atlas/Centaur Flight Profile

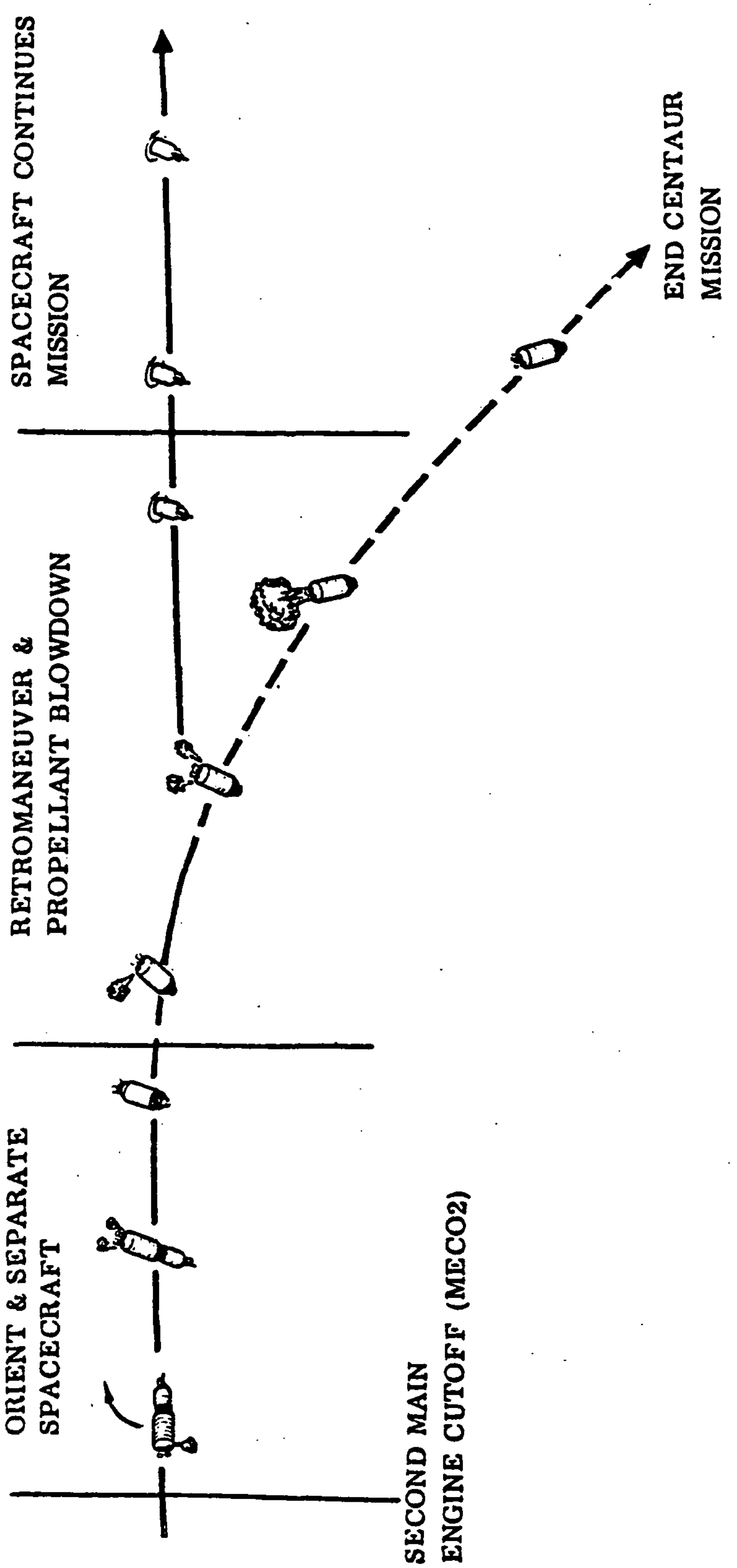


Figure 2.1.2.3-1B. Atlas/Centaur Flight Profile (Continued)

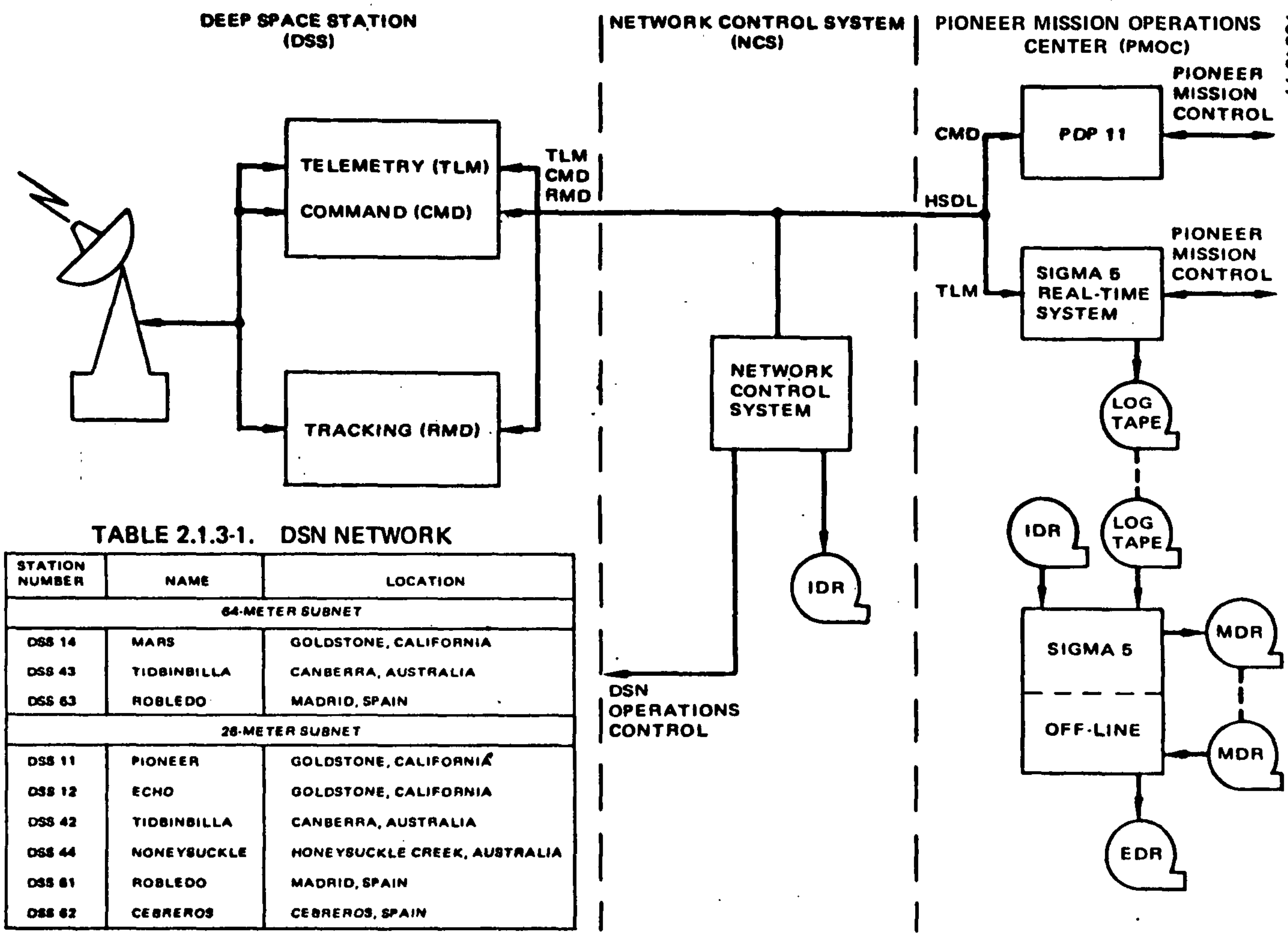


TABLE 2.1.3-1. DSN NETWORK

STATION NUMBER	NAME	LOCATION
64-METER SUBNET		
DSS 14	MARS	GOLDSTONE, CALIFORNIA
DSS 43	TIDBINBILLA	CANBERRA, AUSTRALIA
DSS 63	ROBLEDO	MADRID, SPAIN
26-METER SUBNET		
DSS 11	PIONEER	GOLDSTONE, CALIFORNIA
DSS 12	ECHO	GOLDSTONE, CALIFORNIA
DSS 42	TIDBINBILLA	CANBERRA, AUSTRALIA
DSS 44	HONEYSUCKLE	HONEYSUCKLE CREEK, AUSTRALIA
DSS 61	ROBLEDO	MADRID, SPAIN
DSS 62	CEBREROS	CEBREROS, SPAIN

FIGURE 2.1.3-1. PGDS NETWORK CONFIGURATION (MARK III-74, DIRECT MODE)

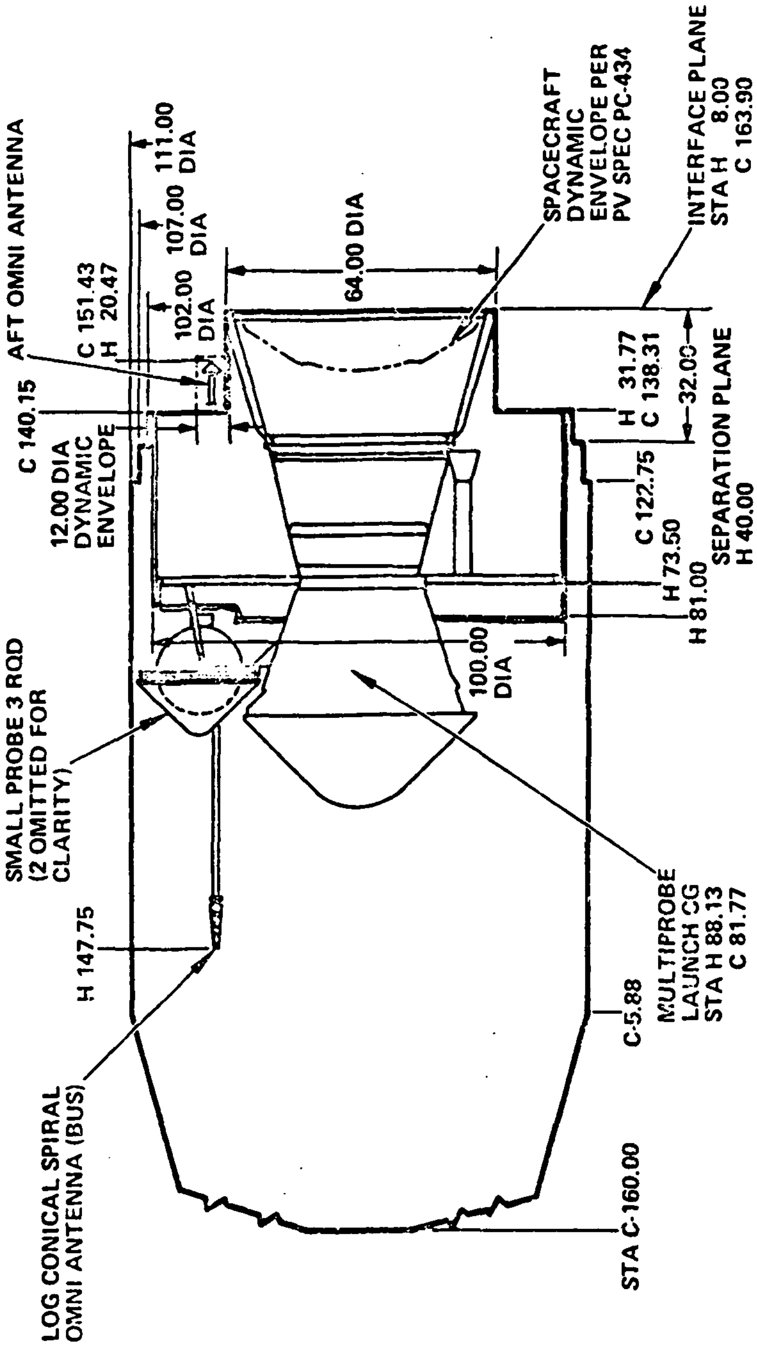


Figure 2.2.1-1. Multiprobe Launch Configuration Envelope

Revision

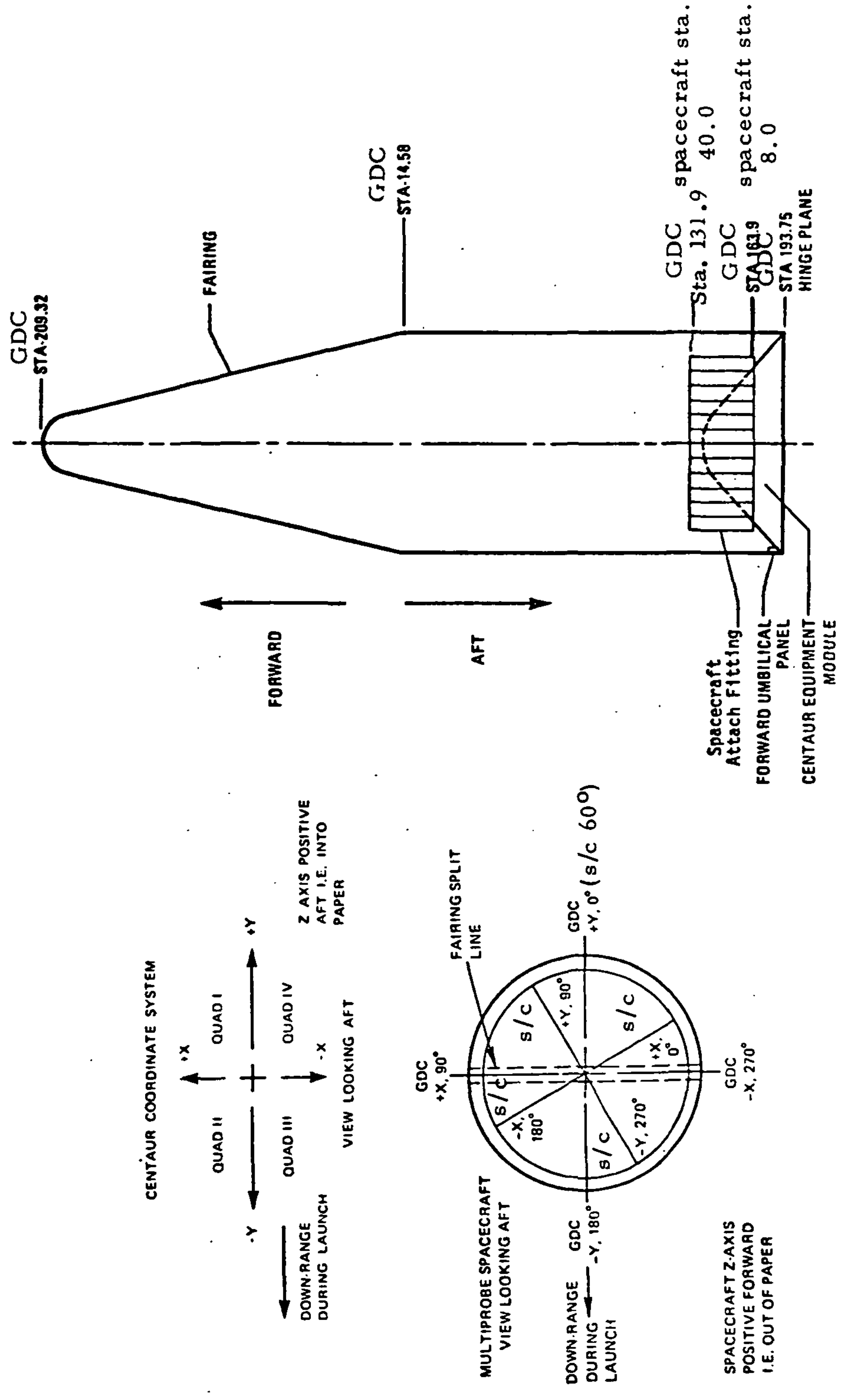


Figure 2.2.1.1-1. Relationship Between Multiprobe Spacecraft and Launch Vehicle Coordinate Systems

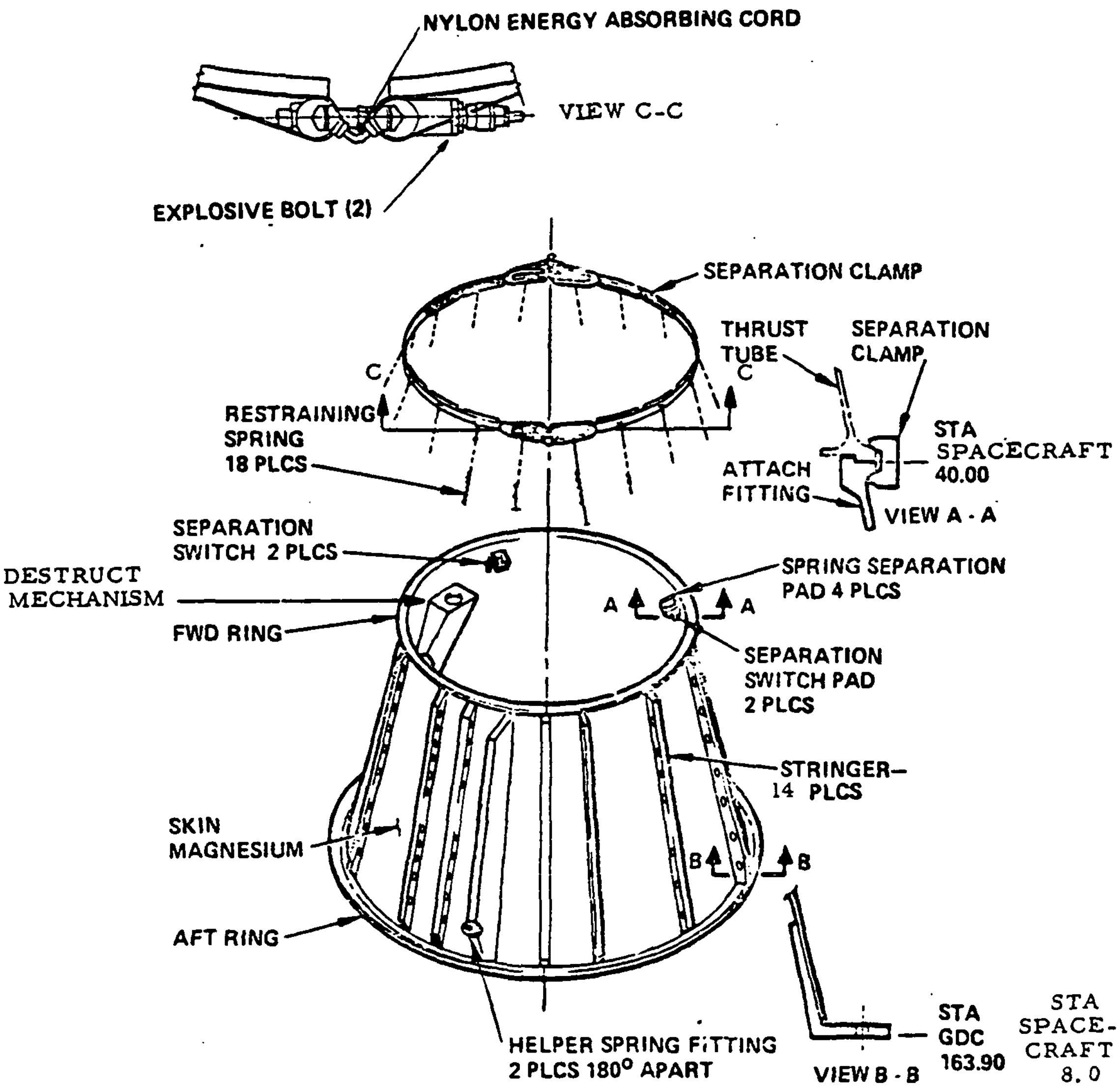


Figure 2.2.1.1-2. Spacecraft Attach Fitting

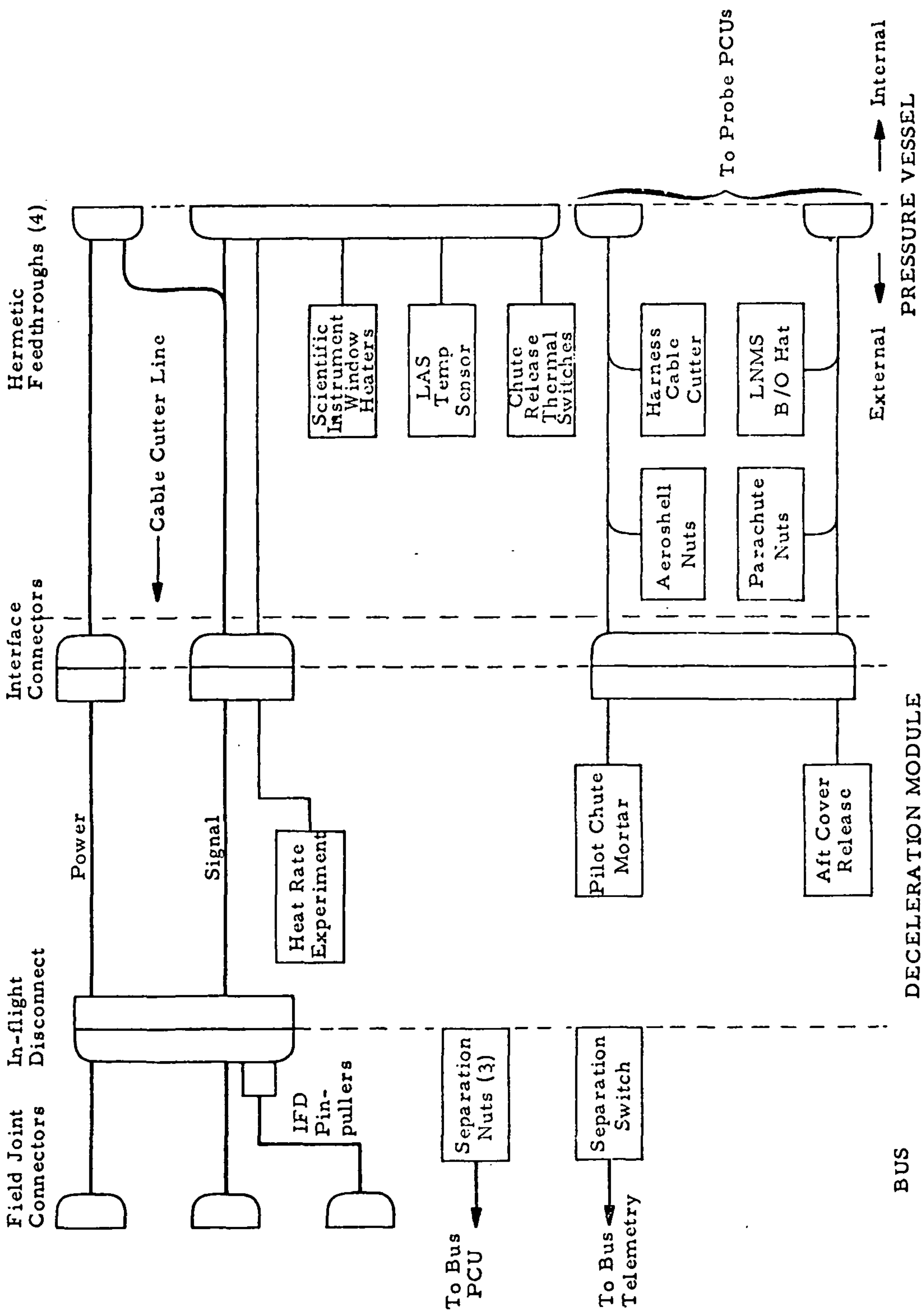


Figure 2.2.2.3-1. Bus/Large Probe Electrical Interface Diagram

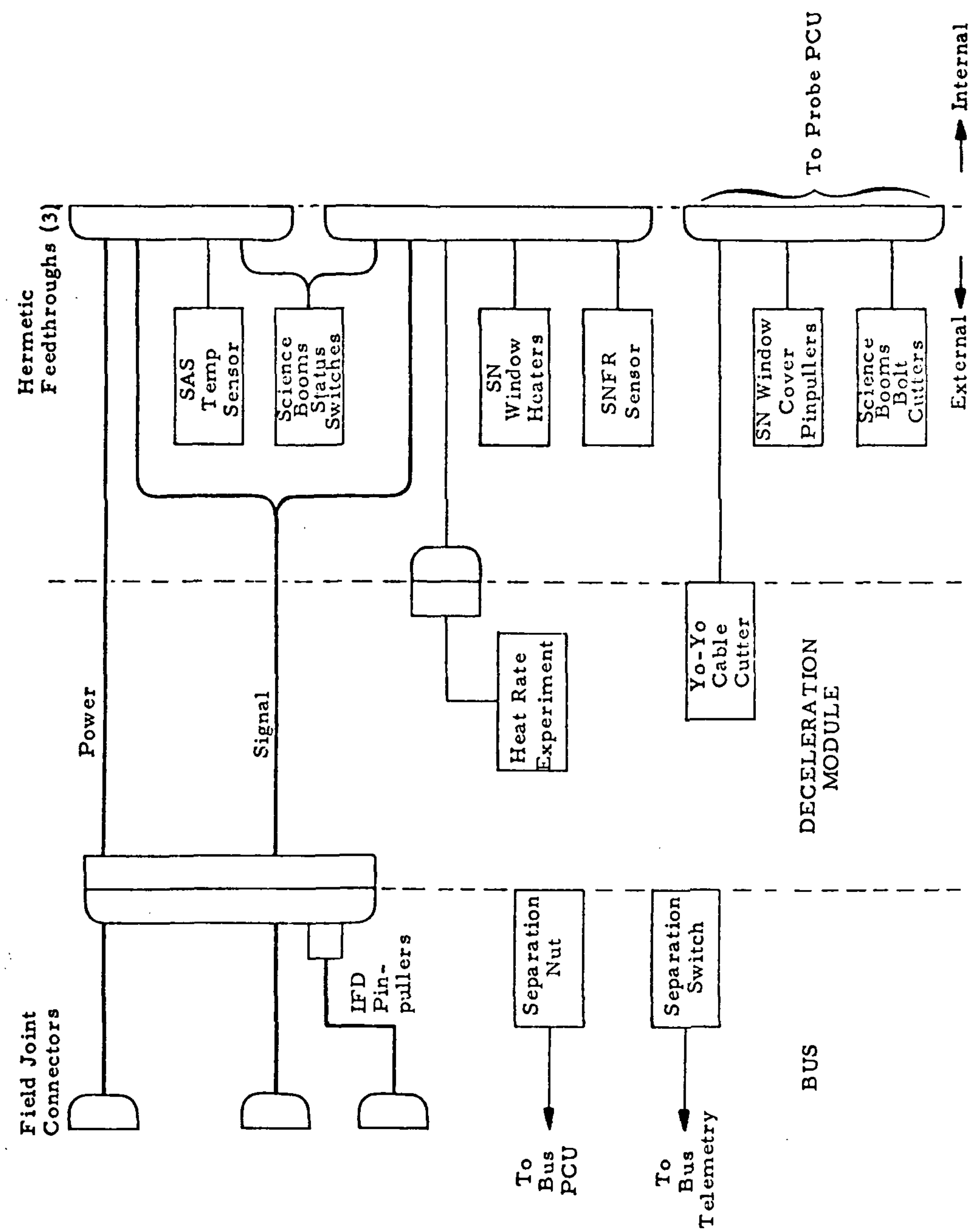


Figure 2.2.2.3-2. Bus/Small Probe Electrical Interface Diagram

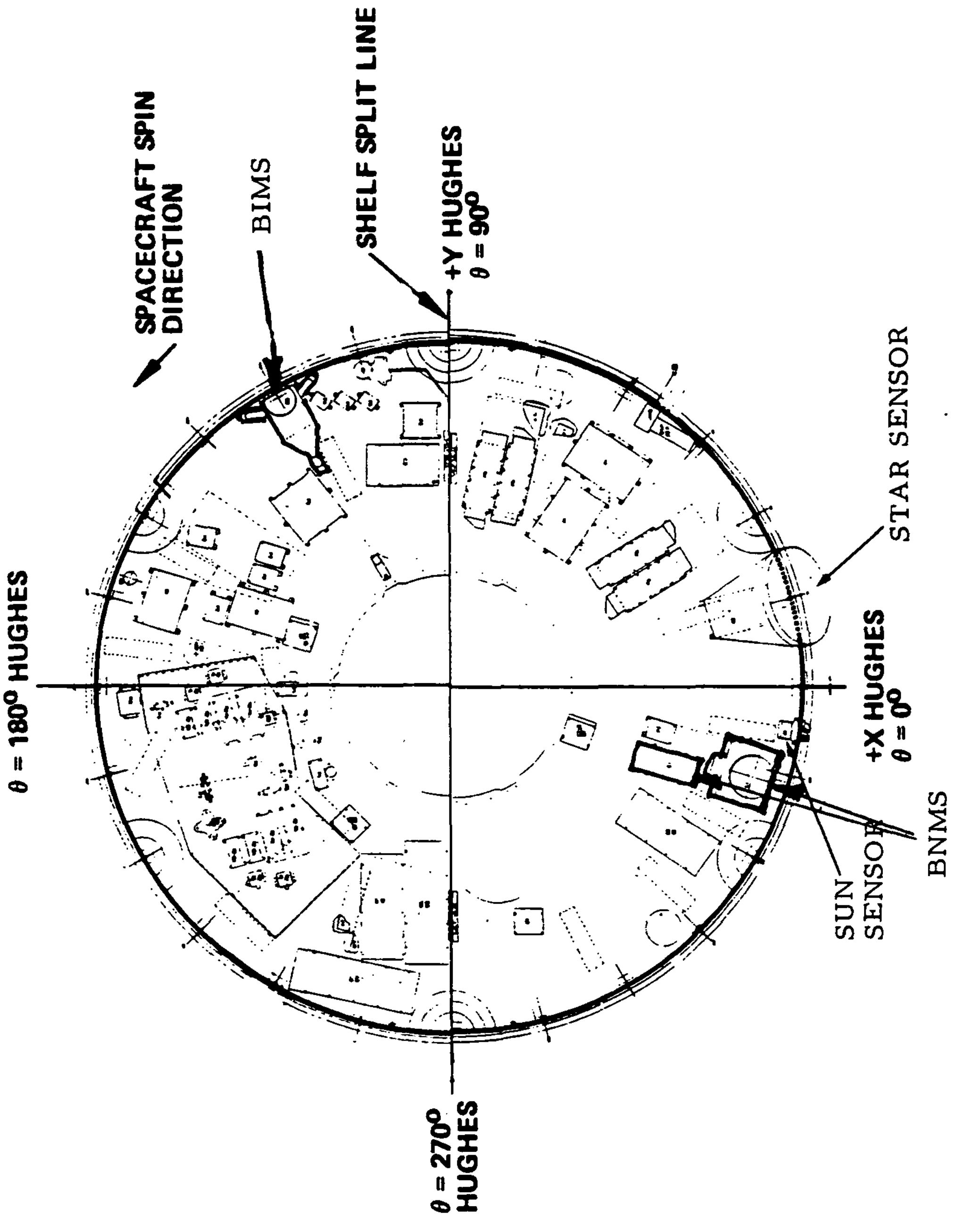


Figure 2.2.3.1.1-1. Bus Instruments Layout on Equipment Shelf

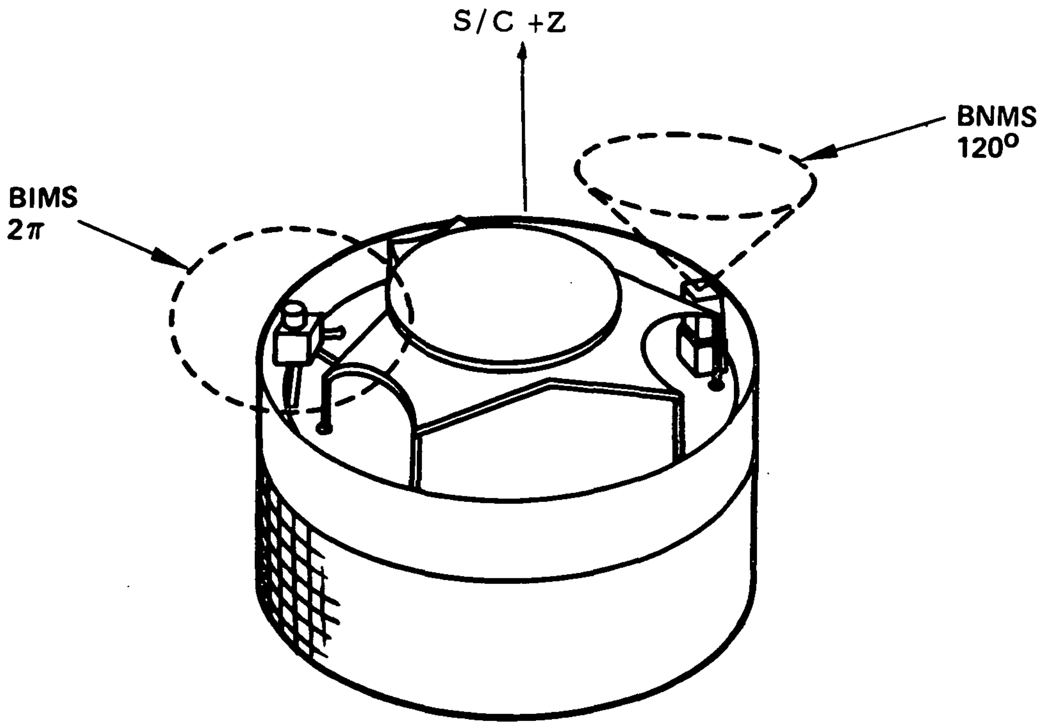


Figure 2.2.3.1.1-2. Bus Instruments Fields-of-View

This Page Intentionally Left Blank.

3.0 BUS SUBSYSTEMS DESCRIPTIONS

Functional and Operational descriptions of the Bus Spacecraft at the subsystem and unit level are presented ahead.

3.1 MECHANICAL DESIGN

3.1.1 Spacecraft Configuration

3.1.1.1 General Description. The mechanical design of the Multiprobe was primarily determined by the use of spin stabilization, the requirement to separate the Large and Small Probe for Venus entry, the use of solar cells for power generation and the design goal of commonality with the Orbiter spacecraft. Spin stabilization and solar cell generated power led to a cylindrical solar panel with its axis of symmetry coincident with the spacecraft spin axis. The diameter of the cylindrical solar panel was made as large as possible within the limitations of the launch vehicle shroud in order to maximize the roll to pitch moment of inertia ratio and thus improve spacecraft stability. The solar panel is 100 inches in diameter.

The equipment shelf is circular in order to fit inside the solar panel. Spacecraft units and scientific instruments are mounted near the periphery of the shelf for improved view angles and to increase the roll to pitch moment of inertia ratio. Since the majority of the shelf mounted units and the solar panel are mounted near the edge of the shelf, struts are provided to support this outer edge. The struts terminate at the spacecraft thrust tube.

A probe support structure is mounted to the forward end of the spacecraft to support the probes during launch and to position them for release and separation. The Large Probe is released along the velocity vector (spacecraft axis of symmetry) while the three Small Probes are released radially so that the centrifugal force due to spacecraft spin can increase the distance between the three impact sites.

The thrust tube is the primary structural element of the Multiprobe. All other structural elements are supported either directly or indirectly by the thrust tube. Since the thrust tube is the only structural element that attaches to the launch vehicle, it must react all loads imposed by the launch vehicle.

The requirement for maximum commonality with the Orbiter does not affect the general mechanical configuration of the Multiprobe; however, the size and thickness of certain items such as the equipment shelf were affected by the commonality requirement.

A scale drawing of the Multiprobe is shown in Figure 3.1.1.1-1 (Appendix C). Table 3.1.1.1-1 shows the location and identification of some units.

3.1.1.2 Spacecraft Coordinate System Definition. The spacecraft coordinate system is a right-handed orthogonal set of axes: X, Y and Z. The positive Z axis coincides with the geometric axis of symmetry of the spacecraft structure and is in the direction of the forward omni antenna. The X and Y axes lie in a plane perpendicular to the Z axis at launch vehicle station 171.9. The positive Y axis lies in the plane defined by the Z axis and the shelf split line and will be in the direction of the forward axial thruster. The X axis completes the orthogonal set. The spacecraft attach fitting/launch vehicle interface is parallel to the X-Y plane and is located at launch vehicle station 163.9 and spacecraft station 8.0. The spin of the spacecraft will be a positive rotation about the Z axis; i.e., the angular momentum vector will be in the nominal direction of the positive Z axis which coincides with the nominal launch velocity vector. The coordinate system is illustrated in Figure 2.1.1-1.

TABLE 3.1.1.1-1

LOCATION AND IDENTIFICATION OF CMD, DATA HANDLING, AND ADP UNITS

NAME	PART NUMBER	S/N	LOCATION (θ ANGLE; TOP IS CLOSEST TO S/C + Z AXIS)
Comm. Output Module	3280540-100	9	335° Middle
Comm. Output Module	3280540-100	25	335° Bottom
Comm. Output Module	3280540-100	27	160° Middle
Comm. Output Module	3280540-100	28	240° Bottom
Comm. Output Module	3280540-100	29	160° Middle
Comm. Output Module	3280540-100	30	240° Bottom
PCM Encoder	3280620-100	8	330° Top
PCM Encoder	3280620-100	9	330° Bottom
Dual Data Input Module	3280630-100	20	160° Top
Dual Data Input Module	3280630-100	25	240° Top
Dual Data Input Module	3280630-100	26	335° Top
Dual Data Input Module	3280630-100	27	285°
Telemetry Processor	3385034-100	3	260° Top
Telemetry Processor	3385034-100	5	260° Bottom
Command Processor	3385042-100	1	255° Top
Command Processor	3385042-100	4	255° Top
Attitude Data Processor	3385055-100	4	255° Bottom
Attitude Data Processor	3385055-100	5	255° Top

3.1.1.3 Structural Elements

3.1.1.3.1 Thrust Tube. The thrust tube is the primary structural element for reacting the launch loads. The thrust tube is a truncated conical frustrum with its axis of symmetry coincident with the spacecraft spin axis. The larger end of the thrust tube faces aft and serves as the separation plane between the spacecraft and the attach fitting. The forward end of the thrust tube supports the center area of the equipment shelf and the Probe support structure.

The thrust tube is a semimonocoque design with magnesium skins, three aluminum rings and twelve machined aluminum longerons. Two of the rings are used as structural interfaces at the ends of the thrust tube while the third is used to support the orbit insertion motor and part of the propulsion subsystem.

The spacecraft is attached to the attach fitting during launch by a Marman clamp that is shown in Figure 2.2.1.1-2. The "U" shaped segments or shoes of the clamp prevent vertical movement while the mating surfaces of the separation rings prevent lateral movement. The shoes are attached to two steel bands to form two semicircular segments. The two segments are joined together and tensioned by two explosive bolts; detonation of either explosive bolt causes the band to release which allows the spacecraft to separate from the launch vehicle with the aid of kick-off springs. The spacecraft separation is controlled from the launch vehicle. During spacecraft separation from the attach fitting, two separation switches on the spacecraft are actuated which initiate the preprogrammed initial spin-up sequence. This sequence is presented in Section 4.2.1.

3.1.1.3.2 Equipment Shelf. The equipment shelf supports the spacecraft units and scientific instruments. It is a circular honeycomb plate 97.5 inches in diameter with a 2.0 inch thick core and 0.010 inch aluminum facesheets. The center area of the equipment shelf is supported by the forward end of the thrust tube while the outer edge is supported by twelve one-inch diameter beryllium struts that terminate at the intersection of the twelve thrust tube longerons and the separation ring. The shelf is manufactured and assembled to the spacecraft in two semicircular segments. After assembly to the spacecraft, the two shelf halves are spliced together to reduce the vibration levels along the split line to levels comparable to those elsewhere on the shelf. The split line is oriented along the $\theta = 90°$, $270°$ line in spacecraft coordinates.

The layout of spacecraft units and scientific instruments on the forward side of the equipment shelf is shown in Figure 3.1.1.3.2-1. The aft side of the equipment shelf is used as a thermal radiator; the locations of the thermal louvers and heaters are shown by dotted lines in this same figure. Mounting points for the spacecraft units and scientific instruments are provided by implanting threaded inserts in the honeycomb core of the shelf with rigid foam.

3.1.1.3.3 Solar Array Substrate. The purposes of the solar array substrate are to support the solar cells that supply spacecraft power and to provide a physical closeout along the periphery of the spacecraft. The substrate is a 100-inch diameter cylinder that is 48 inches long. It is constructed from aluminum honeycomb and fiberglass facesheets. The substrate is aligned with its axis of symmetry coincident with the spacecraft centerline, and it is supported at twenty-four places along the periphery of the equipment shelf.

3.1.1.3.4 Probe Support Structure. The probe support structure attaches to the forward end of the bus spacecraft and is designed to support the probes during launch and transit and to permit the probes to separate from the bus so that they can enter the Venusian atmosphere. The Large Probe support is an inverted conical frustrum that attaches to the forward end of the thrust tube. It is a semimonoque design with two aluminum rings, eighteen aluminum longerons and magnesium skins. The axis of symmetry of the Large Probe support cone is coincident with the bus spacecraft axis of symmetry.

The Large Probe is attached to the support cone by three explosive nuts located at three of the longerons. Large probe separation is accomplished by firing these three nuts. The Large Probe separates from the bus spacecraft in the +Z direction (bus spacecraft coordinates), with the aid of kick-off springs.

The Small Probe support structure attaches to the Large Probe support cone and the equipment shelf. It consists of an aluminum honeycomb shear web and six tubular struts to support the outer edges. Each Small Probe sits in a horizontal semicircular fixed clamp that is attached to the shear web. Each end of each of the three clamps is supported by one of the previously mentioned struts while the middile is supported by a longeron on the Large Probe support cone. Each Small Probe is held against its fixed clamp by a semicircular flyaway clamp. When the Small Probes are to be released, an explosive nut at one end of each flyaway clamp is fired, thus allowing each clamp and Small Probe to move radially outward due to centrifugal force.

3.1.1.4 Electrical Harness. All unit electrical interconnections including ground returns are accomplished with the spacecraft harness. Except for the scientific instruments and the data input modules, all wiring is redundant either by unit or by wire; if two units are redundant, a single wire is used for each function of each unit, but if only one unit performs a function, the wires are looped or doubled so that any one wire break does not affect operation. In general, all signal and power grounds are kept separated from inside the unit to the spacecraft single point ground. The only exceptions to this are the transponder, power amplifier, drivers and certain science instruments. Sensitive signal lines and all pyrotechnic lines are shielded to prevent electromagnetic interference, with each shielded line grounded only at one point to avoid ground loops.

The wire harness is constructed from kapton insulated wire varying in size from 16 AWG to 26 AWG. Low current circuits (command, telemetry) use 26 AWG, the smallest size reliably capable of withstanding the mechanical environment. Except for the 16 AWG high current carrying wires, all power lines are 20 AWG for units using Cannon D connectors and 22 AWG for those using Microdot and Matrix connectors. These are the largest sizes these connectors will take. When the current to be carried exceeds the capability of a single wire, multiple wires are used. The individual wires are bundled and laced together and then tied to the equipment shelf. The harness is constructed in such a manner that minimum stress is applied to the wires. The connectors on the Multiprobe are Matrix double density connectors except that Microdot connectors are used where wire density is too great and Cannon Golden D's are used on units common to the Multiprobe. The harness is composed of four subharnesses - the two equipment shelf harnesses, the pyrotechnic harness, and the propulsion harness.

There is an equipment shelf harness on each half of the equipment shelf. It is routed around the thrust tube on the forward side of the shelf; wire bundles radiate outward from this loop as required. Bus power is taken from the solar panel at $\theta = 115°$ and $245°$. The wiring harness to each of the probes terminates in a 61-pin inflight disconnect (IFD). Upon actuation of a hot wire device, the bus IFD plug separates from the probe IFD receptacle and is pulled towards a catcher that retains the plug.

The pyrotechnic harness carries the firing current to all pyrotechnic devices. detonators. For safety, all wires are completely shielded in a Faraday cage and only ordnance circuits are allowed in the pyrotechnic harness and connectors.

The propulsion harness connects the propulsion subsystem electrical components to the equipment shelf harness. Since the propulsion units are located on the aft side of the shelf, the propulsion harness is routed through slots near the outer edge of the shelf.

3.1.2 Mass Properties. From launch until the end of the mission, the mass properties of the Multiprobe change significantly due to propellant usage and probe separation. During the mission, the center of mass moves in the +Z direction due to hydrazine usage and in the -Z direction due to probe separation. The longitudinal position of the center of mass as a function of propellant usage and key mission events is shown in Figure 3.1.2-1 for a nominal mission. Throughout the mission, the lateral position of the center of mass of the Multiprobe is essentially coincident with the Z axis.

The roll moment of inertia decreases during the mission due to hydrazine usage and probe separation. The change in roll moment of inertia as a function of propellant usage and key mission events is shown in Figure 3.1.2-2 for a nominal mission. The minimum and maximum lateral moments of inertia change in the same manner as the roll moment of inertia. The cchanges in minimum and maximum lateral moments of inertia are shown in Figure 3.1.2-2.

The details of the changes in mass, center of mass and moments and products of inertia throughout the nominal Multiprobe mission are given in Table 3.1.2-1.

TABLE 3.1.2-1

SEQUENCED MASS PROPERTIES DATA - PIONEER VENUS MULTIPROBE

Code	Description	Current Weight (Pounds)	Center of Gravity (In. From Ref. Datum) Z	X	Y	Moment of Inertia (Slug-Ft Square) IZ	IX	IY	Products of Inertia (Slug-Ft Square) IZX	IZY	IXY
4600	Bus EOI-CAP JETT	567.46	72.1	-0.0	0.0	163.1	106.7	100.3	-0.09	0.03	1.23
4750	N2H4 Increment	25.53	60.5	0.0	0.0	4.5	4.2	0.3	0.0	0.0	-1.11
4780	BNMS Cap Jettisoned	0.68	87.4	41.1	-12.6	0.0	0.0	0.0	0.0	0.0	0.0
4800	Bus After SP Sepn.	593.67	71.6	0.0	-0.0	167.8	111.6	101.6	0.00	0.00	0.04
5000	Small Probes (3)	595.09	93.5	-0.0	-0.0	204.3	103.6	103.7	-0.00	0.01	-0.06
5050	D/M Wt. Sep. from Bus	5.73	00.8	-0.0	-0.0	3.6	1.8	1.8	-0.00	-0.00	-0.00
5070	Multi-Prior SP Sepn	1,194.48	82.6	-0.0	-0.0	375.7	247.7	237.7	-0.02	-0.03	-0.03
5150	N2H4 Increment	4.15	60.9	0.0	0.0	0.6	0.6	0.0	0.0	0.0	-0.16
5200	Mid-Pt. Tgt. Maneuvers*	1,198.63	82.5	-0.0	-0.0	376.4	248.7	238.2	-0.02	-0.03	-0.18
5230	N2H4 Increment	4.15	61.0	0.0	0.0	0.6	0.6	0.0	0.0	0.0	-0.15
5270	Multi-After LP Sepn.	1,202.76	82.4	-0.0	-0.0	377.0	249.7	238.6	-0.02	-0.03	-0.33
5300	Large Probe	694.81	105.0	0.0	0.0	33.6	22.7	19.9	-0.00	-0.05	-0.06
5400	Multi-Prior LP Sepn.	1,897.59	90.7	-0.0	-0.0	410.6	320.7	306.8	-0.00	-0.03	-0.38
5550	N2H4 Increment	13.29	61.0	0.0	0.0	1.8	1.7	0.1	0.0	0.0	-0.44
5570	Mid-Pt. of First TCM*	1,910.88	90.5	-0.0	-0.0	412.4	324.8	309.4	-0.00	-0.03	-0.83
5580	N2H4 Increment	11.38	61.0	0.0	0.0	1.3	1.3	0.1	0.0	0.0	-0.34
5600	Multiprobe at Sepn.	1,922.26	90.3	-0.0	-0.0	413.7	328.2	311.7	-0.00	-0.03	-1.17
5700	Attach Fitting	63.91	25.7	0.1	-0.0	10.8	7.6	8.2	0.01	0.02	0.02
5800	Multiprobe - Launch	1,986.17	88.2	-0.0	-0.0	424.5	391.5	375.6	-0.09	0.00	-1.15
5850	Science Contingency	18.70	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5900	S/C Contingency	9.03	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
6000	Total Launch Weight	2,013.90									

*CG Affects Jet Alignment.

TABLE 3.1.2-1 (Continued) Multiprobe (Continued)

Code	Description	Current Weight (Pounds)	Moment of Inertia (Slug-Ft Square) IZ	IP(Max)	IP(Min)	PHI(Max)	IZ/IP(Max)	IZ/IP(Min)
4600	Bus EOL-CAP JETT	567.46	163.1	107.0	100.1	169.5	1.525	1.629
4750	N2H4 Increment	25.53	4.5	4.5	-0.0	15.0		
4780	ENMS Cap Jettisoned	0.68	0.0	0.0	0.0	0.0		
4800	Bus After SP Sepn.	593.67	167.8	111.6	101.6	179.8	1.503	1.652
5000	Small Probes (3)	595.09	204.3	103.7	103.6	49.8	1.970	1.973
5050	D/M Wt. Sep. from Bus	5.73	3.6	1.8	1.8	0.0	1.990	2.010
5070	Multi-Prior SP Sepn.	1,194.48	375.7	247.7	237.7	0.1	1.517	1.581
5150	N2H4 Increment	4.15	0.6	0.6	-0.0	15.0		
5200	Mid-Pt., Target Maneuvers	1,198.63	376.4	248.7	238.2	1.0	1.513	1.580
5230	N2H4 Increment	4.15	0.6	0.6	-0.0	15.0		
5270	Multi-After LP Sepn.	1,202.78	377.0	249.7	238.6	1.7	1.510	1.580
5300	Large Probe	694.81	33.6	22.7	19.9	1.3	1.481	1.686
5400	Multi-Prior LP Sepn.	1,897.59	410.6	320.7	306.8	1.6	1.280	1.338
5550	N2H4 Increment	13.29	1.8	1.8	-0.0	15.0		
5570	Mid-Pt. of First TCM	1,910.88	412.4	324.9	309.4	3.1	1.269	1.333
5580	N2H4 Increment	11.38	1.3	1.3	-0.0	15.0		
5600	Multiprobe at Sepn.	1,922.26	413.7	328.3	311.6	4.0	1.260	1.328
5700	Attach Fitting	63.91	10.8	8.2	7.6	91.5	1.314	1.430
5800	Multiprobe - Launch	1,986.17	424.5	391.6	375.5	4.1	1.084	1.131
5850	Science Contingency	18.70	0.0	0.0	0.0	0.0		
5900	S/C Contingency	9.03	0.0	0.0	0.0	0.0		
6000	Total Launch Weight.	2,013.90						

TABLE 3.1.2-1 (Continued)

PIONEER VENUS LARGE PROBE

Code	Description	Current Weight (Pounds)	Center of Gravity (In. From Ref. Datum) Z	X	Y	Moment of Inertia (Slug-Ft Square) IZ	IX	IY	Products of Inertia (Slug-Ft Square) IZX	IZY	IXY
9000	Pre/Post Sepn. -LP	694.81	0.1	-0.0	0.0	33.6	20.7	21.9	0.05	0.03	1.22
9200	LP - Post Entry	678.49	0.0	-0.0	0.0	32.0	19.7	21.0	0.05	0.03	1.22
9400	Parachute Descent	455.71									
9600	Final Descent - PV Mod.	438.49	0.3	0.0	-0.0	11.6	8.4	8.1	0.06	-0.05	-0.09

Code	Description	Current Weight (Pounds)	Moment of Inertia (Slug-Ft Square) IZ	IP(Max)	IP(Min)	PHI(Max)	IZ/IP(Max)	IZ/IP(Min)
9000	Pre/Post Sepn. - LP	694.81	33.6	22.7	19.9	121.3	1.481	1.686
9200	LP - Post Entry	678.49	32.0	21.7	19.0	121.3	1.472	1.685
9400	Parachute Descent	455.71						
9600	Final Descent - PV Mod.	438.49	11.6	8.5	8.1	15.2	1.372	1.432

TABLE 3.1.2-1 (Continued)

PIONEER VENUS SMALL PROBE 1

Code	Description	Current Weight (Pounds)	Center of Gravity (In. From Ref. Datum) Z	X	Y	Moment of Inertia (Slug-Ft Square) IZ	IX	IY	Products of Inertia (Slug-Ft Square) IZX	IZY	IXY
8000	Pre/Post Sepn - SP	197.99	0.2	-0.0	-0.0	3.0	2.0	2.0	0.00	-0.00	-0.07
8200	Entry - Aft Yo-Yo Jett.	197.11	0.2	-0.0	-0.0	2.9	2.0	2.0	0.00	-0.00	-0.08
8400	SP - Post Entry	189.74	0.1	-0.0	-0.0	2.7	1.8	1.8	0.00	-0.00	-0.08
8600	Small Probe Descent	189.74	0.1	-0.0	-0.0	2.7	1.9	1.8	-0.00	0.00	-0.07

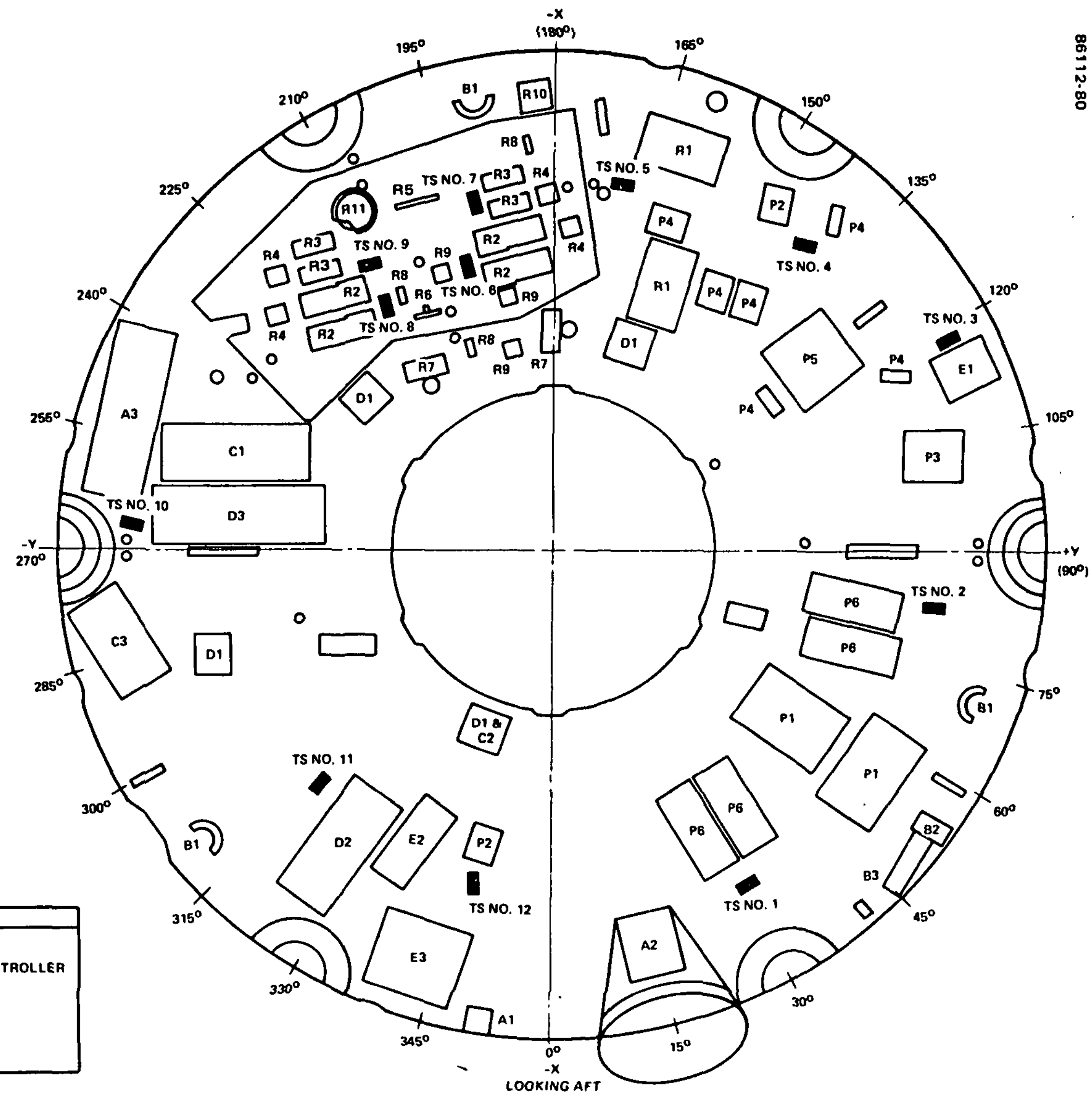
Code	Description	Current Weight (Pounds)	Moment of Inertia (Slug-Ft Square) IZ	IP(Max)	IP(Min)	PHI(Max)	IZ/IP(Max)	IZ/IP(Min)
8000	Pre/Post Sepn. - SP	197.99	3.0	2.0	1.9	40.3	1.447	1.549
8200	Entry - Aft Yo-Yo Jett.	197.11	2.9	2.0	1.9	41.1	1.433	1.559
8400	SP - Post Entry	189.74	2.7	1.9	1.7	41.1	1.416	1.551
8600	Small Probe Descent	189.74	2.7	1.9	1.8	35.8	1.426	1.546

```
******************************************
******************************************
****                                  ****
****    This Figure is a Foldout.     ****
****                                  ****
****         See APPENDIX C           ****
****                                  ****
******************************************
******************************************
```

Figure 3.1.1.1-1

Revision

86112-80

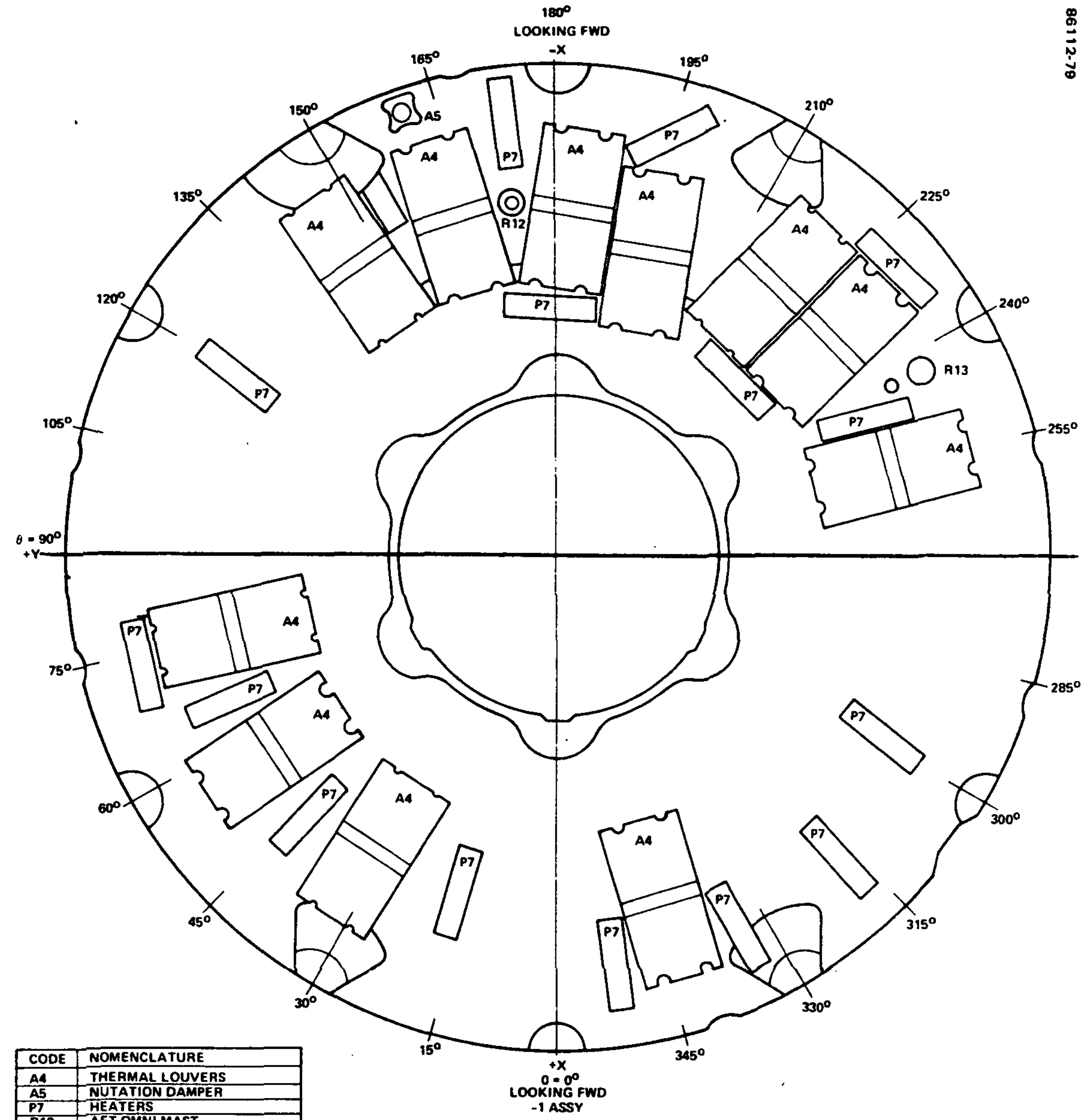


NO.	MNEMONIC
TS1	SHL01T
2	SHL02T
3	SHL03T
4	SHL04T
5	SHL05T
6	SHL06T
7	SHL07T
8	SHL08T
9	SHL09T
10	SHL10T
11	SHL11T
12	SHL12T

CODE	NOMENCLATURE
E1	BIMS SENSOR
E2	BNMS ELECTRONICS
E3	BUMS SENSOR
R1	TRANSPONDER
R2	POWER AMP
R3	DRIVER AMP
R4	ISOLATOR
R5	6 DB HYBRID
R6	3 DB HYBRID
R7	DIPLEXER
R8	S-SWITCH
R9	C-SWITCH
R10	SWITCH DRIVER
R11	FWD OMNI MAST
D1	DUAL DATA INPUT MODULE
D2	PCM ENCODER
D3	TM PROCESSOR
C1	COMMAND PROCESSOR
C2	COMMAND OUTPUT MODULE
C3	PYRO CONTROL UNIT
A1	SUN SENSOR ASSEMBLY
A2	STAR SENSOR
A3	ATTITUDE DATA PROCESSOR

CODE	NOMENCLATURE
P1	CHARGE/DISCHARGE CONTROLLER
P2	BUS VOLTAGE LIMITER
P3	UNDER VOLTAGE/OVER VOLTAGE CONTROLLER
P4	CURRENT SENSOR
P5	POWER INTERFACE
P6	BATTERY PACK (2 PER BATTERY)
B1	SMALL PROBE IFD BRACKET
B2	UMBILICAL BRACKET
B3	OEP/BSP BRACKET

FIGURE 3.1.1.3.2-1A. BUS SHELF LOCATIONS OF MOST UNITS AND SHELF MOUNTED TEMPERATURE SENSORS



CODE	NOMENCLATURE
A4	THERMAL LOUVERS
A5	NUTATION DAMPER
P7	HEATERS
R12	AFT OMNI MAST
R13	MEDIUM GAIN HORN SUPPORT

FIGURE 3.1.1.3.2-1B. BUS SHELF LOCATIONS OF SOME UNITS

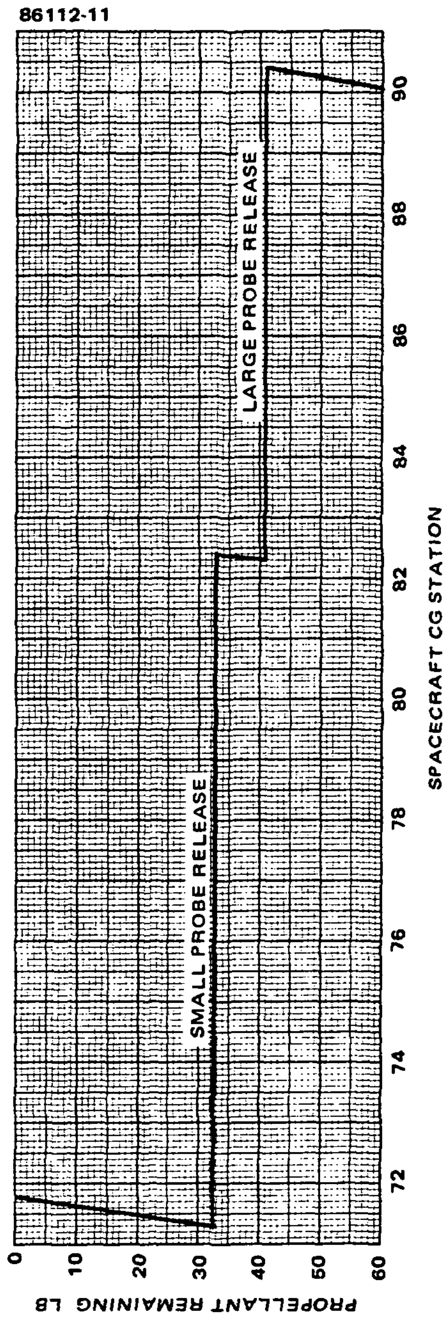


FIGURE 3.1.2-1. MULTIPROBE CG LOCATION VERSUS PROPELLANT REMAINING FOR A NOMINAL MISSION

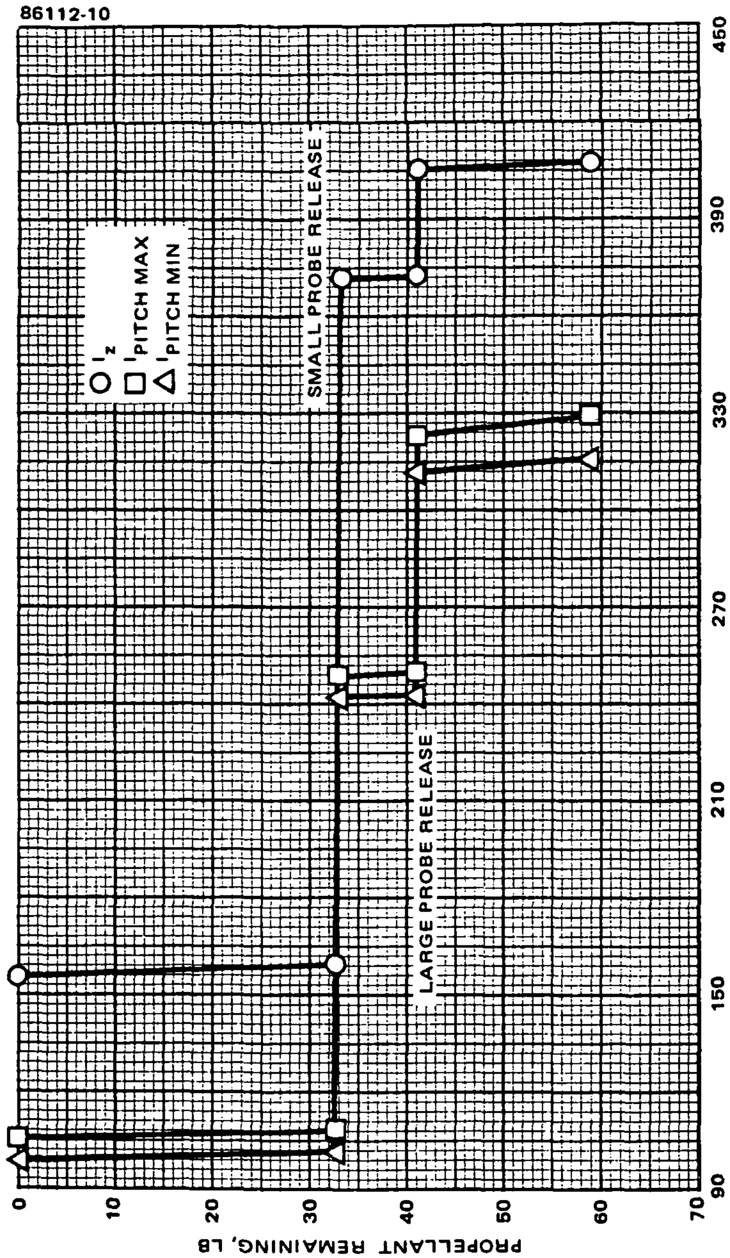


FIGURE 3.1.2-2. MULTIPROBE MOMENTS OF INERTIA VERSUS PROPELLANT REMAINING FOR A NOMINAL MISSION

This Page Intentionally Left Blank.

3.2 THERMAL DESIGN

3.2.1 General Description. The thermal design of the Multiprobe is based on isolating the spacecraft units and scientific instruments from the external environmental extremes encountered during the mission to the maximum extent practical and using a combination of passive and active elements to maintain acceptable unit temperatures. The primary factors affecting the thermal design of the Multiprobe are the increase in solar intensity as the spacecraft travels towards Venus, the variations in internal power dissipation during different phases of the mission, the variations in sun angle during the mission and the different configurations due to probes separation. In addition, the Multiprobe is designed to accommodate failure modes including extreme under powered conditions and extreme off nominal mission attitude conditions. These failure mode conditions plus the variation in environments encountered during the normal mission require the thermal design to be highly adaptive and almost completely automatic. The only elements that can be controlled from the ground are certain propulsion heaters, the Probe heaters and the bus limiters that control the shelf heaters.

The equipment compartment is the space defined by the equipment shelf, the forward thermal barrier and the solar array cylinder. Multilayer Kapton insulation blankets are used to isolate this compartment from the external environment. The isolation is not complete due to the inherent limitations of any insulation system and the various penetrations for spacecraft units and scientific instruments that must view out to space. In addition, the aft side of the equipment shelf is used as the primary radiator for the electrical power dissipation inside the equipment compartment. Eleven thermal louvers are mounted on this radiator surface to help accommodate the large variations in power dissipation by varying the effective thermal emittance as a function of shelf temperature.

Heaters are provided to maintain all elements of the propulsion subsystem at 40°F or higher to prevent propellant freezing. Some of these heaters are commandable on-off, others are commandable primary-secondary, and some are continuously powered. Commandable heaters are provided to control minimum Large and Small Probe temperatures. Heaters are also provided on the equipment shelf to provide additional dissipation during an extreme powered down condition such as an Undervoltage/Overhead (UV/OL) trip. Automatic shelf heater turn-on is required to counteract a sustained UV/OL trip that could occur when the spacecraft is not being monitored. In addition, a heater turn-on override is provided to prevent battery depletion if such a trip occurs because of insufficient solar array power or an extended eclipse condition.

Thermal finishes are used on externally mounted units such as antennas to limit their maximum temperatures and on external surfaces of the spacecraft to achieve thermal balance within the acceptable temperature range.

All heater and temperature sensors, including applicable command and telemetry mnemonics are shown in Figures 3.1.1.3.2-1, 3.2.1-1, 3.4.1-3, 3.8.1.2-1 and 3.8.1.2-2. All temperature sensors are listed and located in Table 3.2.2.5-1.

3.2.2 Thermal Control Units and Components

3.2.2.1 Heaters

3.2.2.1.1 Propulsion Heaters. All elements of the propulsion subsystem except the semicircular run of pressurant line are heated to maintain the propellant temperature at 40°F or higher. The mechanically redundant components and units which are located downstream of the latch valves have non-redundant heaters while the non-redundant components and units upstream of the latch valves have redundant heaters. Figure 3.2.2.1.1-1 shows this division between redundant and non-redundant heaters schematically.

The non-redundant heaters on the radial thrusters and the lines and components are continuously powered from the essential bus; there is no way that these heaters can be turned off either by ground command or any onboard failure mode sequence. The satisfactory operation of these heaters can be verified by monitoring Radial jets 1 through 4 temperatures telemetry channels (VJET1T through VJET4T), R1 and R2 line heater (voltage) status (VJ12HS), and R3 and R4 line heater (voltage) status (VJ34HS).

The forward axial thruster, and the aft axial thruster and associated lines and components both have on-off commandable non-redundant heaters. The forward axial thruster heater is commanded on by HTJ19 (HTJA9 is the redundant command) and commanded off by HTJ1Ø (HTJAØ). The aft axial thruster heater is commanded on by HTJ29 (HTJB9) and commanded off by HTJ2Ø (HTJBØ). All Axial Jet heaters should be ON at launch, and left ON, unless the following occurs: The Forward Axial Jet heater should be commanded OFF if (1) Forward Axial Jet 5 Temperature (VJET5T) reaches or exceeds +154°F <u>or</u> (2) Propellant Line Temperature 2 (VLIN2T) reaches or exceeds +145°F. The Aft Axial Jet heaters should be commanded OFF if (1) Aft Axial Jet 6 Temperature (VJET6T) reaches or exceeds +154°F <u>or</u> (2) Propellant Line Temperature 3 (VLIN3T) <u>or</u> Propellant Line Temperature 4 (VLIN4T) reaches or exceeds +145°F. Whichever of the Jet heaters had been turned OFF should be commanded ON again if any of the associated telemetry temperature indicators identified above reaches or decreases below +61°F. The Forward Axial Jet Heaters are expected to be required OFF following Spacecraft spin axis precession from an ecliptic normal attitude at E-28 days.

The redundant heaters upstream of the latch valves are command selectable between the primary and secondary circuits; one circuit is always powered even during extreme powered down conditions. The command for selecting the primary circuit is HTT19 (HTTA9); the command for selecting the secondary circuit is HTT29 (HTTB9). Satisfactory operation of either of the circuits can be verified by the following telemetry: tanks temperatures (VTNK1T and VTNK2T) PRI/SEC Prop. Tank Heater Status (PHTTKS), L/V Primary heater status (VALVHS), and F&D VALVE Primary Heater status (VFDVHS). The primary circuit should be used throughout the mission; the secondary heater circuit should be switched in if a decrease in Tank 1 temperature (VTNK1T), or Tank 2 temperature (VTNK2T), or propellant Line Temperature 1 (VLIN1T) is observed which is unrelated to environmental changes (e.g., unit switching, attitude change w.r.t. the sun l.o.s.), or occurs for steady-state environmental conditions.

3.2.2.1.2 Shelf Heaters. Heaters are required on the equipment shelf to provide additional dissipation during an extreme powered down condition such as an Undervoltage/Overload (UV/OL) trip. Automatic heater turn-on is required to counteract a sustained UV/OL trip that could occur when the spacecraft is not being monitored. This is accomplished by using thermostatic switches on the equipment shelf to lower the set points on two of the bus voltage limiters of the power subsystem. When the battery shelf temperature decreases to 35 ±5°F, the thermal switches mounted on it close, thereby forcing the first 33 watts of excess solar panel power into the battery shelf heaters and the next 33 watts into the RF shelf heaters. The thermal switches open at 55 ±5°F. Similarly when the RF shelf temperature decreases to 60 ±5°F, the thermal switches mounted on it close thereby forcing the first 33 watts of excess solar panel power into the battery shelf heaters and the next 33 watts into the RF shelf heaters. The thermal switches open at 70 ±5°F. Supplying power to the battery shelf first regardless of which thermal switch is tripped ensures adequate heat for the batteries even in the event of a failure in the battery shelf heater or heater control circuitry. The fact that the RF shelf is warmer than the battery shelf due to greater power dissipation dictates the use of the relatively high 60°F set point for the RF shelf thermal switch. Should a failure in the battery shelf heater or heater circuitry occur, the RF shelf temperature will drop to 60°F or less and thus activates the thermal switch before the batteries reach their minimum allowable temperature.

The locations of the shelf heaters and thermostatic switches are shown in Figures 3.8.1.2-1 and 3.8.1.2-2.

Since the bus limiters dissipate only excess solar panel power, the shelf heater cannot operate during conditions such as an off attitude failure mode where there is not an excess of solar panel current. Shelf heaters can be turned off by turning off the appropriate bus limiter.

The functioning of the shelf heaters is discussed more fully in the Power/Thermal Management Section 4.3.4.

3.2.2.1.3 Probe Heaters. The Large and Small Probes have commandable shelf heaters to maintain internal Probe temperatures early in the mission. The Large Probe uses two 17 watt heaters on the forward shelf and a 2 watt heater on the aft shelf. The Small Probes each use one 2 watt heater on the forward shelf. All of the heaters are powered and commanded from the Bus spacecraft. The commands for controlling the Probe heaters are given in PC-455, Reference paragraph 1.5.1.

The probe heaters are designed to maintain the internal Probe temperature at or above the minimum non-operating temperature of -40°F at all times. In addition, the Large Probe heaters will raise the internal Probe temperature to at least the minimum operating temperature of -4°F by (Launch + 60 days) so that Large Probe checkout can be accomplished. Large Probe temperatures can be verified by examining telemetry channels SLFWIT and SLAFIT for the forward and aft shelf temperature. Similarly SXFWIT and SXAFIT are the telemetry channels for the Small Probe forward and aft shelf temperatures where X is either 1, 2 or corresponding to Small Probe 1, 2 or 3.

The Large Probe heaters should be turned off when either telemetered shelf temperature exceeds +60°F without other internal power dissipation; the Small Probe heaters should be turned off when either telemetered shelf temperature exceed +60°F without other internal power dissipation. After removal of heater power, the Probe heater should be turned ON again if either telemetered shelf temperature reaches or decreases below +5°F.

The effects of the thermal design on mission operations are discussed in Sections 5.1.2.1.2.1 and 5.2.2.1.2.1.

3.2.2.2 <u>Thermal Louvers</u>. The eleven thermal louvers mounted on the aft side of the equipment shelf are used to regulate the heat flow from the equipment shelf by varying the effective thermal emittance with movable louver blades. When the blades are fully closed, their polished aluminum surface has an effective thermal emittance of 0.1. When the louver blades are fully open, the exposed quartz mirror has an effective thermal emittance of 0.61.

Each thermal louver covers an 8-inch by 16-inch area and has four pairs of louver blades. Each pair of louver blades is actuated by a coiled bi-metallic strip mounted in a housing between the blades; the louver blade pair is rotated from the closed to the open position as the bi-metallic strip is heated. The bi-metallic actuators are conductively isolated from the louver structure so that the louver blade position is determined by the heat radiated from the equipment shelf. Each blade pair in a particular louver is set to open and close at the same temperature. However, there are two different set point ranges used in order to obtain the proper unit temperatures. Nine louvers are set to close completely at or above 50°F and to open completely at or below 85°F. The remaining two louvers which are used under the batteries are set to close completely at or above 40°F and to open completely at or below 75°F. The physical arrangement of these two different louver types is shown in Figure 3.1.1.1-1 (Appendix C).

The thermal dissipation from the louver varies with temperature due to the position of the louver blades and the temperature of the radiating surface. In addition, louver performance is affected by spacecraft spin induced radial acceleration that increases the bearing friction and thus creates a hysteresis effect. The performance of the two louver types is shown in Figures 3.2.2.2-1 and 3.2.2.2-2.

3.2.2.3 Thermal Doublers. Because of the honeycomb construction of the equipment shelf, lateral heat conduction is relatively low. In order to adequately spread out the heat from high power units so that it can be dissipated, a thermal doubler is used. One thermal doubler on the Multiprobe is a 0.070-inch thick beryllium plate mounted between the high power communications equipment such as the power amplifiers and the equipment shelf. Another is used between the star sensor and the equipment shelf.

To enhance the thermal conduction across the unit mounting interfaces, RTV-566 is applied to the mounting surfaces of units with high power dissipation.

3.2.2.4 Insulation Blankets. Insulation blankets are used on the spacecraft to help isolate it from the environmental extremes encountered during the mission. The blankets are made from fifteen layers of aluminized Kapton. The interior layers of these blankets are 1/4 mil thick, aluminized on one side only, and crinkled to achieve the minimum contact between layers. To minimize weight, separators are not used between layers. The outer layer is 1 mil Kapton, aluminized on one side only with the bare Kapton side exposed.

The blankets are made in various segments to ease access and assembly. The blanket segments are attached to each other with Velcro fasteners. In order to minimize heat leaks through these attachments, the number of fasteners and the stitching used to attach the Velcro to the blankets is minimized.

3.2.2.5 Temperature Sensors. Temperature sensors are mounted throughout the spacecraft to measure and provide telemetry of the thermal status of the spacecraft and selected units. Table 3.2.2.5-1 lists each of these temperature sensors, its location, reference designator, and it's Hughes part number.

3.2.3 Solar Heating Restrictions. The normal attitude of the Multiprobe is with the spin axis perpendicular to the sun's rays (sun angle of 90° as measured from the forward spin axis). Planned and unplanned excursions from this normal attitude affect the thermal balance of the spacecraft. Depending on the operating mode of the spacecraft and its distance from the sun, off attitude orientations can be sustained for varying lengths of time before spacecraft overheating occurs. Figure 3.2.3-1 shows the time limits for various sun angles as a function of time from launch and spacecraft operating mode.

3.2.4 Ranges of Expected Mission Temperatures. Table 3.2.4-1 gives the range of expected temperatures in a nominal mission for each telemetered temperature.

TABLE 3.2.2.5-1

THERMAL TELEMETRY DESCRIPTION

Reference Designator	Telemetry Description	HAC Part Number
	COMMUNICATIONS SUBSYSTEM	
RVCO1T	Receiver 1 VCO Temperature	TI RTH-42
RVCO2T	Receiver 2 VCO Temperature	TI RTH-42
ROSC1T	Transmitter Aux Oscillator 1 Temp	TI RTH-42
ROSC2T	Transmitter Aux Oscillator 2 Temp	TI RTH-42
RAMP1T	Power Amp 1 Temperature	908683-4
RAMP2T	Power Amp 2 Temperature	908683-4
RAMP3T	Power Amp 3 Temperature	908683-4
RAMP4T	Power Amp 4 Temperature	908683-4
	CONTROL SUBSYSTEM	
ASTART	Star Sensor Temperature	908631-32
	POWER SUBSYSTEM	
PB1P1T	Battery 1, Pack 1 Temperature	908635-4
PB1P2T	Battery 1, Pack 2 Temperature	908635-4
PB2P1T	Battery 2, Pack 1 Temperature	908635-4
PB2P2T	Battery 2, Pack 2 Temperature	908635-4
PPAN1T	Solar Panel 1 Temperature	908683-4
PPAN2T	Solar Panel 2 Temperature	908683-4
	PROPULSION	
VTNK1T	Tank 1 Temperature	908631-32
VTNK2T	Tank 2 Temperature	908631-32
VJET1T	Radial Jet 1 Temperature	908683-4
VJET2T	Radial Jet 2 Temperature	908683-4
VJET3T	Radial Jet 3 Temperature	908683-4
VJET4T	Radial Jet 4 Temperature	908683-4
VJET5T	Fwd Axial Jet 5 Temperature	908683-4
VJET6T	Aft Axial Jet 6 Temperature	908683-4
VLIN1T	Propellant Line Temp 1	908631-32
VLIN2T	Propellant Line Temp 2	908631-32
VLIN3T	Propellant Line Temp 3	908631-32
VLIN4T	Propellant Line Temp 4	908631-32

TABLE 3.2.2.5-1. (Continued)

Reference Designator	Telemetry Description	HAC Part Number
	STRUCTURE-HARNESS SUBSYSTEM	
SHLØ1T	Shelf Temperature 1	908631-32
SHLØ2T	Shelf Temperature 2	908631-32
SHLØ3T	Shelf Temperature 3	908631-32
SHLØ4T	Shelf Temperature 4	908631-32
SHLØ5T	Shelf Temperature 5	908631-32
SHLØ6T	Shelf Temperature 6	908631-32
SHLØ7T	Shelf Temperature 7	908631-32
SHLØ8T	Shelf Temperature 8	908631-32
SHLØ9T	Shelf Temperature 9	908631-32
SHL1ØT	Shelf Temperature 10	908631-32
SHL11T	Shelf Temperature 11	908631-32
SHL12T	Shelf Temperature 12	908631-32
SBNMST	BNMS Temperature	908631-32
SBIMST	BIMS Temperature	908631-32

TABLE 3.2.4-1

RANGE OF EXPECTED MISSION TEMPERATURES

NOTE: Representative nominal mission temperatures are described in Reference: 1.5.33).

Reference Designator	Telemetry Description	Expected Mission Temperatures, °F	
		Maximum	Minimum
	COMMUNICATIONS SUBSYSTEM		
RVCO1T	Receiver 1 VCO Temperature	108	57
RVCO2T	Receiver 2 VCO Temperature	106	55
ROSC1T	Transmitter Aux Oscillator 1 Temperature	106	56
ROSC2T	Transmitter Aux Oscillator 2 Temperature	106	51
RAMP1T	Power Amp 1 Temperature	128	75
RAMP2T	Power Amp 2 Temperature	126	81
RAMP3T	Power Amp 3 Temperature	123	45
RAMP4T	Power Amp 4 Temperature	115	47
	CONTROL SUBSYSTEM		
ASTART	Star Sensor Temperature	128	43
	POWER SUBSYSTEM		
PB1P1T	Battery 1, Pack 1 Temperature	93	30
PB1P2T	Battery 1, Pack 2 Temperature	93	30
PB2P1T	Battery 2, Pack 1 Temperature	93	36
PB2P2T	Battery 2, Pack 2 Temperature	93	38
PPAN1T	Solar Panel 1 Temperature	146	-64
PPAN2T	Solar Panel 2 Temperature	146	-64
	PROPULSION		
VTNK1T	Tank 1 Temperature	122	78
VTNK2T	Tank 2 Temperature	122	76
VJET1T	Radial Jet 1 Temperature	112	45
VJET2T	Radial Jet 2 Temperature	117	40
VJET3T	Radial Jet 3 Temperature	122	53
VJET4T	Radial Jet 4 Temperature	113	49
VJET5T	Fwd Axial Jet 5 Temperature	136	50
VJET6T	Aft Axial Jet 6 Temperature	116	55
VLIN1T	Propellant Line Temp 1	117	53
VLIN2T	Propellant Line Temp 2	99	60
VLIN3T	Propellant Line Temp 3	113	49
VLIN4T	Propellant Line Temp 4	110	61

TABLE 3.2.4-1 (Continued)

Reference Designator	Telemetry Description	Expected Mission Temperatures, °F	
		Maximum	Minimum
	STRUCTURE-HARNESS SUBSYSTEM		
SHLØ1T	Shelf Temperature 1	92	10
SHLØ2T	Shelf Temperature 2	96	25
SHLØ3T	Shelf Temperature 3	117	35
SHLØ4T	Shelf Temperature 4	109	50
SHLØ5T	Shelf Temperature 5	101	49
SHLØ6T	Shelf Temperature 6	115	68
SHLØ7T	Shelf Temperature 7	104	61
SHLØ8T	Shelf Temperature 8	111	48
SHLØ9T	Shelf Temperature 9	104	49
SHL1ØT	Shelf Temperature 10	116	52
SHL11T	Shelf Temperature 11	116	27
SHL12T	Shelf Temperature 12	97	15
SBNMST	BNMS Temperature (NOTE 1)	113	19
SBIMST	BIMS Temperature (NOTE 2)	135	33

NOTE 1: Temperature predictions are for thermal model Node #301, and NOT for the telemetered temperature sensor whose location is not well defined.

NOTE 2: Temperature predictions are for thermal model Node #303, and NOT for the telemetered temperature sensor whose location is not well defined.

```
****************************************
****************************************
****                                ****
****    This Figure is a Foldout.   ****
****                                ****
****         See APPENDIX C         ****
****                                ****
****************************************
****************************************
```

Figure 3.2.1-1

Revision

86112-12

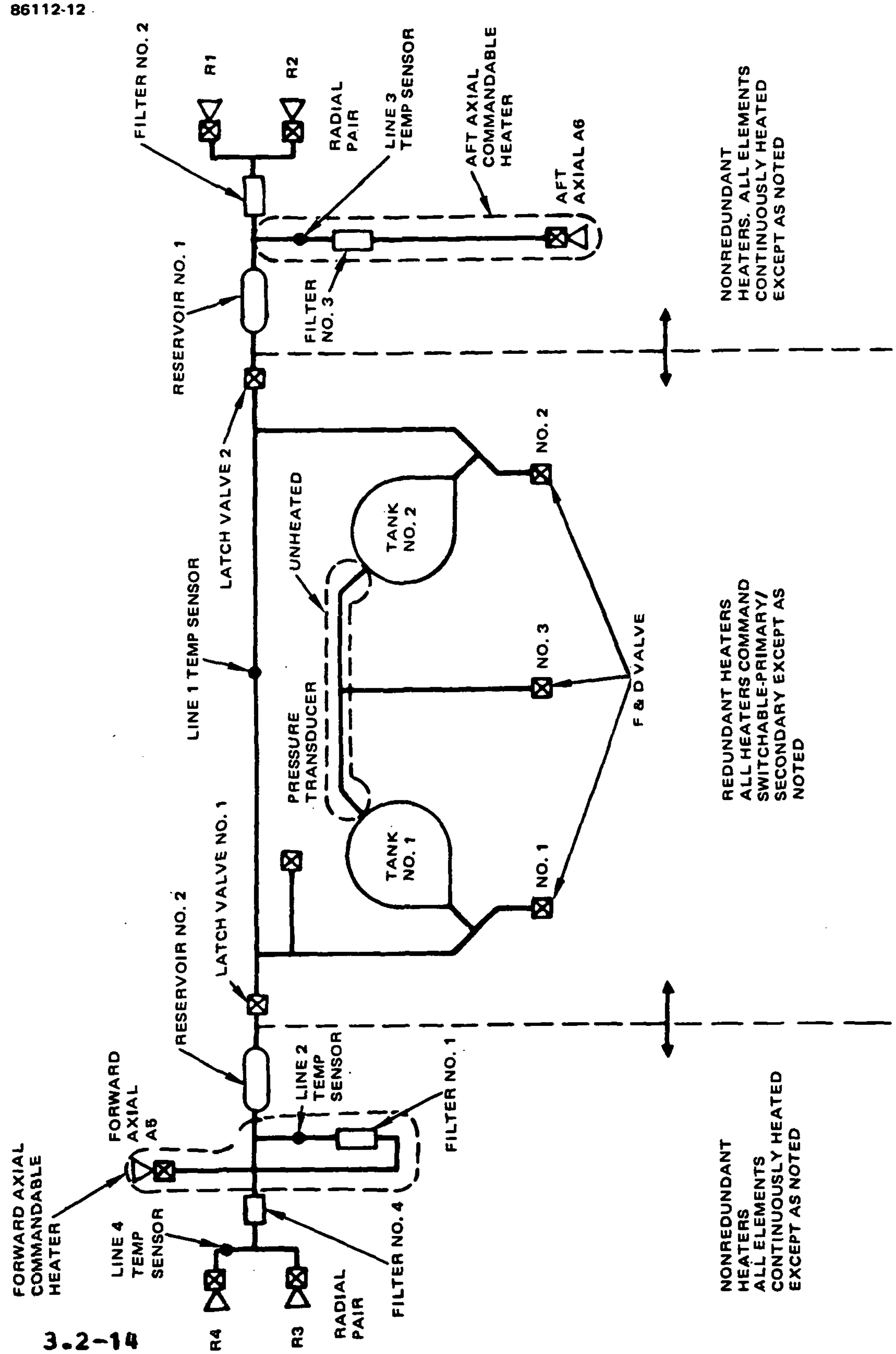


FIGURE 3.2.2.1.1-1. MULTIPROBE PROPULSION SUBSYSTEM HEATER CONFIGURATION

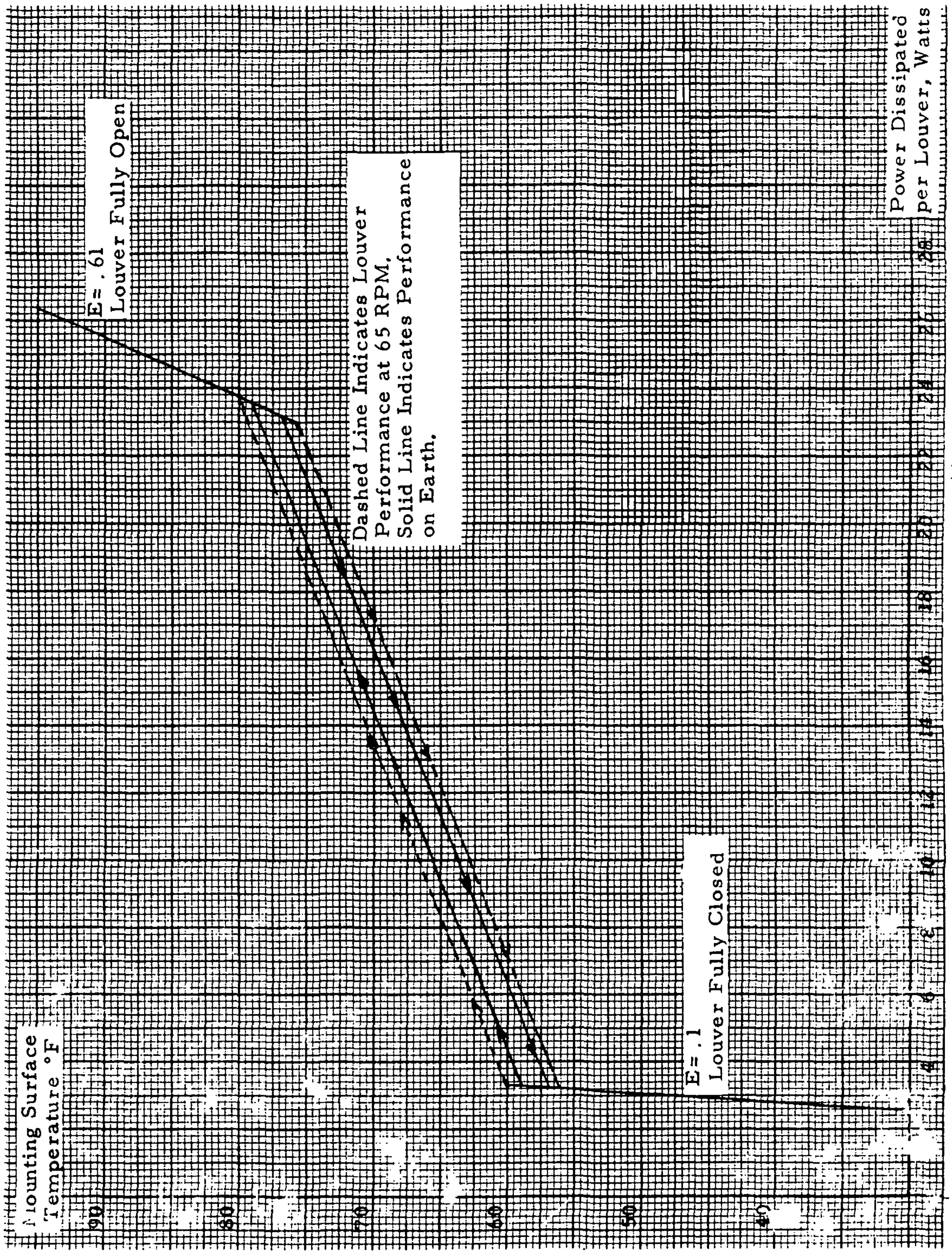


Figure 3.2.2.2-1. Louver Performance (General Type)

Section No. 3.2.4
Doc. No. PC-403
Orig. Issue Date 5/22/78
Revision No.

Revision

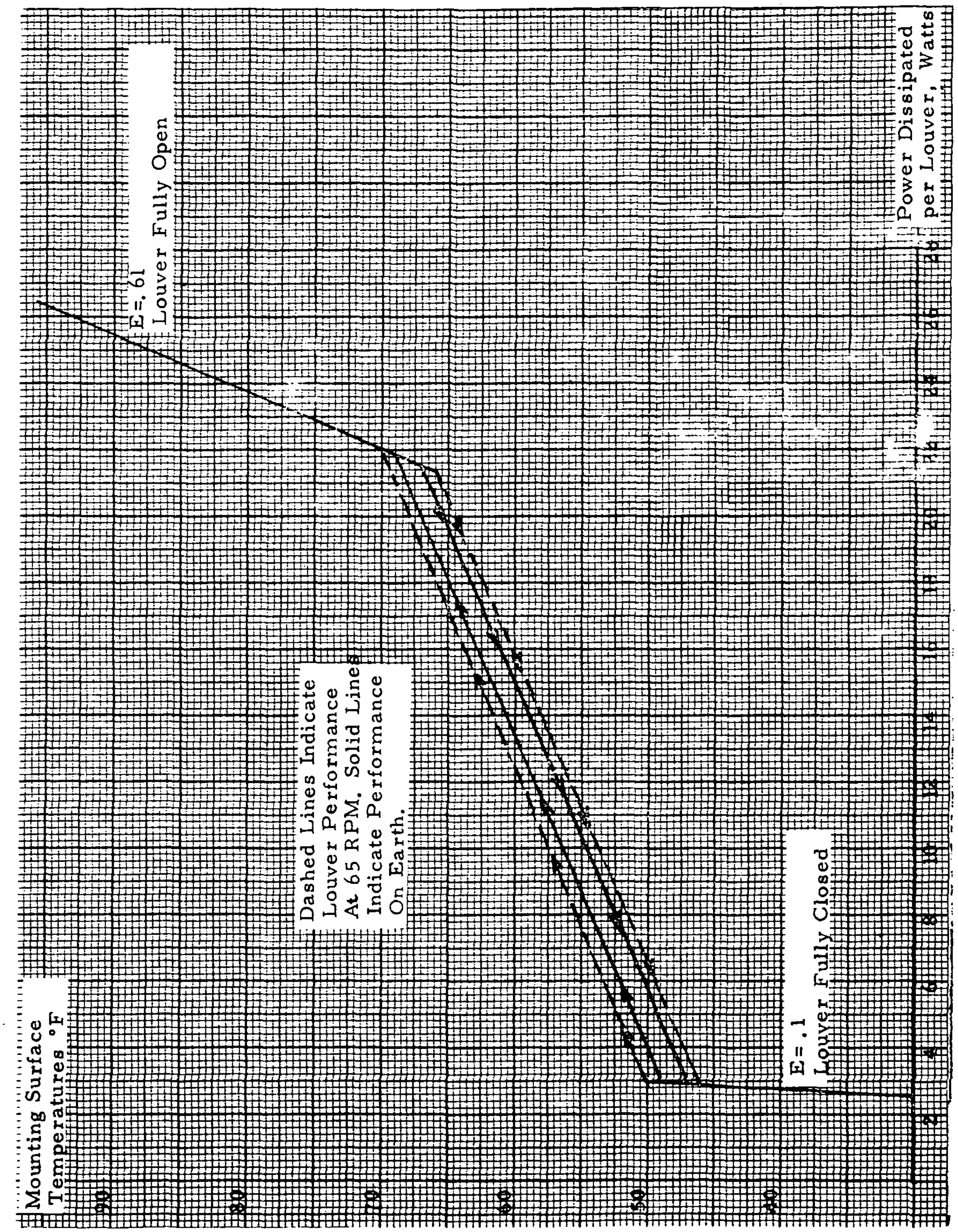


Figure 3.2.2.2-2. Louver Performance (Battery Type)

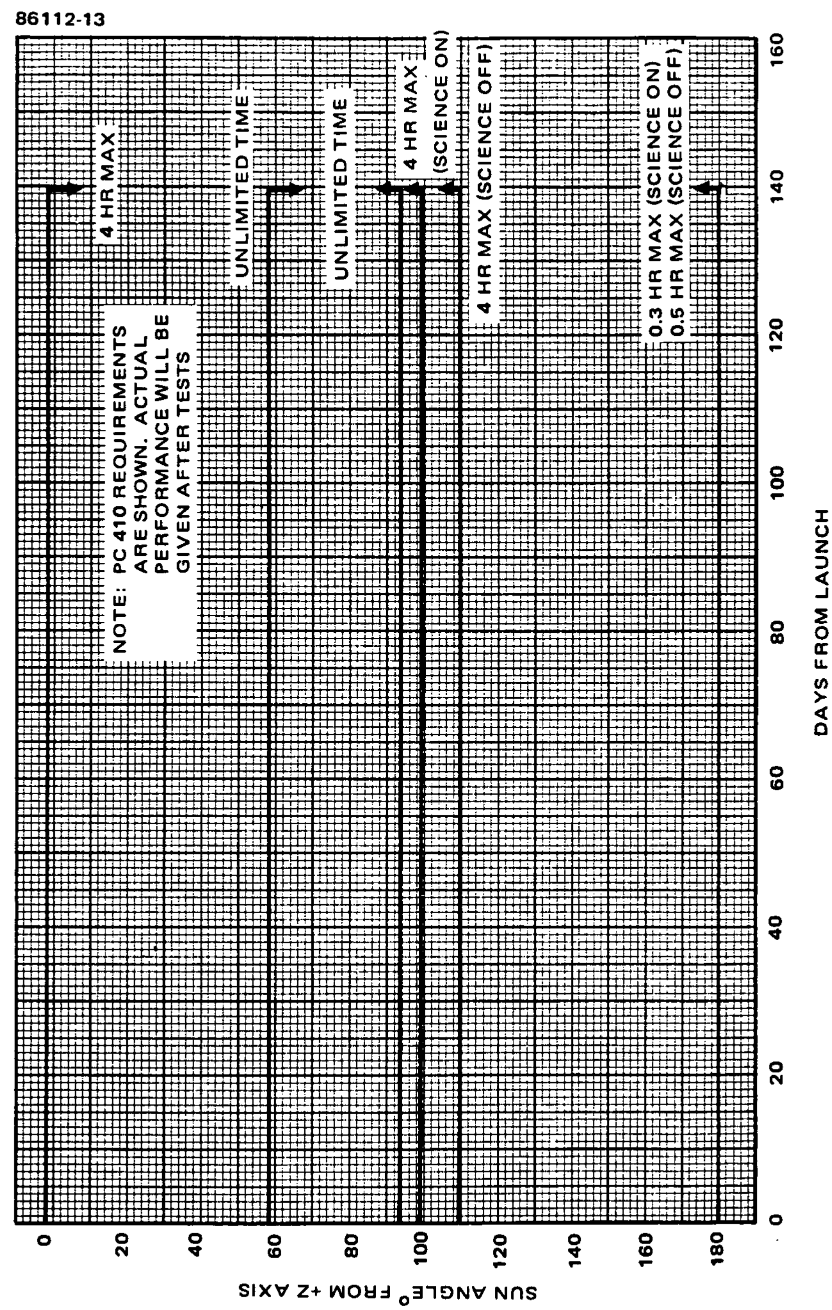


FIGURE 3.2.3-1. ALLOWABLE TIME AT VARIOUS SUN ANGLES

This Page Intentionally Left Blank.

3.3 CONTROL SUBSYSTEM

3.3.1 Subsystem Description

3.3.1.1 Functional Description. The control subsystem provides the sensors and onboard logic to accomplish the following functions:

(a) Spin rate and spacecraft attitude determination.
(b) Roll reference signals for science instruments and for timing the spin angle release of the Small Probes.
(c) Control of thrusters for spin axis precession maneuvers, spin speed control, and spacecraft velocity maneuvers.
(d) Nutation damping.

The simplified block diagram of Figure 3.3.1-1 illustrates the control subsystem elements that perform these functions.

The attitude determination design utilizes dual slit sun and star sensors to measure the sun and star positions relative to spacecraft coordinates. The sensors detect the selected celestial target crossing through their field-of-view (FOV) each spin period due to vehicle spin rotation. Both sensors use silicon detectors to develop an electrical pulse when the target crosses the FOV. The star sensor is capable of detecting 25 (≥ +1.5 silicon magnitude) stars. Time interval measurements are made between command selected sensor outputs and then telemetered to a ground station for computation of spacecraft attitude. A spin angle star gate, changeable in time occurrence in the spin period by command, is utilized in conjunction with a star detection threshold setting that is changeable in threshold level by command to provide discrimination of star references.

The inertial reference can be command selected to be either the sun or a star. This detected signal then becomes the onboard roll reference for attitude maneuvers, attitude measurements and the science reference signals. A "free-run" mode is also selectable which substitutes an electronically derived pulse for the inertial reference.

The Multiprobe spacecraft utilizes six thrusters to provide the capability for performing attitude maneuvers, spin speed control, and velocity change maneuvers. Possible thruster configurations and the selection required to perform a desired maneuver are discussed in Section 3.4. The control subsystem provides the capability of command selecting the thrusters and controlling the fire mode and firing duration. Thrusters may be fired in a pulsed or continuous mode. In the pulse mode the firing pulse can be command selected to occur at any azimuth angle. An additional option permits the generation of a firing pulse 180 degrees from the first pulse (alternate fire mode). Further options permit two selectable pulse widths (128 ms or 512 ms) and the pulsed firing duration to be controlled by time or pulse count. Details and application of this feature are given in Sections 3.3.2.3.5, 3.3.3, 3.3.4, and 4.3.1.

The control subsystem also provides passive nutation damping of the spacecraft.

3.3.1.2 Hardware Implementation. The equipment required to perform the control subsystem functions consists of an attitude data processor, sun sensor assembly, star sensor assembly, and damper. The functions performed by these units are as follows:

(a) Attitude Data Processor (ADP). The ADP is a command controlled unit which processes the sun and star inputs for attitude determination and on-board roll reference, controls thruster firing modes for precession, velocity and spin maneuvers, and provides the roll index pulse (RIP), roll reference pulse (F_S) and sector pulses for the scientific instruments. The RIP pulse is used also for generating the roll oriented release signal for separation of the three Small Probes. Two units are provided for redundancy.

(b) Sun Sensor Assembly. The sun sensor assembly contains a mid-range sensor, an upper sensor and lower sensor to provide the required field-of-view. The output from a vertical slit (ψ pulse) provides an inertial reference and spin speed information. The output from a canted slit (ψ_2 pulse) provides indirectly sun pitch or aspect angle (sun elevation angle wrt the spacecraft +Z axis) information. Each sensor is redundant.

(c) Star Sensor Assembly. The star sensor is a command controlled unit which contains a vertical (ψ*) slit and canted (ψ_2*) slit either of which can be used to provide star pulses for an inertial reference or attitude determination.

(d) Nutation Damper. The nutation damper is a passive, self contained device which uses the wave action of liquid freon within a tube to absorb the nutational energy of the spacecraft.

3.3.1.3 Spacecraft Coordinates Definition. The coordinate system for use with the controls subsystem is defined in Section 3.1.

3.3.1.4 Equipment Locations. The control subsystem units are mounted on the spacecraft as shown in Figure 3.3.1-2 and Figure 3.3.1-3.

3.3.1.5 Attitude Sensors Geometry

3.3.1.5.1 Sun Sensor Field of Views. The mid-range and extended range sun sensor field-of-views provide coverage (ψ output) from 10° to 170° from the +Z axis. The ψ_2 output provides coverage from 15° to 165° from the +Z axis. The field-of-view of each sensor is shown in Figure 3.3.1-4. For the direction of spacecraft rotation indicated in Figure 3.3.1-4, the ψ sensor will see the sun slightly ahead of the ψ_2 sensor in time.

3.3.1.5.2 Star Sensor Field of View. The star sensor field-of-view (FOV) is 24 degrees in elevation with the center of the FOV located at 58.272 degrees from the +Z axis as shown in Figure 3.3.1-5. For the direction of spacecraft rotation indicated in Figure 3.3.1-5, the ψ* sensor slit will see a selected star slightly ahead of the ψ_2* sensor slit in time.

3.3.2 Units Description

3.3.2.1 Sun Sensors. Three redundant spin scan sun sensors mounted in each of two sensor assemblies are provided to meet the field-of-view requirements. Both sun sensor assemblies are located at the outer periphery of the shelf in order to have an unobstructed field-of-view. The midrange sun sensor assembly provides spin reference pulses (ψ) over a sun aspect angle range of 45 to 135 degrees from the vehicle spin axis and output pulses (ψ_2) for sun aspect angle measurements over a sun angle range of 55 to 125 degrees. The geometry of the midrange sun sensor is shown in Figure 3.3.2.1-1. The extended range sun sensor assembly contains two integral sensors. The upper sensor provides spin reference pulses over the range from 10 to 100 degrees and sun aspect angle pulses from 15 to 90 degrees. The lower sensor provides corresponding pulses over the range of 80 to 170 degrees and sun aspect angle pulses from 90 to 165 degrees, respectively. The geometry of the extended range sun sensors is shown in Figure 3.3.2.1-2.

One set of three sensors is connected to each ADP providing redundancy. The sensor assemblies are passive; there is no command or telemetry interface with them directly nor is any power provided. Electronic processing of the sensor outputs and the telemetered time occurrence of the pulses is performed by the ADP. This is more fully discussed in Section 3.3.2.3.

Only one of the three sensors can be selected by command at a time so that there is no ambiguity in pulse usage. The leading edge of the ADP detected sun pulse is the time reference (0.150

microampere threshold level of sensed sun signal). The boresight location is nominally at 352.5 degrees in the spacecraft coordinate system. The relationship of the sun sensor boresight to the sun pulse as detected by the ADP is shown in Figures 3.3.2.1-3 and -4. The ψ_2 pulse is used solely for sun look (aspect) angle telemetry measurements.

The variation of the sun pulse at Earth and at Venus is also shown in Figure 3.3.2.1-3. Shifts in the leading edge of the sun pulse at Venus can be compensated for in the calibration data of the sun pulse.

The relationship of the PSI and PSI2 outputs over the field-of-view provided by the mid-range and extended range sun sensors is shown in Figure 3.3.2.1-5.

3.3.2.2 Star Sensor

3.3.2.2.1 General Description. The star sensor is a spin scan sensor having a dual slit solid-state detector with a fan-shaped FOV. One slit, ψ^*, is aligned in a plane containing the spin axis and the second slit, ψ_2^*, is canted 20 degrees relative to the ψ^* slit. The direction of cant causes the time interval between the ψ^* and ψ_2^* to be the shortest when the star is closest to the +Z axis. The electronics of the ψ^* and ψ_2^* channels are completely independent; providing redundancy. Both channels can operate with either ADP.

The boresight of the star sensor is located 56.182 degrees from the spacecraft spin axis. The elevation FOV is 24 degrees centered about the boresight axis. This is shown in Figure 3.3.1-5. The azimuth FOV of each slit is 0.5 degree. The star sensor is mounted on the outer periphery of the shelf with the boresight at an azimuth location of 15 degrees from the spacecraft +X axis, measured toward the +Y axis in the spacecraft X-Y plane. A star passing through the center of the FOV will nominally traverse the ψ^* slit at an azimuth angle of 21.6°

and the ψ_2* at 15°. It is provided an unobstructed field-of-view.

The sensor is an integral unit having a single conical sunshade, a common optical system and an electronics housing assembly containing the two redundant electronic modules.

The star sensor is not damaged by sunlight entering the optics. The sun shade design employs a two-stage baffle system to limit the input of stray light. This permits stars to be detected that are 55 degrees or more from the sun and 37 degrees or more from an illuminated Venus, Earth, or moon. These angles include an electronic recovery time of 66 milliseconds (worst case).

A common optical system provides energy to the two photodetectors of the ψ* and ψ_2* channels.

A block diagram of one of the two identical electronics channels is shown in Figure 3.3.2.2-1. The basic function of the sensor is to output a position pulse from each operating channel for use by the ADP whenever the detected amplitude level exceeds the commanded threshold value, thus providing amplitude discrimination against background noise and dim stars. Angle discrimination for the position pulse is provided by the ADP. A star pulse from either channel can be selected as the on-board inertial reference in the ADP or supplied to telemetry for attitude determination. This is more fully discussed in Section 3.3.2.3.

3.3.2.2.2 Command Functions. The two star channels are individually controlled by redundant power on and off pulse commands from COMs 1 and 2, (STR19 or STRA9 for PSI* ON; STR1Ø or STRAØ for PSI* OFF; STR29 or STRB9 for PSI2* ON; STR2Ø or STRBØ for PSI2* OFF). The two channels are not operationally mutually exclusive and thus can be used simultaneously.

The commands control a mechanical relay which connects a fused bus line to the unit regulated power supply. A single redundant quantitative command (STRQ1 or STRQA) controls the desired individual threshold level and bandpass selection in each sensor channel. The quantitative command bit structure is as follows:

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40

Don't care bits

PSI2* bandpass select (bit 33)

PSI2* threshold select (bits 34–36)

PSI* bandpass select (bit 37)

PSI* threshold select (bits 38–40)

Bandpass selection: 0 = 19 Hz; 1 = 2 Hz

The threshold selection coding is shown in Reference: Paragraph 1.5.1. Calibration is listed in Appendix A.

The bandpass selection controls the low frequency cut-off to be at 19 Hz or 2 Hz to enhance star detection over the broad spin range. A summary of the pulse commands, command assignments and associated mnemonics is shown in Section 3.3.4. The present recommendation is that the 19 Hz bandpass be used for all spin rates, based on vendor tests (Reference: Paragraph 1.5.23).

3.3.2.2.3 <u>Telemetry Functions</u>. The star brightness measurement is telemetered directly from each star sensor channel as an 8-bit analog measurement. The measurement is the peak amplitude of any detected light source appearing within a spin angle gate provided by the ADP, and is reset by the leading edge of that spin angle gate when it opens again one spin period later. The spin angle gate is 11.25 degrees wide and can be positioned at any azimuth angle by command to the ADP.

The commanded threshold setting for each star sensor channel is also telemetered as an 8-bit analog measurement. The nominal expected voltage for each threshold setting is shown in Section 3.3.4.

The On/Off status and the bandpass selection status of each star channel are provided as bi-level status bits to the DIMs. A temperature sensor, located near the common star sensor detector chip is telemetered. This permits an assessment of the star sensor detection capability at a known telemetered temperature (effect of temperature on performance is discussed in Section 3.3.2.2-5 below). A summary of the star sensor telemetry is shown below.

STAR SENSOR TELEMETRY SUMMARY

TELEMETRY MEASUREMENT	TYPE	DESIGNATOR
PSI* Brightness	Analog	A*1BRM
PSI2* Brightness	Analog	A*2BRM
PSI* Threshold Setting	Analog	A*1THS
PSI2* Threshold Setting	Analog	A*2THS
Star Sensor Temperature	Analog	ASTART
PSI* ON/OFF Status	Bilevel	A*1ONS
PSI2* ON/OFF Status	Bilevel	A*2ONS
PSI* Bandpass State Status	Bilevel	A*1BPS
PSI2* Bandpass State Status	Bilevel	A*2BPS

3.3.2.2.4 Star Pulse Relationships. The position pulse is generated within the ADP when the peak amplitude of the star sensor video pulse exceeds the commanded threshold level. The leading edge of the position pulse occurs when the trailing edge of the video pulse falls to 25 percent of the peak amplitude. The relationship of the star sensor boresight axis and the position pulses, star sensor boresight axis, and PSI* optical axis is shown in Figures 3.3.2.2-2, 3.3.2.2-3A through 3.3.2.2-3D, 3.3.2.2-4A and 3.3.2.2-4B. These curves were generated from information contained in References 1.5.23 and 1.5.24.

3.3.2.2.5 Detection Performance. Star detection capability, as affected by temperature, selected channel, and selected threshold level, is seen in Figures 3.3.2.2-5A through 3.3.2.2-5G. Detected star brightness will vary as a function of the spacecraft spin rate. This effect is shown in Figure 3.3.2.2-6 and shall be compensated for in telemetry data processing using calibration curves provided with each flight unit. Likewise, there is a predictable and calibratable effect as a function of the command selectable bandpass. This effect is shown in Figures 3.3.2.2-7. elevation field-of-view is shown in Figure 3.3.2.2-8A and 3.3.2.2-8B. Reference: Paragraph 1.5.23 states that the maximum usable field-of-view of the star sensor is ±11.5°.

3.3.2.2.6 Summary of Biases for Sun Sensor and Star Sensor. Figure 3.3.2.2.6-1 shows a top view of the spacecraft coordinate system and azimuthal relationship between it, the sun sensor and the star sensor.

(This information is also contained in the oblique views shown in Figure 3.3.1-4 and Figure 3.3.1-5).

Table 3.3.2.2.6-1 and Table 3.3.2.2.6-2 list and generally explain the sun sensor biases and star sensor biases, respectively, that play a significant role in Attitude determination (Section 4.3.2) and Roll Reference Transfer (Section 4.3.3).

3.3.2.3 Attitude Data Processor (ADP)

3.3.2.3.1 General Description. Each of the two ADPs contains all the logic and signal processing to determine time intervals between sun and star sensor outputs for transmission to the ground station for computation of spacecraft attitude. For onboard functions, azimuth spin pulses are generated for spin period synchronization of star gates, reaction jet firing start signal, despin reference and science roll index pulse reference.

Each ADP interfaces with a dedicated set of three sun sensors. (The two ADPs and the two sets are not cross-strapped.) One of the three sun sensors is command selected based on the spacecraft spin axis to sun LOS pitch (aspect) angle. The sun aspect angle relative to the spin axis is determined by measuring the time interval between pulses ($\psi - \psi_2$). With the spin period time measurement ($\psi - \psi$), the aspect angle time measurement can then be quantized in terms of spin degrees. This then can be converted to obtain the resulting sun aspect angle.

Each ADP can operate with either channel of the star sensor. Normally, more than one star will appear in the FOV of each revolution, therefore angle and brightness discrimination is provided to obtain valid data from a known star. A commandable detection threshold setting in the star sensor will be used to provide amplitude discrimination against background noise and dim stars. A star gate, commandable in spin angle from the sun ψ pulse, will be used in the ADP to angle-discriminate against other bright stars. The star aspect angle relative to the spin axis is determined by the time difference of the ψ^* and ψ_2^* pulses. Time interval measurements from the sun ψ pulse to the star ψ^* pulse or ψ_2^* pulse are also made, thus permitting the use of only one star slit for attitude determination when favorable sun-star geometry exists.

Either the sun or star can be command selected for the onboard inertial reference. The selected input controls a phase lock loop (PLL) which then provides a quantization of the spacecraft spin period. The PLL is able to provide 2^{10} Fs sector pulses to the scientific instruments. A commandable delay from the inertial reference is provided by the Roll Index Pulse (RIP). A roll reference pulse is also provided by the PLL which is phase synchronous with the selected inertial reference.

The ADP also controls all the firing modes of the thrusters via command, and the operation of the latch valves.

3.3.2.3.2 Discrete Command Functions. Redundant discrete commands using COMs 5 and 6 control power application to each of the two redundant ADP units. The ON command only affects the designated unit and does not turn the other unit OFF. Thus both units can be on simultaneously. This feature is used for the initial spin up maneuver following launch vehicle separation to assure maximum success probability. It is imperative that the onboard command sequencer be programmed for this maneuver to only use COM 5 to turn on ADP No. 1 and only COM 6 to turn on ADP No. 2. These particular command lines are grounded through the separation switch and preclude the ADPs from being commanded on and executing the spin up maneuver due to any failure or spurious pulses that may affect the automatic sequence prior to spacecraft separation.

The ADP will initialize in a random manner at unit turn on except for the thruster control functions. These will initialize as follows:

CIRCUIT	STATUS
JCE Output Buffer	Disabled
JCE Magnitude Count	Random
Jet Fire Command Latch	Reset
Jet Interlock Command Latch	Reset

Additionally, system tests have shown that it may take up to 0.5 second after an ADP has been turned ON before the ADP will respond to a quantitative command.

A summary of the five ADP discrete commands and their mnemonics is as follows:

COMMAND DESCRIPTION	MNEMONIC	REMARKS
ADP 1 ON	ADP19	Not operative prior to separation
	ADPA9	
ADP 1 OFF	ADP1Ø	
	ADPAØ	
ADP 2 ON	ADP29	
	ADPB9	Not operative prior to separation
ADP 2 OFF	ADP2Ø	
	ADPBØ	
Latch Valve #1 Open	VAL19	
	VALA9	
Latch Valve #1 Closed	VAL1Ø	
	VALAØ	
Latch Valve #2 Open	VAL29	
	VALB9	
Latch Valve #2 Closed	VAL2Ø	
	VALBØ	

3.3.2.3.3 Quantitative Command Functions. A single but separate quantitative command "ADP configure," is provided to ADP #1 only via COM 5 (ATQØ1 through ATQ12), and ADP #2 only via COM 6 (ATQØA through ATQØL), i.e., there is NO command redundancy to each unit. The quantitative command is structured such that the first four LSBs of the available 16 bit quantity are dedicated for address identification to control 12 internal functions. An example of these functions and bit designations is shown in Table 3.3.2.3-1 for ATQØ1 (ADP#1) or ATQØA (ADP#2). Measurements A and B in the table are equivalent to measurements 1 and 2 respectively, in the MNEMONIC assignments of PC-455. The remaining command functions may be found in PC-455 (Reference: Paragraph 1.5.1). The structure of the quantitative commands will be discussed in detail in the following sections to facilitate use of the ADP.

The command interface is shown in Figure 3.3.2.3-1 (Appendix C).

3.3.2.3.4 Mode Control

3.3.2.3.4.1 Circuit Description. Operational control of the PLL is provided by use of the ADP Mode Select quantitative command that uses Address #2 (ATQØ3 or ATQØC of the ADP Configure Command). The logic processing diagram is shown in Figure 3.3.2.3-1.

The PLL is a variable frequency pulse generator which maintains both frequency and phase lock with the command selected input reference: sun, star, or simulated SRR. The prime element in the PLL is a voltage controlled oscillator (VCO) whose frequency is controlled by an analog compensation network. Under zero error conditions for the phase locked loop (i.e., phase and frequency lock with the input signals), the VCO will generate a frequency whose value counted down by either 2^{20}, 2^{21}, 2^{22}, or 2^{23} will provide an output pulse F_S essentially coincident with the input pulse. The different countdown chains are command selected ("PLL spin range select": ATQØ3; SR) to provide an operating range consistent with the spacecraft spin rate. For non-zero error conditions, the sensor input pulse and the output F_S are phase compared, and the resulting phase error forces the VCO frequency to be altered until zero phase error condition is achieved. In addition to generating the spin synchronous F_S output, the PLL also generates a synchronous high multiple of 2^{12} F_S and 2^{10} F_S (1024 sector pulses).

The output from the PLL is used to drive one fixed gate, gate-A, which is centered over the F_S pulse and two command variable (ATQØ5 or ATQØE for ACS Angle Delay Magnitude; ATQØ6 or ATQØF for Roll Index Delay Magnitude) delayed outputs. In addition, Gate A has two command selectable widths of 11.25 degrees and 45 degrees, both of which are centered over the F_S pulse. The 45 degrees width is only obtained when star acquisition is command selected ("Star

Acquisition/Normal:ST=N/A). The two variable delay generators provide output pulses at independently command controlled angle delays from the F_S reference. The desired delay is loaded into a 10-bit counter which is counted down by the 2^{10} F_S pulses.

One generator is used as the Roll Index Pulse (RIP) reference for use by the scientific instruments. The second generator is used to provide the desired roll angle delay for jet firing. This delay is also shared to locate a second gate, Gate B, which provides angle discrimination for star pulses.

Whenever the SRR pulse does not appear within Gate A, a "Missing SRR" pulse is generated at the end of Gate A. The generation of this pulse is not dependent upon the loss of lock enable command bit. The missing SRR pulse serves two functions outside the loss of lock logic; they are:

(a) To generate a "PLL Master Reset Pulse" in the PLL which is used to reset the phase detector, error integrator and PLL timing, so that the PLL frequency is not changed as a result of missing the SRR pulse.

(b) The pulses are accumulated for telemetry in a counter ("Missed SRR Count": AMISSC). This counter is reset each time it is read out to telemetry.

The SRR to control the PLL can be one of three inertial pulses (ψ, ψ*, ψ_2*) or the simulated SRR. The sun (ψ) is preferred because it provides an unambiguous reference. The initial sun acquisition is performed with the sun gate disabled. This allows the PLL to lock onto the sun wherever it is in the spin cycle. Once locked, the gated sun mode can be used and the loss of lock detection enabled. In the gated mode, the F_S pulse must occur within Gate A. If the pulse is missing for two or more consecutive revolutions, the PLL automatically transfers to the simulated SRR. When the sun pulse appears within Gate A for two consecutive revolutions, the reference automatically transfers back to the sun. Any star pulse from either the PSI* or PSI*2 channel can be selected for a reference. All star references are constrained to be used in the gated mode.

The simulated SRR utilizes a commandable spin period (ATQØ7 or ATQØG for MSBs; ATQØ8 or ATQØH for LSBs of PLL spin period magnitude) which can be set from 0.25 msec to 16.384 seconds in 0.25 msec increments. This reference can be selected by either a command (ATQØ3, REF = SIM) or the loss of lock logic (if enabled by command ATQØ3, PL=E) if two consecutive missing SRR pulses occur. The period should be set within 0.78 percent of the actual spin period (at 5 rpm) when operating with the loss of lock enabled. The accuracy is required to maintain the inertial location of Gate A so that the reference pulse will reappear in the gate after the two revolution criteria is satisfied to transfer back to the inertial reference.

A star advance mode ("SRR Advance": SRA = ADV) is provided which permits switching the SRR from sun to star, from star to star, from star to sun, from simulated SRR to sun, or from simulated SRR to star. When transferring to sun, the sun gate must be disabled. When transferring to star, the star gate (Gate B) must be located over the desired star. Upon execution of the SRR advance command, the PLL will reset the phase so that the new reference will occur within Gate A. The advance status reverts to normal status once a new SRR pulse is detected and the phase is reset.

A star acquisition mode (ST = N/A) is provided which allows locking to a star pulse in the absence of a sun reference.

Use of the mode control command is described in the following section.

3.3.2.3.4.2 <u>Command Usage</u> Mode control is accomplished using the ADP Mode Select command (ATQØ3 or

ATQØC). The use of each bit in the command is described below:

(a) SRR Advance (SRA = NRM/ADV) - The SRR (Selected Roll Reference) Advance is used to switch from one reference to another. Typically reference switching from sun to star, from star to star, from star to sun, from simulated SRR to sun, or from simulated SRR to star (Star Acquisition) is accomplished by this command bit. The telemetered "SRR Advance" bit (ADVANS) will change automatically from logical one (Advance State) to logical zero (Normal State) once a new SRR pulse is detected and the phase is reset.

Since the "SRR Advance" can effect various modes of transfer, the interdependence of this bit with other logic bits of (ATQØ3 or ATQØC) requires careful elaboration. Transfer of roll reference, such as sun to star or star to star, is accomplished by positioning Gate B over the desired star that is to be the next primary reference. After receipt of the "SRR Advance" CMD bit, the phase of the PLL is reset so that the desired star now appears in Gate A and becomes the reference for the PLL. The other additional modes of transfer can best be described in subsequent paragraphs in conjunction with other CMD bit usage.

(b) Enable/Disable Sun Gate (SUG = E/D) - The Enable/Disable Sun Gate accomplishes a two-fold purpose: a) for a logical 1 (enable), it allows the loss of reference detector ("missing SRR" counter) to telemeter (AMISSC) the number of missed sun pulses during the sample interval; i.e., between leading edges of successive occurrences of ADP Status Read envelopes (the interval time period depends on bit rate and selected TM format - refer to Paragraph 1.5.2 for details). If two or more consecutive missing sun pulses (not within Gate A) do occur and assuming "PLL loss of lock" is enabled, it allows SRR transfer to simulated SRR; b) for a logical 0 (disable), it allows normal sun acquisition.

(c) Star Acquisition/Normal (ST = N/A) - Star acquisition allows phase locking to a star pulse (likely used in the absence of a sun pulse). Normally, the simulated SRR signal will have been selected as a PLL reference and the commanded spin rate will be chosen as close to the actual spin rate as possible. However, some small frequency difference may exist which allows star gates A and B, which are now phase locked to the simulated SRR, to slip in phase with respect to the actual stars, making it impossible to keep a star within Gate B.

(d) SRR/Simulated SRR Select (part of REF = SUN/STR/SIM) - For a logical 1, the primary input to the phase lock loop will be the SRR (ψ, ψ* or ψ_2*), unless an automatic transfer is accomplished by two consecutive missing SRR pulses. In the latter case (PLL loss of lock enable = 1) the simulated SRR is selected. For a logical 0, the only input to the PLL is the simulated SRR irrespective of all other conditions. Note: Prelaunch tests on one Flight Model ADP have shown that the SRR to P_S delay at 4.7 rpm had changed from typically 2.6 milliseconds with use of a real SRR, to typically 4.85 milliseconds with use of the simulated SRR.

(e) Sun/Star Select (part of REF = SUN/STR/SIM) - Either the selected ψ (logical "1") or the Gated Star A (logical "0") is selected by the "sun/star select" (serial command bit). The resulting signal is the Selected Roll Reference (SRR).

(f) Sun Sensor Select (SRR = M/U/L/S) - Signals ψ and ψ_2 are received, each on twisted and shielded pair, from each of three sun sensors, midrange, extended range upper and extended range lower. These six signals are processed by six identical sun sensor electronics subassemblies. The outputs of these electronics are input to a digital multiplexer where one ψ and ψ_2 pair (both from the same sensor) are command selected for use by the ADP.

(g) Star Gate B; Channel 1/2 Select (SGB = 1/2) - One of the two star processor channels (ψ* or ψ_2*) within the ADP can be chosen by "star Gate B channel select" (logic 1 = ψ*). The selected star sensor channel would then be for: (a) telemetered time interval measurement (SRR to Gated Star B ATTM) between the SRR and that star that is the first chronologically in Gate B that exceeds the commanded threshold; (b) telemetered star brightness measurement (A*1BRM or A*2BRM) of the brightest star appearing in Gate B before the telemetry is sampled; and (c) use in the SRR Advance Logic for transferring the SRR to the first star appearing in Gate B that exceeds the CMDED threshold level.

(h) Star Gate A; Channel 1/2 Select (SGA = 1/2) - One output (ψ* or ψ_2*) from the two-channel star processor is chosen by "star Gate A channel select" (logic 1 = ψ*). The selected channel is gated by Gate A from the PLL to become "gated star A." Gated Star A is sent to the sun select/star select logic and the command and TM subunit for attitude measurements ("SRR to SRR," when the SRR is that star in Gate A).

(i) PLL Loss of Lock Enable/Inhibit (PL = E/D) - The "loss of lock" logic should more properly be labeled "loss of SRR" logic, as that is exactly what it detects.

The last commanded state of the "loss of lock" logic is telemetered as the bilevel signal" PLL Loss of Lock Enable/Inhibit" (ALOLES).
The actual state of the PLL w.r.t. lock on the SRR is telemetered as the bilevel signal "PLL Loss of Lock" (ALOCKS).

After receipt in the ADP of a command to enable or inhibit the logic circuitry, it may take up to one spin period for that commanded logic state to be established (commanded state is set by the occurrence of Gate A). Once enabled (logical "1"), this logic will detect and output a signal to the reference select logic, the JCE, and telemetry (ALOCKS will = 0) whenever two or more successive SRR pulses are found to be outside of Gate A. This loss of lock will cause automatic Roll Reference transfer to the simulated SRR, and inhibit the jets from being fired. If, while the PLL loss of lock logic is still enabled, and two or more successive SRR pulses were to reappear within Gate A, the "PLL loss of lock" signal would return automatically to its normal (in-lock) state. (The requirement for two or more successive reappearances prevents a single spurious noise pulse on the SRR line from reselecting the SRR as the PLL reference.
If inhibited (logical "0") by command, the "PLL Loss of Lock" signal (ALOLES) will indicate an apparent "in-lock" state (logical "1"), whether or not the PLL is truly locked on the SRR.

(j) PLL Spin Range Select (SR = 4-71) - PLL VCO output (nominal operating frequency) is divided down by a counter with variable length. The PLL Spin Range Select command bits select the appropriate variable divider output to give a total countdown chain length versus spin range as shown below:

COUNTDOWN CHAIN LENGTH	PLL OPERATING RANGE
$\div 2^{23}$	4.0 - 8.85 rpm
$\div 2^{22}$	8.0 - 17.7 rpm
$\div 2^{21}$	16.0 - 35.4 rpm
$\div 2^{20}$	32.0 - 70.8 rpm

These operating ranges have overlap designed to avoid dead zone regions and also to allow some flexibility with respect to the transmission of a new PLL spin range select command. Nominal range switching would be done only associated with substantial spin speed changes during which the PLL loss of lock would be inhibited, as it cannot track continuous burn spin speed changes. After such a change, the PLL would require approximately 70 revolutions minimum, as a rule of thumb, before it has settled and is ready for lock-on to an SRR.

3.3.2.3.5 Maneuver Control

3.3.2.3.5.1 Circuit Description. Attitude, velocity and spin speed control are provided through the use of a combination of axial and radial thrusters. Four radial, one forward and one aft axial thrusters are common to the Orbiter spacecraft and Multiprobe spacecraft.

The jet controlling logic functions are generated within each of the ADPs. However, only one driver module is provided for each thruster or latch valve. Each ADP contains half of the required drivers. Either ADP can operate any thruster even though the driver is physically located in the other ADP. Each driver is connected to the spacecraft essential bus and separately protected using 5 amp fuses. There is no need to activate the drivers by command; they are permanently activated. The logic diagram of the thruster control functions is shown in Figure 3.3.2.3-1.

The drivers, thrusters and latch valves are interconnected so that the ADP that controls the latch valve also controls all the thrusters that are supplied hydrazine by that latch valve. A complete loss of an ADP will still permit maneuvers to be performed.

In the continuous firing mode the firing duration is controlled by use of the JET countdown quantitative command that uses Address #3 (ATQØ4 or ATQØD of the ADP configure command) which permits up to 2096.64 seconds in 0.512 second increments.

The pulsed firing mode has two commandable pulse widths, 128 ms and 512 ms. The roll oriented firing location is selected by using the ACS variable delay output from the roll reference generator (via ACS Angle Delay quantitative command that uses Address #4 (ATQØ5 or ATQØE of the ADP configure command)). The occurrence of the delayed output initiates the start of the fire pulse. In the command selected normal mode, a fire pulse will be generated every revolution. In the alternate fire mode (2 pulse firings per spacecraft revolution), a second fire pulse will be generated 180 azimuthal degrees with respect to the first pulse. The duration of the pulsed firing mode can be controlled by pulse count or time duration. With pulse count selected, the magnitude command converts to a pulse counter which then permits up to 4095 pulses in one pulse increments.

After the firing logic has been selected, the actual firing is initiated by commanding the interlock function (via JET Fire Interlock quantitative command that uses Address #10 (ATQ11 or ATQØK of the ADP configure command) followed by the fire command (via JET Fire quantitative command that uses Address #11 (ATQ12 or ATQØL of the ADP configure command). There is no time limitation after the interlock command but the command sequence should group these two commands together. The firing sequence will terminate when the magnitude decrements to zero count. The JCE output buffers are automatically commanded off and the magnitude counter will underflow, terminating with a count of 4095 (indicated in telemetry by "JCE Countdown" AJEAGC - as twelve "ones").

A duty cycle detector is activated during all pulsed firing modes which monitors the JCE buffer output to the solenoid drivers. If the duty cycle exceeds the limits shown in Figure 3.3.2.3-2, firing will be terminated. The detection point is a function of spin speed and will exclude the normal operation of the wide pulse mode above 20 rpm. This feature cannot be disabled by command.

One other protective feature is provided for use with the pulse mode. Since the roll angle delay requires the PLL to be locked for accurate firing, a PLL loss of lock detection can be enabled which will terminate any maneuver. To use this capability, the PLL loss of lock is enabled and the PLL reference should be operated in the narrow gate mode. If two successive SBR pulses occur outside Gate A indicating the PLL is not locked, the out of lock detector will terminate the sequence. This feature should be disabled for spin speed maneuvers since the PLL cannot track large spin speed changes and will terminate the spin sequence if loss of lock is detected.

3.3.2.3.5.2 <u>Command Usage</u>. Operational use of the thrusters is accomplished by use of the commands described below:

(a) Jet Control (Quantitative command that uses Address #1 (ATQØ2 or ATQØE of ADP configure command)).

(1) Jet Select (R1, R2, R3, R4, A5, or A6; each = E/D) - six bits of the magnitude data are dedicated to selecting any set of four radial and two axial thrusters, by commanding a logical "1" state for each thruster intended to be active. Any or all thrusters can be selected; however, during normal mission operations, a

maximum of two thrusters will normally be selected. PC-455 (Refer to Paragraph 1.5.1) shows the specific magnitude data bit associated with a given thruster. Possible thruster options per maneuvering function is specified below:

FUNCTION	SELECTED FIRING MODE	THRUSTER SELECTION
Spin Speed Change	Continuous or Pulse	2R+4R (Spin-down) 1R+3R (Spin-up)
Velocity Change	Continuous	5A+6A
	Pulse	1R•2R+3R•4R
	Pulse (Twice Spin Frequency)	5A+6A
Attitude Change	Pulse	5A+6A+5A•6A 1R•4R+2R•3R

(2) Spin Rate Detector Enable/Inhibit (SRD = E/D) - Onboard detection is provided which will terminate any firing sequence when specific limits are detected and a logical 1 is commanded for this magnitude bit. A spin rate detector measures the spin period and will terminate the maneuver if the spin frequency is less than 3.7 rpm or over 67 rpm. This capability is provided by a special counter which measures the time interval between two successive SRR pulses using a 15.625 Hz clock rate. A failure is indicated if the detected count is less than 14 or greater than 254. If the gated reference mode of the PLL is being used during the first firing sequence, (i.e., "loss of lock" logic is enabled) firing will be terminated if the SRR moves outside Gate A. The loss of SRR pulses will also be detected as a low rpm condition. This detection capability can be disabled by command (logical "0") but this should only be

done for the initial spinup maneuver so that the spin rate can be set above the lower threshold.

(3) Continuous/Pulse Fire Select (F = P/T) - The thrusters will be fired in two modes of operation, namely: a) continuous (logical "1") and, b) pulse. In the continuous mode the thrusters are fired continuously until terminated by a proper completion of the count sequence or automatic override features previously discussed.

(4) Pulse/Time Count (C = P/T) - The jet countdown magnitude (contained in quantitative command that uses Address #3 (ATQØ4 or ATQØD of ADP configure command)) initializes a 12 bit counter (AJMAGC), that is subsequently counted down to zero based upon the logical state of this bit. The countdown counter will count elapsed time when either (or both) the continuous fire mode or the time count mode is selected, and counts pulses when both the pulse fire mode and the pulse count mode are selected. (If the alternate mode is selected and pulses are to be counted, each of the two pulses per revolution is counted). Thus, if both the pulse fire and time count modes are commanded the pulse duration of the firings is still controlled by the pulse width selection signal. The countdown counter counts elapsed time from the start of the first pulsed firing. In pulse mode, the firing pulse can be commanded to occur at any spin azimuth point using the ACS angle delay command.

(5) Pulse Width Select: (512/128 ms; P = 2/8 respectively) - The firing signal to the solenoid drivers in the pulse mode of operation will either be 512 milliseconds (logical "1") or 128 milliseconds (logical "0").

(6) Normal/Alternate Fire Mode (FM = N/A) - In the normal fire mode (logical "1") a single fire pulse will be generated every spin period at the commanded angle delay. In the alternate fire mode (logical "0") a second firing pulse is generated 180 degrees after the first pulse in each revolution. The alternate fire mode provides the capability to translate the spacecraft axially in a vernier increment, i.e., pulsed, while keeping nutation at a small level. The first pulse firing to occur after the Alternate Fire Mode is selected will correspond to the first occurrence of the SRR pulse (0° azimuth angle), even if Mode selection occurs on-board before occurrence of the 180° azimuth angle event.

(b) Jet Countdown (Quantitative command that uses Address #3 (ATQØ4 or ATQØD of ADP configure command)). Twelve bits of magnitude represent 4095 pulses or 2096.64 seconds maximum count. In the time count mode, a clock frequency of 1.953 Hz or 0.512 seconds per bit is used to countdown the elapsed time as represented by the contents of the 12-bit jet countdown register, that has been loaded by command. In the pulse mode, the same register is used, and decrements for each fire pulse generated.

(c) ACS Angle Delay Magnitude (Quantitative command that uses Address #4 (ATQØ5 or ATQØB of ADP configure command)). The JCE jet fire angle is determined by the ACS delay generator. The ACS delay pulse is generated by a 10-bit (360°) delay counter. The command is the binary magnitude of the desired delay with a quantization of 0.351 degree per bit. This counter is initialized to the commanded "ACS delay" angle by the PLL output signal F_S. The counter is then counted down by the "2^{10} F_S" clock pulse until it underflows. The counter underflow is the "ACS delayed pulse," and initiates thruster firing in the pulse mode.

The ACS delay generator is shared with the star gate B function. The angle delay locates the starting edge of Gate B (Gate A is centered on P_S). The ACS delay quantity has a one count bias, i.e., the actual delay is one count greater than the commanded value. Therefore, to accomplish a zero degree angle delay, a delay of 360° (ten "ones") must be commanded into the countdown register.

(d) JCE Buffer Output Enable (Quantitative command that uses Address #8 (ATQØ9 or ATQØI of ADP configure command)). The +5 volt and the +15 volt to the JCE logic circuitry is controlled by the ADP power on/off discrete command. However, an additional line switch provides command control of the +15 volt bus to the JCE output buffers. The line switch is latched on (and indicated by AJCE1S or AJCE2S, corresponding respectively to the use of ADP1 or ADP 2), whenever the "JCE buffer output enable" command is sent. The latch is reset to logical "0" and the switch turned off whenever any of the six disable conditions listed below occurs:

(1) Countdown counter (AJMAGC) reaches zero count and underflows (normal operation)

(2) ADP initialization (insures that the JCE buffer output is off at ADP power turn-on)

(3) Receipt of the "JCE buffer output disable" cmd (see item (e) ahead)

(4) "PLL Loss of Lock" condition when enabled

(5) "Over/under spin rate" condition when spin rate detector is enabled

(6) "Duty cycle failure" condition when operating in pulse firing mode

(e) JCE Buffer Output Disable (quantitative command that uses Address #9 (ATQ1Ø or ATQØJ of ADP configure command)). This command will terminate any thruster firing in progress irrespective of the state of the countdown counter or the onboard JCE failure protection. The command disables the JCE output buffers and will reset the fire and interlock logic. The counter will indicate the incompleted count of the maneuver.

(f) Jet Fire Interlock (Quantitative command that uses Address #10 (ATQ11 or ATQØK of ADP configure command)).

The Jet Fire interlock Command controls the state of the JCE Fire Enable/Disable logic. This Enable/Disable logic does not allow the jets to fire (as indicated by AJCEFS - JCE Fire Enable Status remaining a logic "0") until both the jet fire interlock command and the jet fire command have been received in that order without an intervening JCE Buffer Output Disable command. The interlock command sets a flip flop which then permits another flip flop to be set when the fire command is received. Should the JCE Buffer Output Disable command be received after the jet fire interlock command but before the jet fire command, both flip flops will be reset and the jet fire interlock command must be transmitted again followed by the jet fire command before the jets will begin to fire. There is no time lapse requirement between the jet fire and jet fire interlock command.

(g) Jet Fire (Quantitative command that uses Address #11 (ATQ12 or ATQØL of ADP configure command)). This CMD starts the thruster firing sequence if proper initialization described above is true.

3.3.2.3.6 Attitude Measurements

3.3.2.3.6.1 Circuit Description. A logic diagram of the attitude measurement function is shown in Figure 3.3.2.3-1.

A total of six different time separation measurements may be computed, however, only two may be selected at a time, in any combination from the table below:

SELECTABLE MEASUREMENT (ATTM)	ADDRESS		
SRR - SRR	0	0	0
SRR - Minor Frame Start Signal	1	0	0
SRR - RIP (Roll Index Pulse)	0	1	0
SRR - MIP (Invalid on Multiprobe)	1	1	0
SRR - Gated Star B	0	0	1
SRR - PSI2	1	1	1
SRR - Major Frame Start Signal	0	1	1

The signal SRR (selected Roll Reference) may be command selected to be: SUN (ψ), Star Channel 1 (ψ*), Star Channel 2 (ψ_2*), or Simulated SRR, while operating in the so-called "free-run" mode. Note that for measurement number 5, Gated Star B may be either ψ* or ψ_2*.

The actual measurements are computed in binary counters using a clock frequency of 4 kHz, thus giving a measurement resolution of 0.25 milliseconds. After the measurement is completed, the counter is frozen and cannot be reset by the SRR until the 16-bit magnitude along with the measurement identification, A or B, (equivalent to 1 or 2 respectively in the mnemonic assignments of Reference: Paragraph 1.5.1) and the command selected addresses are parallel transferred into the serial TM registers at the leading edge of the first of the three readout envelopes. This clears the counter and enables the start of a new measurement. The 24 bits of data in the TM register is then shifted out serially least significant bit first in three groups of 8 bits. The readout format is shown in Table 3.3.2.3-2. This same measurement will be continued to be read out until the on-going measurement is complete and the transfer and readout process is repeated.

A summary of the attitude measurement processing is as follows:

(a) The selected roll reference pulse generates a "start" pulse which:
   (1) Resets the ATTM counter to zero
   (2) Enables the 4 kHz clock pulse to the ATTM counter
(b) The selected "stop" pulse (SRR, Minor Frame Start, Major Frame Start, RIP, MIP, Gated Star B, or ψ_2):
   (1) Disables the 4 kHz clock pulse to the ATTM counter
   (2) Sets a "measurement start/stop control" flip-flop which freezes the ATTM counter until the telemetry subsystem is ready to read the measurement
(c) The first of the three ATTM read envelopes will:
   (1) Transfer the measurement, if complete, to the TM register
   (2) Enable the start of new measurement at the next SRR
   (3) Start the TM readout process

The synchronization of the attitude measurements with telemetry will insure that information will only be loaded into the TM register when the previous measurement has been completely read out at least once. If no stop pulse is detected before a second selected roll reference pulse is generated, the counter will be reset and the same measurement will be started again.

The serial TM register shall be shifted out during 3 TM read envelopes; 3 subcommutation envelopes in the subcommutation mode (S MODE) or 3 minor frame envelopes in the minor frame mode (M MODE). In each mode, the envelopes are counted modulo 3 and the data are output to two separate DIMs providing redundancy.

If a new measurement is not ready when the first envelope of the next group arrives, the old measurement is shifted out again (recirculated). The repeated measurement word can be identified by Sel A/B (equivalent to 1/2 respectively), bit; Sel A/B is toggled as each new measurement is loaded and the previous measurement is read out in TM. The state of this bit remains unchanged if a measurement is repeated.

3.3.2.3.6.2 Data Valid Determination. The attitude measurements can be telemetered in either the subcom(s) mode or in a minor frame format (M mode). If data is being telemetered in a minor frame format, the subcom data is invalid. The onboard processing is as follows:

S mode is arbitrarily assumed at the start of each major frame. M mode is then entered and S mode is inhibited if a single minor frame envelope is detected. If M mode is envoked, subcom envelopes will be ignored for the remainder of the major frame. If M mode is not entered, the subcom data will remain valid. A summary of this process is as follows:

(a) If no minor frame envelopes occur, the data in the subcom words will be valid.
(b) If subcom envelopes and minor frame envelopes both occur within the same minor frame, the minor frame data will be valid and the subcom data will be all logical zero's except for the first subcom word in each major frame.
(c) When switching from the subcom only sampling format to the minor frame sampling format, the minor frame data may be out of sequence until the next measurement is completed.
(d) When switching from the minor frame sampling format to the subcom only format, the subcom data will be all zero's until the start of the next major frame, or until the next measurement is completed, whichever occurs last. If the new measurement occurs last, the subcom words following

the first major frame pulse may be randomly permutated until the new measurement is completed.

(e) If any of the selected measurement pulses disappear, the ATTM will "hang up" and the previous measurement will be repeated.

3.3.2.3.6.3 Unique Attitude Measurements. Several types of measurements are somewhat unique and bear mentioning.

(a) The SRR-Minor Frame Start or Major Frame Start measurement. Whenever the spacecraft rotor rate is higher than the minor frame rate, two or more SRR pulses may be expected between two minor frame rate pulses. The ATTM logic is implemented to calculate the smallest time separation between these two pulses. This is accomplished by the logic automatically resetting the ATTM counter whenever an SRR pulse occurs, regardless if a measurement is in progress or not.

(b) Whenever an inertial reference is lost and the Simulated SRR becomes the SRR, all measurements will be computed with the Simulated SRR acting as the "start" pulse. This insures that such vital science data as SRR-RIP signal, or SRR-Major Frame Start signal will not be lost. The drift rate between the Simulated SRR and the actual rotor rate can be checked by (SRR-ψ_2) or (SRR-Gated Star B) telemetry measurement.

3.3.2.3.7 Roll Index Delay Magnitude (Quantitative Command that Uses Address #5 (ATØØ6 or ATØØF of ADP Configure Command)). The Roll Index Pulse (RIP) is generated by a 10-bit (0° to 359.65°) counter in exactly the same manner as is the ACS delayed pulse. The counter is initialized to the commanded roll index delay magnitude by F_S. The command is the binary magnitude of the desired delay with a quantization of 0.351 degree per bit. The RIP is transmitted to science, to the command and telemetry subunit for attitude measurements and to the Small Probes release interface circuit.

Unlike the ACS delay, the actual RIP delay is equal to the number of magnitude counts commanded.

3.3.2.3.8 <u>PLL Spin Period Magnitude - MSB (Quantitative Command that Uses Address #6 (AT007 or AT00G of ADP Configure Command))</u>. Eight bits of this magnitude command are used to control the eight most significant bits of a 16-bit counter that control the simulated SRR generator. The simulated SRR generator outputs pulses at a period proportional to the PLL Spin Period Magnitude serial commands. The 16-bit command words are loaded into a counter which is counted down at 4 kHz. Upon reaching zero, the counter outputs a pulse and reloads itself (in the "free-run" mode) with the 16-bit command word and continues to count down).

The output frequency of this generator can be made nearly equal to the actual rotor spin rate within 1/2 of the 0.25 msec resolution. The phase of the output signal is made equal to the PLL output, F_S, by using F_S to load the counter prior to manual or automatic switching to the simulated SRR as the PLL reference. Thus, negligible phase transients are introduced to the PLL when switching to Simulated SRR as a PLL reference.

The generator is allowed to "free run" (not being loaded by F_S but by the Simulated SRR pulse itself) whenever the Simulated SRR is used as the PLL reference, which occurs whenever a "Loss of Lock" condition exists or whenever so selected by the SRR/Simulated SRR select command bit.

3.3.2.3.9 <u>PLL Spin Period Magnitude - LSB (Quantitative Command that Uses Address #7 (AT008 or AT00H of ADP Configure Command))</u>. Eight bits of this magnitude CMD are used to control the eight least significant bits of a 16-bit counter described in the preceding paragraph.

3.3.2.3.10 ADP Status Measurements

3.3.2.3.10.1 Circuit Description. There are 79 bits of status information within the ADP that is formatted into ten serial digital TM words, consisting of 8 bits per word. Most of the ADP configure commands (ATQØ1 through ATQ12) status bits are telemetered as part of these ten words, allowing for command verification. In addition, the following data is contained within these ten TM words:

(a) Small Probe Release Status (1 bit)
(b) Missed SRR Count (4 bits)
(c) PLL Loss of Lock Status (1 bit)

The data format for each of these words is contained in PC-454 (Reference: Paragraph 1.5.2).

The ADP Status TM can be read out in either a subcom mode or in a minor frame mode. In the subcom mode, the ten words are read out one word per minor frame for ten consecutive minor frames. In the ACS format, or minor frame mode, the words are read out in ten consecutive word slots, all ten words being read twice per minor frame.

The interface with the telemetry subsystem is done by way of a three signal interface. The ADP receives a TM read envelope and TM read clock and outputs ADP status data serially, one bit per TM read clock, 8 bits per read envelopes. The ADP has redundant output buffers to each DIM which precludes a single failure in one DIM from destroying the data read into the redundant DIM.

3.3.2.3.10.2 ADP Status Data Valid Determination. The ADP status data can be telemetered in either the subcom mode or in the minor frame format. If data is being telemetered in the minor frame format the subcom data is invalid. The onboard process is as follows:

A unique read envelope is sent to the ADP for each of the two formats. This allows the ADP to determine which format has been selected and output the data on a single serial interface. A summary of this process is as follows:

(a) If no minor frame envelopes occur, the data in the subcom words will be valid.
(b) If subcom envelopes and minor frame envelopes both occur within the same minor frame:
   (1) The data appearing in the subcom words will be word 1 of the ten word sequence.
   (2) The data in the minor frame words will be valid.
(c) When switching from the subcom only sampling format to the minor frame sampling format, the minor frame data will be valid within the first complete minor frame.
(d) When switching from the minor frame sampling format to the subcom only format, the data will be valid within the first complete major frame.

3.3.2.3.11 Bilevel Telemetry. Each ADP provides two bilevel bits directly to separate DIM channels as follows:

TM DESCRIPTION	MNEMONIC
ADP No. 1 ON/OFF Status	AADP1S
ADP No. 2 ON/OFF Status	AADP2S
JCE No. 1 Buffer Output Status	AJCE1S
JCE No. 2 Buffer Output Status	AJCE2S

3.3.2.4 Nutation Damper. The nutation damper on each spacecraft is a passive, sealed straight tube of liquid freon E3.

Wave action in the peripheral surface of the liquid absorbes the nutational energy of the spacecraft, assuring positive stability of the vehicle. The damper is required to reduce vehicle nutation to 1/3 ($\approx$ 37 percent) of the initial value in a time interval of ≤60 minutes for all phases of the mission.

A fill fraction of 25 percent of the total cavity volume is employed, based upon results of optimization studies.

The damper longitudinal axis is mounted 46.5 ±1.0 inches radially outboard from and parallel to the vehicle spin axis.

Liquid Freon contained within the damper is free to move at all times. Its boiling point of +306°F and freeze point of -160°F are well outside the qualification temperature range of 138°F and 0°F. Because of favorable inertia ratios during probes separation, it will not produce unfavorable dedamping or amplification of nutational motion. After spinup, centrifugal force acts on the liquid Freon forcing it to spread out along the tube longitudinal axis. During spacecraft nutational motions, free surface waves are propagated in the liquid Freon. Energy of nutation is dissipated by means of viscosity friction effects of these surface waves relative to the damper tube wall.

The performance of the damper is dependent on spin rate, temperature and the mass properties of the vehicle. The performance of the damper at specific phases of the mission is summarized in Table 3.3.2.4-1.

3.3.2.5 Engineering Instrumentation

3.3.2.5.1 Temperature Monitoring. A temperature sensor for the control subsystem, as shown in the table below, provides thermal information directly to DIMs. The measurement is only sampled in the subcom format and there is no redundancy.

DESCRIPTION	LOCATION	MNEMONIC
Star Sensor Temperature	Near the detector chip	ASTART

3.3.3 Operational Description

3.3.3.1 Sun Acquisition. Sun acquisition occurs when the PLL in the ADP obtains a frequency and phase such that the F_S output pulse is in lock with the detected sun pulse. Since the sun can be detected unambiguously, acquisition can be accomplished using the ungated (360 degree gate) sun mode. However, certain command information must be supplied to the ADP to achieve and verify a successful acquisition. After the ADP "ON" command is generated (ADP19 or ADPA9 for ADP No. 1; ADP29 or ADPB9 for ADP No. 2), then the ADP must be configured via the quantitative commands in the following manner (Measurements A and B correspond to measurements 1 and 2 respectively of the Mnemonic listing contained in Reference: Paragraph 1.5.1):

(a) Measurement Select (ATQØ1 or ATQØA)

COMMAND BIT #:	25 thru 28	29 thru 34	35 thru 37	38 thru 40
	$\underbrace{0000}$	$\underbrace{XXXXXX}$	$\underbrace{010}$	$\underbrace{010}$
	Address	Don't care	Measurement A; SRR to RIP	Measurement B; SRR to RIP

(b) Jet Control (ATQØ2 or ATQØB)

```
                                1000  0000000  1 0 0 0 1
Address ------------------------'        |     | | | | |
No jets selected ------------------------'     | | | | |
Normal fire mode selected ---------------------' | | | |
128 ms pulse width selected ---------------------' | | |
Time count selected -------------------------------' | |
Pulse mode selected ---------------------------------' |
Spin rate detector enabled ----------------------------'
(assumes spin rate >3.7 rpm)
```

(c) ADP Mode Select (ATQØ3 or ATQØC)

```
0100  10  0  X  X  XY  1  1  0  0  0
```

Address

8.0 to 17.7 rpm spin range

PLL Loss of Lock Enable/Inhibit (Inhibited)

Star Gate A channel select (don't care)

Star Gate B channel Select (don't care)

Sun Sensor Select (attitude dependent)

Sun/Star Select (Sun selected)

SRR/Simulated SRR Select (SRR Selected)

Star Acquisition/Normal (Normal Selected)

Enable/Disable Sun Gate (Disable Selected)

SRR Advance (Normal Selected)

(d) Roll Index Delay Magnitude (ATQØ6 or ATQØF)

```
1010  0000000000  XX
```

Address

Zero Angle Delay

Don't Care

The "SRR to RIP" measurement is selected to verify normal Phase Lock Loop Acquisition. At ADP power turn-on, the PLL is initialized to the center of one of the randomly selected spin ranges; i.e., 6.425 rpm, 12.85 rpm, 25.7 rpm or 51.4 rpm. Proper initialization by the ADP Mode Select Command above is necessary so that the PLL can slew in the correct frequency range using the sun as a reference until the PLL rate is identical with the spacecraft rotor rate. The only time the PLL will not slew in frequency is if the PLL is in phase and frequency lock with the SRR. Therefore, by selecting the RIP angle to be zero, the attitude measurement SRR to RIP will provide a measurement of the PLL phase bias from ADP input leading edge detection (SRR pulse) to F_S (PLL overflow pulse). This is identified as SRR to F_S bias in Figure 3.3.2.1-3. (Any roll index delay can be used if this phase bias is not to be measured.) During sun acquisition, the measurement of SRR to RIP will behave like a damped sine wave (see Figure 3.3.3.1-1). Once the PLL is in lock, the measurement provides the steady state phase bias between the SRR and the PLL output for attitude determination update. The steady state bias is not expected to exceed ±0.25°.

A time history of representative phase lock loop acquisition for 5 rpm and 15 rpm (Figures 3.3.3.1-1 and 3.3.3.1-2) shows total acquisition time of 450 seconds and 76 seconds, respectively. These figures were derived from a computer simulation that excluded any effect of nutation and wobble.

The following time interval measurements are obtained upstream from the phase lock loop, and their validity is therefore unaffected by the state of the phase lock loop: (ψ-ψ, ψ-ψ_2, ψ-Major Frame Start, and ψ-Minor Frame Start).

Proper telemetry indication after Sun Acquisition is as listed in Table 3.3.3.2-1 in the next section.

3.3.3.2 ADP Normal Operating Mode. The normal operating mode will be considered to be sun reference while spinning at 15 rpm for the transit phase. The sun acquisition operation discussed in the preceding paragraph is considered to have been accomplished at this point.

The initial conditions for transferring to the normal operating mode are assumed to be as follows:

(a) Spacecraft spinning at 15 rpm.
(b) ADP is ON and the PLL is in lock with the sun as reference.

The jet control logic should have been commanded to the safest configuration; i.e., no jets selected with the spin rate detector enabled during sun acquisition (see 3.3.3.1).

Telemetry indications for these initial conditions should indicate as shown in Table 3.3.3.2-1.

The jet countdown magnitude (ATQØ4 or ATQØD) should be commanded to a zero count. (The initial spin-up maneuver would have terminated with a maximum count in the count register.) Verify the JCE buffer output power (AJCE1S or AJCE2S) is off, or disable the output by command (ATQ1Ø or ATQØJ).

The ATTM measurements (ATQØ1 or ATQØA) should be command selected to be:

(a) SRR to SRR
(b) SRR to PSI2.

This will provide a measurement of the actual spin period and the sun aspect angle. These two measurements provide a quick look of the spacecraft inertial relationship. After the average spin period has been determined, the simulated spin period of the same magnitude should be commanded ((ATQØ7 and ATQØ8) or (ATQØG and ATQØH)). It is recommended that the spin period be continuously telemetered throughout the mission if other attitude measurements are not being made. The simulated spin period then should be updated to the nearest integral value (25 millisec, resolution) when command opportunities exist. A one-bit error will permit operation in the free-run mode for 16-minutes at 15 rpm without the inertial reference moving outside of Gate A.

After the correct simulated spin period has been commanded, the ADP should be commanded to operate in the gated sun mode with the PLL loss of lock enabled. This will permit transfer to the simulated roll reference if two consecutive sun pulses are missed for any reason. The ADP mode command to select this configuration is as follows:

ATØØ3 or ATØØC:

COMMAND BIT #:	25 thru 28	29 & 30	31	32	33	34 & 35	36	37	38	39	40
	0100	10	1	1	1	XY	1	1	0	1	0

Address (bits 25 thru 28)

8.0 to 17.7 rpm Spin Range (bits 29 & 30)

PLL Loss of Lock Enable/Inhibit (Enable Selected) (bit 31)

Star Gate A Channel Select (Star 1 Selected) (bit 32)

Star Gate B Channel Select (Star 1 Selected) (bit 33)

Sun Sensor Select (Attitude dependent) (bits 34 & 35)

Sun/Star Select (Sun Selected) (bit 36)

SRR/Simulated SRR Select (SRR Selected) (bit 37)

Star Acquisition/Normal (Normal Selected) (bit 38)

Enable/Disable Sun Gate (Enable Selected) (bit 39)

SRR Advance (Normal Selected) (bit 40)

3.3.3.3 Star Sensor Control Including Star Acquisition

3.3.3.3.1 Channel Control. Each channel of the star sensor is individually controlled by ON and OFF discrete commands. Both channels will normally be commanded ON for attitude determination measurements.

Typical telemetry response for the star sensor with both channels operating is shown in Table 3.3.3.3-1.

3.3.3.3.2 Star Detection Capability. The star sensor has capability of detecting (one at a time) any one of 25 stars or the planet Saturn within the celestial sphere, dependent upon the spacecraft attitude. The detection capability at any point in time will vary with spacecraft spin rate, star elevation within the FOV, sensor temperature, and the selected bandwidth within the sensor. These effects are presented in 3.3.2.2 and are to be used to correct the telemetered star brightness value. The sequence for star brightness calibration are listed in Table 3.3.3.3-2. Temperature affects the probability of detection for any given star, but does not affect the brightness to TM volts relationship significantly.

The brightest stars are listed in Table 3.3.3.3-3 in decreasing order of brightness index ("silicon magnitude" as determined by the star sensor silicon detector spectral response and tabulated in Reference: Paragraph 1.5.23) with their positions expressed in terms of Earth Centered Ecliptic coordinates. In-flight calibration shall be used to correct the list to remove intensity errors created by a non-perfect star standard during test.

The minimum star brightness levels (as telemetered) required for time interval measurements versus selected parameters, are listed in Table 3.3.3.3-5.

3.3.3.3.3 Star Brightness Measurements. Star brightness measurements are telemetered directly from each channel of the star sensor, A*1BRM for the PSI* channel and A*2BRM for the PSI2* channel. To measure the brightness of a given star, the azimuth position of Gate B must be commanded via the ADP (ATQØ5 for ADP No. 1; ATQØE for ADP No. 2) to encompass the star location within the 11.25 degree gate. Formulation of the proper command bit structure to position Gate B properly is discussed in 3.3.2.3.5.2. The commanded angle delay is the leading edge of Gate B with respect to the occurrence of the SRR pulse. This same Gate B is used within the ADP to make a time interval measurement to the star pulse occurring within Gate B.

Important distinctions between the brightness measurement and the time interval measurement should be understood.

For the sake of understanding the brightness measurement, the opening of Gate B can be interpreted in terms of time occurrence as well as azimuthal location. The commanded azimuthal position for Gate B is translated by the Controls S/S into a time delay w.r.t. the SRR occurrence, for the opening of Gate B. At the instant that Gate B begins to open, the star <u>brightness detector</u> resets, and begins to look for the peak signal that occurs <u>while</u> Gate B remains open (for the time period equivalent to 11.25° azimuthal angle). At the instant that Gate B <u>closes</u>, the brightness detector stops looking for any signal, and instead, holds the peak signal in storage for nearly one spin period, when Gate B begins to open again, and the brightness detector to reset again.

The telemetered value will be the peak star brightness detected during the last occurrence of Gate B if the telemetry system samples the measurement between gate occurrences. If the telemetry system samples the measurement during the gate occurrence, the telemetered value will be the peak value obtained up to the time of telemetry sampling occurrence. For example, if the telemetry sampling time occurs instantly <u>after</u> the opening of Gate B, but <u>before</u> a star is <u>brightness</u> detected azimuthly later in Gate B, the telemetered value for brightness would be the background level (approximately 120 mv. or less). As a second example, if the TM sampling time occurred between two stars that are within Gate B, the telemetered brightness value will correspond to that star that has already been brightness detected, even though the star yet to be detected may be brighter.

The brightness measurement in the star sensor is independent of the commanded threshold value. For instance, the commanded threshold value may be insensitive to a star within Gate B, thus precluding a time interval measurement w.r.t. the SRR occurance, but the star's brightness will be telemetered (if the TM sampling interval occurs after the star has been brightness detected).

Since only one star channel may be assigned to Gate B at any instant, only that star channel's brightness detector is usable to produce valid telemetry. The other star channel's brightness detector will produce meaningless telemetry.

It follows also that, in the ADP, only one star channel can be selected at a time for the Gated B star time interval measurement. Operationally, it may be desirable to move the location of Gate B, which is 11.25 degrees wide, around a given star to determine if there are any interference sources near that star. Similarly, the threshold setting of each channel can be varied above and below a given setting to determine the optimum setting as well as possible interference effects from dimmer stars.

3.3.3.3.4 Star Mapping. A star map can be attained by moving the location of Gate B to successive angular positions. The peak brightness reading of any detected star will be telemetered directly by the star sensor. The attitude measurement capability of the ADP is used to obtain the azimuth location of a star by selecting the Star 1 input and the elevation of a star by selecting the Star 2 input. Since these measurements are made with respect to the PLL reference pulse, it may take up to three spacecraft revolutions to acquire a single star measurement depending upon the selected alternate attitude measurement. Furthermore, if a detected star is not bright enough to exceed the commanded threshold level, the star sensor will not output a star pulse. Consequently, if no pulse is generated at the location of Gate B, the ADP will "hang up" attempting to make the gated star measurement and will no longer alternate between measurement A and B. The telemetered star brightness would then be used to confirm that a detectable star is not present at the gate location, and the gate can be moved to a new azimuth location.

The use of the star mapping capability would be dependent upon the current knowledge of the spacecraft attitude and the probable number of stars that can be detected. If the approximate attitude is known, Gate B can be moved to the probable star locations. If the attitude is totally uncertain, Gate B (11.25 degrees wide) may be moved in 10-degree increments to attain a 360-degree sweep of the sensor field-of-view or until the approximate attitude can be derived. This would minimize the number of gate locations that would have to be commanded and the time involved since only three to five stars can be expected to be detected in any single spacecraft sweep, i.e., spin revolution. In all conditions, set the star detectable threshold to level 7 (to level 8 only if no stars are detected with level 7), select the ATTH measurements to be the same (SRR to PSI*, or SRR to PSI 2*), and allow telemetry to gather at least 10 revolutions of data (5 samples of each of two measurements). A suggested flow chart to perform a star map is shown in Figure 3.3.3.3-1.

3.3.3.3.5 Star Acquisition. Star acquisition is a process by which the PLL will lock onto the first detectable star anywhere in the spin period irrespective of the location of Gate B. It is intended to be used to lock-up the PLL in the absence of a sun pulse. Conceivably, this would only be necessary if the spacecraft spin axis is within 10 degrees of the sunline (non-standard attitude) or during a condition when either the inertial reference is lost due to an incorrect transfer to a star reference or it becomes necessary to transfer to the redundant ADP. A successful star acquisition can only be accomplished when the star threshold setting has been selected to exclude detectable stars within 45 azimuthal degrees of each other.

The sequence of events to lock to a star pulse is as follows:

(a) Transmit the ADP mode command that contains the following information:
(1) Star acquisition bit = logical "1"
(2) SRR advance bit = logical "1"
(3) Star Gate A and Star Gate B channel select are both equal to the desired star channel (Star "1" or Star "2"). NOTE: Star Gate B is selected in anticipation of making a star brightness measurement after the PLL is phase-locked to the star pulse.

(b) The star acquisition bit (logical "1") performs two functions:
(1) Widens star Gate A in the PLL to 45 degrees.
(2) Enables the SRR advance logic to accept the first star pulse appearing in star Gate A channel select.

(c) The first star pulse that appears on the selected star Gate A channel is used to phase reset the PLL counters and the Hold Zero Phase Error flip flop, thus forcing that star to be in the middle of the 45-degree Gate A.

(d) The PLL then proceeds to produce the correct phase and frequency to lock on this star pulse. The wide Gate A can accommodate the phase transients in this process, which can be in excess of the normal 11.25 degree gate.

Formulation of the command bit structure and the telemetry response is presented in 3.3.4.

3.3.3.4 JCE Control. All command sequences for a thrust maneuver including the latch valve control should be carefully checked before the commands are transmitted. The proper receipt of each command should be carefully verified by telemetry prior to initiating the thrusting sequence. (Status should be as shown in Table 3.3.3.4-1). It is highly recommended that all maneuvers be performed within view of a ground station so that prompt corrective action can be taken, if necessary. For any maneuver sequence, the attitude measurements should be selected to measure spin period (SRR to SRR) and sun aspect angle (SRR to ψ_2) to provide a direct indication of expected performance as well as any anomalous thruster behavior and its effect on spin rate or spacecraft attitude. The command sequence should include a back-up command, "JCE Buffer Output Disable," (ATQ1Ø or ATQØJ) following normal expected completion of the jet firing.

During periods when there is no ground station coverage, it is recommended that both latch valves be closed to protect against any possible failure.

If a maneuver sequence has to be performed in a manner where the actual magnitude of each maneuver is not verified prior to the start of a maneuver, then the magnitude should be transmitted at least two times. This will preclude the possibility of a command rejection causing an improper magnitude execution. The repeated transmission is to insure that the countdown register is changed to the desired value since it would have terminated at the maximum count after completion of the previous maneuver.

A JCE quantitative command or mode command should not be transmitted while a desired maneuver is in progress. The commands will be accepted and will immediately reconfigure the JCE accordingly.

Equally important to selecting the JCE configuration is establishing the proper configuration of the PLL for a particular maneuver. One of the basic determinations is whether it is <u>necessary</u> for the PLL to remain in lock for successful execution of the maneuver, in which case the PLL loss of lock function and the sun gate must be enabled for the maneuver. This would apply for all pulse firing modes where the

PLL is used to generate the angle delay for the azimuth phase angle, but would exclude spin speed changes using the pulse mode. In operating axial thrusters in the alternate fire mode, the PLL phase accuracy is not necessary for the start fire angle, but is necessary to generate the 180° pulse separation (pulse firings 180° apart) to provide cancellation of precession inputs in order to preclude an unwanted attitude disturbance. Again, in this case, the PLL loss of lock function and the sun gate must be enabled.

All continuous firing initiations and terminations are accomplished by time alone, and are independent of the PLL operation. However, when using the axial thrusters in the continuous fire mode, the spin rate should remain essentially constant, and additional failure protection can be obtained by using the loss of lock detection capability, since the PLL should stay in lock during this maneuver. Under such circumstances, any unwanted radial thruster firing (due to a logic failure or improper command selection) would produce a spin change, causing the PLL to lose lock and terminate the firing sequence.

A summary of the use of PLL loss of logic for the various maneuver sequences is presented in Table 3.3.3.4-2.

The PLL has four discrete operating ranges as discussed in 3.3.2.3.4. When a spin change is commanded that causes transition of the spacecraft into a new spin range, the command sequence must include an ADP mode select command to the new spin range. When the spin rate is increased to 15 rpm or higher, the PLL will reacquire lock (based on unit test data) within 120 seconds maximum after the maneuver is completed and the proper spin range has been selected. When the spin rate is decreased to 5 rpm, it may take up to 8 minutes for the PLL to reacquire lock.

Normally, the midrange sun sensor will be selected to provide the sun reference pulses to the ADP. Since its field-of-view (FOV) is limited to ±45 degrees, any precession maneuver outside this range will require that the appropriate extended range sun sensor be selected to complete the maneuver. However, the extended range sun sensor can be used for the entire sequence, since its FOV extends past the 90-degree sunline angle (see Figure 3.3.2.1-5).

The disadvantage of this last approach is the loss of resolution in the sun aspect angle in the initial part of the maneuver. The disadvantage of the first approach is that it would require a two-step sequence. This assumes that the bias angle between the two sensors would have to be compensated for to achieve the desired accuracy. If the bias error was minimal and could be accommodated, the transfer could be done while the maneuver was in progress provided it was controlled so as not to occur while a sun pulse is being generated. This could result in double pulses being provided to the PLL, and cause it to slew off frequency, thereby generating a loss of lock condition. Therefore, it is recommended that while a sun sensor is acting as the SRR, it should NOT be replaced by a different sun sensor <u>during</u> a maneuver.

After the spacecraft has been spun up to 15 rpm (after S/C/Launch vehicle) separation, spin rate detection should be enabled (SRDX selection of ATQØ2 or ATQØB) to provide that protective capability for the remainder of the mission.

The detector uses a 15.625 Hz clock rate to measure the time interval between successive PLL reference pulses. When the PLL is operated in the gated mode, the gate will exclude the reference pulse, if the PLL phase error is greater than half the gate's width (11.25° or 45°). The loss of the reference pulse for more than 16.256 seconds will appear as a low rpm condition (≤3.7 rpm) and terminate any maneuver when this detection is enabled. To preclude premature termination of a spin change maneuver for this reason, large spin change maneuvers, which will create large PLL phase errors, should be performed with the sun gate disabled. The narrow pulse firing mode for spin changes will not cause large phase errors and thus, can be used in the gated mode.

The F_S phase error generated during spin maneuvers using the pulsed firing mode is shown in Figures 3.3.3.4-1 and 3.3.3.4-2 for the narrow (128 ms) pulse mode and in Figures 3.3.3.4-3 and 3.3.3.4-4 for the wide (512 ms) pulse mode.

The typical command sequence for any maneuver is shown in Table 3.3.3.4-3. Use of the PLL loss of lock detection and sun gate mode should be controlled as previously shown in Table 3.3.3.4-1. The table illustrates only a single maneuver. If the sequence is to be followed by another maneuver, the listed commands following the completion of the first maneuver can be revised to configure the JCE accordingly, rather than configuring to an intermediate "SAFE" configuration.

3.3.3.5 Attitude Determination. Attitude determination in its simplest form requires sun aspect measurements and star aspect measurements or multiple Star aspects measurements.

The electrical signals from the sensors are processed by a common attitude data processor (ADP) unit. The time interval between pulses are measured and transmitted to the Ground Station for computation of spacecraft attitude. The understanding of how these time interval measurements are made will facilitate a better overview of the attitude determination problem.

A total of six different time separation measurements may be computed; however, only two may be selected at a time, in any combination. What is actually measured is shown in Table 3.3.3.5-1.

Several types of measurement are unique and are affected by the relation of the TM sample rate with the spin rate. The SRR - Minor Frame Start is one such measurement. Whenever the spacecraft spin rate is higher than the minor frame rate (Figure 3.3.3.5-1), two or more SRR pulses may be expected between two minor frame rate pulses. The ATTM logic is implemented to calculate the smallest time separation between these pulses. This is accomplished by always resetting the ATTM counter with the SRR pulse, if the measurement has not been terminated by the detection of the selected reference pulse. Another interesting point illustrated by Figure 3.3.3.5-1 is that the completion of the measurement does not automatically allow the next measurement to start. The initiation of the readout sequence, i.e., the leading edge of the first of three read envelopes, will parallel transfer the measured data to a TM register for readout during that word time and the next two word times (24 bits total). This same event also allows the alternate measurement to be started at the occurrence of the next SRR pulse. This alternate measurement, if completed, cannot be transferred to the TM register until initiation of the next group of three read envelopes. This insures that the previous measurement is read out at least once. If measurement A is not completed at the initiation of the read out sequence, measurement B will be read out again. However, measurement B (SRR to minor frame start) was measured to the start of the minor frame and the data is only valid in the first minor frame it is read out.

Figure 3.3.3.5-2 on the other hand, represents a condition of high TM sample rate, i.e., many minor frames between successive SRR pulses. The resultant shown here, is that a lot of repeated measurements are sampled. All repeated measurements of SRR to SRR (measurement A) are valid, whereas, only the readout of measurement B in the first minor frame is valid.

3.3.3.6 Roll Reference Transfer. The following is an overview description of transfers. Detailed command sequences are shown in Section 4.3.3.

3.3.3.6.1 Sun Sensor to Sun Sensor. Typically, the sun sensor will be used to provide the roll reference pulse for the PLL. When the attitude of the spacecraft is changed, which requires the use of another sun sensor, e.g., extended range sensor, only the appropriate sun sensor need be selected by command (ATQØ3 or ATQØC). If there is a bias difference between the pulse outputs, there will be a momentary phase transient in the PLL outputs, the magnitude of which is equal to the bias difference. The bias difference can be measured by making an attitude measurement to a known star before and after the transfer process. The RIP location should be corrected if the bias difference is greater than a one-bit change (0.351 degree) in the RIP angle delay capability. The phase transient that occurs during transfer can be avoided by commanding SRR Advance (i.e., phase reset) when the new sun sensor is selected.

3.3.3.6.2 Sun Sensor to Star Sensor. The transfer from a sun reference to a star reference requires that Gate B be commanded to an angular location containing the desired star. The star sensor threshold setting should then be set commensurate with the predicted star intensity to achieve an acceptable probability of detection and false alarm rate. The transfer is accomplished by using the SRR Advance bit and selecting the desired star channel (ATQØ3 or ATQØC).

The PLL must be in lock for a successful transfer so that the PLL is not being controlled by the SIMSRR mode. Otherwise, after the phase is reset by the star reference, the SIMSRR mode will attempt to slew the PLL back so the initial phase, causing Gate A to move. If the phase error is large (i.e., 5.625° = one-half the width of Gate A), Gate A will no longer contain the desired star after 2 revolutions. If lock status is not known at the time of transfer, the PLL loss of lock should be inhibited during the transfer process to prevent the loss of lock detection from interfering with the transfer. The loss of lock detection should be immediately enabled after the transfer has been accomplished.

The transfer will change the phasing of all the PLL outputs. The RIP location and the location of Gate B are affected and require additional commands to restore operation to normal.

3.3.3.6.3 Star Sensor to Sun Sensor. The transfer from a star reference to a sun reference is made easier because the sun is an unambiguous reference requiring no spin gate for initial capture. Consideration to the phase changes in the PLL outputs as discussed in the preceeding paragraph is still necessary.

3.3.3.6.4 Star Reference to Star Reference. When transferring from one star reference to another star reference, the new star reference must be located within Gate B. Configuring the ADP mode command (ATQØ3 or ATQØC) to perform an SRR advance will cause the SRR to transfer to the new star. If the transfer involves transferring to the alternate star channel, the ADP mode command (ATQØ3 or ATQØC) must include selection of the desired channel for Gate A as well as the SRR advance command bit.

3.3.3.7 Science Output Control

3.3.3.7.1 General. All of the reference pulses provided by the controls subsystem for use by the scientific instruments are dependent upon the operation of the PLL in the ADP. This implies that the PLL must be in lock with the selected roll reference for proper performance of any of these signals. When the simulated SRR is used as a reference, the phase accuracy of any output is directly dependent on how accurately the commanded simulated spin period matches the actual spin period.

3.3.3.7.2 Roll Reference Signal (F_S). The roll reference signal (F_S) is an output of the PLL that occurs once per spin revolution synchronous with the occurrence of the selected roll reference (a small bias can exist between the SRR and the F_S, the magnitude and polarity of which can be calibrated). The time occurence of this pulse in inertial coordinates is dependent upon the selected roll reference, such as the sun or a particular star. A discussion of these phase changes is presented in 3.3.2.4.4.

The occurrence of the F_S pulse can be related to telemetry by making the measurement of SRR to Minor Frame Start or to Major Frame Start (ATTM1Z and ATTM2Z). The phase difference between the SRR and the F_S event time can be calibrated at each spin speed by setting the Roll Index delay magnitude to zero which makes the occurrence of the RIP coincident with F_S. By making the measurement SRR to RIP, the phase delay through the PLL is then measured.

3.3.3.7.3 Spin Period Section Pulses (2^{10} F_S). The spin period sector pulses are generated by the PLL as a continuous pulse train having 1024 near-equally spaced pulses per spin revolution.

Phase Lock of the F_S pulse to the SRR is controlled in the PLL by correcting the VCO frequency as a function of the magnitude and polarity of the phase difference between the F_S and the SRR pulses. The phase error correction is completed within 1.8 seconds following receipt of both pulses (SRR and F_S). This means that the pulse to pulse jitter of the sector pulses caused by the VCO frequency correction will be at a maximum for the first 1.8 seconds following receipt of the later pulse. The maximum jitter is expected to be less than 0.03 degree. Following the 1.8 second correction interval, the pulse to pulse jitter is essentially unmeasurable.

3.3.3.7.4 Roll Index Pulse (RIP). The RIP is a pulse that occurs once per spin revolution at a commanded angular delay from the occurrence of the F_S pulse. The angular delay is counted using the 1024 sector pulses (2^{10} F_S), providing an angular increment of 360°/1024 or 0.351 degree. The RIP pulse occurrence is coincident with the sector pulse at the commanded angular delay. The actual location can be measured with respect to the input reference by making the measurement of SRR to RIP. Whenever the selected roll reference is changed or a different star is selected when using the star reference, the inertial location of the F_S pulse will change. Since the RIP will shift in phase accordingly, a new angular delay must be commanded at the time of transfer to maintain the RIP at the same inertial location. However, unless the transfer command and the new RIP delay are timed to occur between the occurrence of the old reference pulse and the new reference pulse, the RIP may occur at the wrong angle for one revolution during the transfer process. The fact that the commanded RIP delay is loaded into a working register, each revolution at the occurrence of the F_S pulse can cause the one revolution ambiguity. The timing relationship further clarifying this point can be seen in Figure 3.3.3.7-1.

3.3.3.8 Unit Redundancy. Redundancy within the control subsystem is achieved by using the following complement of units:

(a) Two ADPs
(b) One star sensor having two channels which are electrically redundant.
(c) Two sets of sun sensors with each set containing a mid-range and a lower and upper extended range sun sensor.

The interconnection of these units and the interface with other units to achieve redundancy is presented in Table 3.3.3.8-1. It is emphasized that each ADP configure command interfaces with only one COM, whereas each discrete command interfaces with either of two COMs.

3.3.3.9 Operational Restrictions

3.3.3.9.1 Operation of two ADPs Simultaneously. The command capability permits both ADPs to be operating simultaneously. This feature was intended to be used solely for the purpose of performing the initial spin-up maneuver which occurs out of view of any ground station. At all other times, only one ADP should be on at a time, because the telemetry channel for attitude measurements and status data are shared between the two ADPs. These data channels would become meaningless with both units operating simultaneously. Additionally, the interface with the star sensor would be controlled by both ADPs, producing non-standard operation.

3.3.3.9.2 ADP Attitude Measurements. Attitude measurements made in the ADP alternate between the two command selected measurements. If either of the two measurements is not completed due to that event not occurring, the ADP will "hang up" waiting for this event to occur, i.e., there is no default condition to cycle to the alternate measurement. Telemetry will, however, continually readout the previously measured data. This possibility is likely to occur when transferring to a star reference for the SRR. Typically, the desired star is located within Gate B, and the attitude measurement selected would be "SRR to Gated Star B". When the transfer is made to this star, it will no longer occur in Gate B, causing the attitude measurement to "hang up". After the reference transfer, the gate should be moved or a new attitude measurement selected to allow attitude measurements to continue.

The attitude measurement "SRR to Minor Frame Start" shall only be selected when ACS data is being telemetered at minor frame rate in order to be able to identify the specific minor frame to which the measurement is referenced. Furthermore the data is valid only in the first minor frame that is read-out, since that is the frame to which the measurement was made. The same data will however, be continuously read out until the alternate (on-going) measurement is complete.

The attitude measurement "SRR to Major Frame Start" shall only be selected when ACS data is being telemetered at a subcom rate, so as to provide an unambiguous reference to telemetry time. The measurement can be made in the minor frame mode, but could take an entire major frame to complete. In the meantime, the alternate measurement would be read out each minor frame, thus defeating the purpose of the high sample minor frame rate.

3.3.3.9.3 JCE Restrictions. The JCE countdown command (ATQØ4 or ATQØD) or the jet control command (ATQØ2 or ATQØE) should not be sent to the ADP while a desired maneuver is in progress. The commands will be accepted and will configure the JCE accordingly.

If a maneuver appears to have terminated prematurely, possibly due to on-board detection logic, the maneuver should not be restarted until the contents of the countdown register are verified. If the register had counted down completely, it would have terminated at full count. Restarting the sequence would then initiate firing for maximum duration.

3.3.3.9.4 Programmed Initial Spin-Up Sequence. It is imperative that the commands to turn the ADPs on for the initial spin-up maneuver be programmed through the proper COMs. The ADP No. 1 ON command via COM 5 (ADP19) and ADP No. 2 ON command via COM 6 (ADPB9) are the only ones to be loaded into the stored command sequencer for this maneuver. These commands are inter-locked through the separation switch and are not operative until spacecraft separation has occurred. This prevents any failure or an anomalous event from initiating the stored sequence and firing thrusters while in the launch configuration.

3.3.3.10 Non-Standard Modes

3.3.3.10.1 Loss of Roll Reference - (Outside Gate A). When the ADP is operated with the PLL loss of lock enabled, it will transfer to the simulated SRR reference, "free run mode" if the reference pulse is missing for two consecutive revolutions. If the correct simulated spin period is commanded, the PLL will automatically transfer back to the selected roll reference pulse when the pulse reappears for two consecutive revolutions. If the commanded spin period is not sufficiently accurate, causing Gate A to move more than 5.5

degrees inertially, the reference pulse will be outside Gate A and the transfer will not occur (recognized by ALOCKS and AMISSC, indicating sustained loss of lock and missed SRR counts accumulating). To recover from this condition, two possibilities are presented. The first, which is preferred and the easiest, assumes that the sun is visible or the operation is delayed until it is visible. The second method assumes that the sun is not visible and a star reference must be used.

(a) Method One (Sun Reference Available): If the sun is visible, a single ADP mode command will allow the ADP to perform a sun acquisition and re-establish lock. The command (ATQØ3 or ATQØC) would be configured as follows:

PLO:	Loss of lock inhibited selection (X = 0)
SSRX:	X Selected as required for spacecraft attitude.
REF SUN:	To select sun reference.
SUGD:	Disable sun gate for 360 degree sun detection.
SRRADV:	Transfer phase reference to sun when detected.

Restoration of PLL lock can be verified by measuring SRR to RIP as discussed in 3.3.3.1, and observing the decay of the phase transient.

(b) Method Two (Sun Reference Not Available): If the sun is not visible, a star acquisition would have to be performed to obtain a reference. Consideration would first have to be given to the commanded star sensor threshold setting to insure a high probability of detecting a star. If a star reference has been reliably detected previously, and has merely slipped out of Gate A, the threshold setting should be left the same. It could possibly be lowered one setting to enhance the detection capability, provided that there are no interference sources within 45 degrees of the desired star. When star acquisition is commanded, the PLL will lock onto the first detectable star. The command sequence is presented in 3.3.3.3.5. By locating Gate B over Gate A (ATQØ5 or ATQØE), the star intensity can be measured to ascertain if this star is the desired reference. If it is not, Gate B can be moved to find the desired star reference. When found, transfer to it may be performed as previously described in 3.3.3.7.4.

3.3.3.10.2 Loss of Roll Reference - (Outside Sensor FOV - Maneuver Induced). The roll reference may be lost during the execution of a precession maneuver if the sun moves out of the sun sensor FOV. If the maneuver was being performed on the midrange sun sensor, the selection of the appropriate extended range sun sensor (ATQØ3 or ATQØC) will provide a sun reference pulse. If phase reference has been lost, a sun acquisition must be performed as described in Method One, above.

If the FOV of an extended range sun sensor is exceeded as a result of the maneuver (sun angle less than 10 degrees), the SRR would have transferred to the "free run mode". This mode still provides a reference pulse which can be used to perform a maneuver in the reverse direction. Even if the phase of the PLL has changed due to an incorrect simulated spin period, the maneuver should be performed to reacquire the sun. After the attitude has been established, a maneuver to the correct attitude can be performed. To perform a precession maneuver using the free run mode, the ADP mode command (ATQØ3 or ATQØC) would be configured as follows:

PLO:	Select PLL Inhibit (X = 0) to prevent loss of lock from terminating the maneuver.
SSRX:	X Selected for the required extended range sun sensor.
REF SIM:	Simulated reference selected for maneuver reference. (SUN is not selected since it may introduce PPL transient when sun is detected).

3.3.4 Control Subsystem Command Response. Table 3.3.4-1 presents the command responses for the Control Subsystem. The table lists every command that directly affects the subsystem, and the telemetry indication that verifies the proper execution of the command. The mnemonics for the command and telemetry parameters are also included.

When a unit is in the unpowered OFF state, all telemetry from that unit will be reading zero scale (uncalibrated), except temperature telemetry which is powered by DIMS.

TABLE 3.3.2.2.6-1

S/C + X AXIS/SUN SENSOR BORESIGHT BIAS

(1)	Sun Sensor Leading Edge Detection Level (150 μamp level) to Leading Edge of ADP Detected Pulse (SRR Pulse when Sun is Selected)	-0.02°	(Note 1)
(2)	Sun Sensor Boresight to Sun Sensor Leading Edge Detection Level (150 μamp Level)	+0.75°	(Note 2)
(3)	+X Axis to Sun Sensor Boresight	-7.43°	
(4)	SRR to P_S Bias (PLL) (Maximum)	±0.25°	(Note 3)

NOTE 1: A positive time delay reflects itself as a negative angle (Positive angle is in the positive spin direction).

Varies essentially with spin rate. A value may be estimated via use of Figure 3.3.2.1-4.

NOTE 2: Varies with solar intensity. The absolute value may be estimated via use of Figure 3.3.2.1-3 and interpolating geometrically as necessary. The change in this value due to solar intensity may be measured by comparing ATTM measurements to the same star, using the sun as the SRR for all measurements. The absolute value may be determined by measurements to two or more stars, and determining star sensor biases concurrently via iterative estimation.

NOTE 3: This is measurable by commanding the SRR to RIP delay to zero degrees and reading "SRR to RIP" ATTM. (See Figure 3.3.2.1-3 and Paragraph 3.3.3.1)

TABLE 3.3.2.2.6-2

S/C + X AXIS/STAR SENSOR BORESIGHT BIAS

(A)	Delay between PSI* Optical Axis (Star sensor totaling edge detection level, i.e., 25% of peak) and PSI* position pulse (SRR pulse when PSI* is selected)	-0.25° (Note 1)
(B)	Star Sensor Boresight to PSI* Optical Axis	+6.15° (Note 2)
(C)	+X Axis to Star Sensor Boresight	+15.1°
(D)	SRR to F_S Bias (PLL) (Maximum)	±0.25° (Note 3)

NOTE 1: A positive time delay reflects itself as a negative angle (Positive angle is in the positive spin direction).

The value shown here is equivalent to a nominal 2 millisecond delay at 15 rpm. Actual delay is dependent on the star brightness, star elevation, star channel, bandpass selection, and spin rate. A value may be determined iteratively with determination of star elevation, via use of Figure 3.3.2.2-3A and 3.3.2.2-3B if the PSI* channel is being used for the SRR, or via use of Figure 3.3.2.2-3C and 3.3.2.2-3D, if the PSI2* channel is being used for the SRR.

NOTE 2: If PSI2* slit is selected for star reference, this value would be deleted from the list.

NOTE 3: This is measurable by commanding the SRR to RIP delay to zero degrees and reading the "SRR to RIP" ATTM.

TABLE 3.3.2.3-1

ATQØ1 (ADP #1) OR ATQØA (ADP #2)
QUANTITATIVE COMMAND ASSIGNMENTS

Command Description	Magnitude Data															
	25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
Attitude Data	LSB														MSB	
Processor Configure	Address				Magnitude											
Measurement Select	0	0	0	0	X	X	X	X	X	X	A	A	A	B	B	B
A) Measurement A Select																
1) SRR - SRR											0	0	0			
2) SRR - Minor F Start											1	0	0			
3) SRR - RIP											0	1	0			
4) SRR - MIP (Invalid on Multiprobe)											1	1	0			
5) SRR - Gated Star B											0	0	1			
6) SRR - PSI 2											1	0	1			
7) SRR - Major F Start											0	1	1			
B) Measurement B Select																
1) SRR - SRR														0	0	0
2) SRR - Minor F Start														1	0	0
3) SRR - RIP														0	1	0
4) SRR - MIP (Invalid on Multiprobe)														1	1	0
5) SRR - Gated Star B														0	0	1
6) SRR - PSI 2														1	0	1
7) SRR - Major F Start														0	1	1

TABLE 3.3.2.3-2

ATTM TM FORMAT

Word No.	Ref. Desig.	LOC Code	Telemetry Description	TM Word 0	1	2	3	4	5	6	7
1	ATTM1Z (Redun.)	6SD06 7SD05 (For Minor Frame)	Attitude Measurement	LSB							
			Measurement Data (8 LSB's of 16 bit word)	0							7
			Measurement Data (8 MSB's of 16 bit word)	8							15
			Range: 0 sec to 16.38 sec (0 to 65535 counts)								MSB
2	ATTM2Z	6SD07 7SD06 (For Sub-com)	Resolution: 0.25 msec								
3	AM1ADS		Measurement A Address	0	1	2					
			SRR - SRR	0	0	0					
			SRR - Minor Frame Start	1	0	0					
			SRR - RIP	0	1	0					
			SRR - MIP (Invalid on Multiprobe)	1	1	0					
			SRR - Gated Star B	0	0	1					
			SRR - PSI 2	1	0	1					
			SRR - Major Frame Start	0	1	1					
	AM2ADS		Measurement B Address				3		5		
			SRR - SRR				0	0	0		
			SRR - Minor Frame Start				1	0	0		
			SRR - RIP				0	1	0		
			SRR - MIP (Invalid on Multiprobe)				1	1	0		
			SRR - Gated Star B				0	0	1		
			SRR - PSI 2				1	0	1		
			SRR - Major Frame Start				0	1	1		
	ATTMSS		Data Measurement A = 1, B = 0							6	
			Unused Bit								7 X

TABLE 3.3.2.4-1

PREDICTED DAMPER PERFORMANCE SUMMARY FOR MULTIPROBE

MULTIPROBE				
Mission Phase	Temp. (^{o}F)	Inertia Ratio	Spin Speed (rpm)	τ* (Min)
Separation/Cruise	21	1.25	15	7.2
	77			12.0
	140			18.3
After Large Probe Separation	21	1.49		22.3
	77			28.6
	140			36.0
	21		48.5	10.0
	77			13.4
	140			17.6
After Small Probe Separation	21	1.46		4.6
	77			6.3
	140			8.4
Bus Entry	21		10	13.7
	77			17.6
	140			22.0

*τ is the time required to produce $1/\epsilon$ of the starting value.

TABLE 3.3.3.2-1

TELEMETRY INDICATIONS FOR ADP STATUS

Mnemonic for Pertinent TM	Status After Sun Acquisition	Status After Commanded to Normal Operating Mode
ATTM1Z and ATTM2Z	• SRR to SRR equal to expected spin period. • SRR to RIP = 0.0° ±0.25° (0° delay command) • SRR to PSI2 equal to 90.0° ±10°.	• SRR to SRR equal to expected spin period. • SRR to RIP equal to commanded delay ±0.25°. • SRR to PSI2 equal to 90.0° ±2°. • SRR to Gated Star B (dependent on selected star).
AJMAGC	• JCE countdown at maximum count 4095.	• JCE Countdown at a count of zero.
APULLS	• Pulse width selection = 128 ms	• Pulse width selection = 128 ms.
AMAGCS	• Pulse time count selection = time count.	• Pulse time count selection = time count.
AJETMS	• Pulse fire mode selected.	• Pulse fire mode selected.
ASDETS	• Spin rate detector enabled.	• Spin rate detector enabled.
AJETMS	• Normal fire mode select.	• Normal fire mode selected.
AJETSS	• No jets selected.	• No jets selected.
AJCEFS	• JCE fire status disabled.	• JCE fire status disabled.
A*GASS	• Star Gate A channel select as commanded.	• Star Gate A channel select as commanded.
A*GBSS	• Star Gate B channel select as commanded.	• Star Gate B channel select as commanded.
ASUNSS	• Sun sensor selection as commanded.	• Sun sensor selection as commanded.
ASRRMS	• Sun selected as SRR	• Sun selected as SRR.
A*ACQS	• Star normal mode selected.	• Star normal mode selected.
ASPINS	• Spin range selection = 8.0 to 17.7 rpm.	• Spin range selection as commanded.
ALOLES	• PLL Loss of Lock inhibited.	• PLL Loss of Lock enabled.
ASUNGS	• Sun gate disabled.	• Sun gate enabled.
ADVANS	• SRR advance set to normal.	• SRR advance set to normal.
ARIPAD	• RIP delay equal to zero degrees.	• RIP delay as commanded.
ALOCKS	• In Lock Status.	• In Lock Status.
AMISSC	• Normally = 0 Counts	• Normally = 0 Counts.
AACSAD	• ACS Angle Delay is a Random Selection.	• ACS Angle Delay as Commanded.
ASIMSZ	• PLL Spin Period as Commanded.	• PLL Spin Period as Commanded.

NOTES:
- Status after Sun Acquisition applies for initial ground station acquisition.
- Status for Normal Operating Mode applies for 15 rpm and Cruise Configuration.

TABLE 3.3.3.3-1

TELEMETRY INDICATIONS FOR STAR SENSOR RESPONSE

MNEMONIC FOR PERTINENT TM	TM CHANNEL	TYPICAL STATUS OR EXPECTED VALUE
ASTART	Star Sensor Temperature	Dependent on spacecraft attitude and solar intensity; 57±28°F at sun aspect of 90±5°
A*1BRM	Star 1 Brightness	Dependent on location of star gate B and star magnitude within the gate.
A*1BPS	Star 1 Bandpass State	Consistent with command selected value (S1BWX selection of STRQ1 or STRQA).
A*1THS	Star 1 Threshold Setting	See Section 3.3.4 for expected value associated with commanded threshold setting.
A*2BPS	Star 2 Bandpass State	Consistent with command selected value (S2BWX selection of STRQ1 or STRQA).
A*2BRM	Star 2 Brightness	Dependent on location of star gate B and the star magnitude within the gate.
A*2THS	Star 2 Threshold Setting	See Section 3.3.4 for expected value associated with commanded threshold setting.
A*1ONS	Star 1 ON/OFF	Logic Level "1" if ON.
A*2ONS	Star 2 ON/OFF	Logic Level "1" if ON.

TABLE 3.3.3.3-2

STAR BRIGHTNESS CALIBRATION METHOD

Case A: Starting with telemetered data, determine silicon magnitude of the star.

(A.1) Calculate the mean and 3σ deviation from the mean of 30 successive TM samples of A*1BRM or A*2BRM (corresponding to that star channel that was assigned to Star Gate B at the time of data collection). Reject all samples outside the 3σ deviation, and calculate the mean of the remaining samples ($\equiv \mu_1$).

(A.2) For worst case conditions (to account for end-of-life, possible errors in system calibration, and Johnson Reference), increase μ_1 by 14 percent:

$$\mu_2 = (1.14)\ \mu_1$$

(A.3) To account for star sensor FOV effect on the sensor input signal level, calculate $\mu_3 = (1 - .01P)\ \mu_2$, where P = the percent change obtained from Figure 3.3.2.2-8A for the PSI* channel and Figure 3.3.2.2-8B for the PSI2* channel.

(A.4) Calculate the equivalent silicon magnitude for the star:

$$M_{si} = \text{silicon magnitude} = -2.512 \log\left(\frac{\mu_3 - A}{BI_\Theta}\right)$$

where: A = Offset
B = Gain
Io= DC video for "0" magnitude star
μ_3= AC video for the subject star.

TABLE 3.3.3.3-2. (Continued)

	Parameters A and B are obtained from Table 3.3.3.3-4, corresponding to the prevailing spin rate and selected star channel bandpass. Parameter Io = 344.9 mv, for the PSI* channel; Io = 362.1 mv. for the PSI2* channel.
Case B:	Starting with the silicon magnitude of a star, determine the expected telemetered value for star brightness (A*1BRM or A*2BRM).
(B.1)	Calculate the uncorrected star sensor signal input level, μ_3 = uncorrected signal $A+BI_\Theta(2.152)^{-M_{si}}$ where the terms are as defined in step (A.4) of this table.
(B.2)	To account for star sensor FOV effect on the sensor input signal level, calculate: $\mu_2 = (1+0.01P)\mu_3$, where P = the percent change obtained from Figure 3.3.2.2-8A or 3.3.2.2-8B (as applicable).
(B.3)	For worst case conditions (to account for end-of-life, possible errors in system calibration, and Johnson Reference), decrease μ_2 by 12 percent: $$\mu_1 = (.88)\ \mu_2$$ μ_1 is the expected telemetered value in mv. for A*1BRM or A*2BRM, as applicable.

TABLE 3.3.3.3-3

DETECTABLE STAR LIST

Bright Star Number**	Silicon Magnitude PSI* Channel	Silicon Magnitude PSI2* Channel	Star Astronomical Nomenclature	Location Celestial Latitude	Location Celestial Longitude
2491	-1.35	-1.35	Sirius (A CMA)	-39.599	103.723
2061	-0.73	-0.75	Betelgeuse (A ORI)	-16.031	88.391
2326	-0.74	-0.74	Canopus (A CAR)	-75.827	104.602
5340	-0.46	-0.47	Arcturus (A BOO)	30.754	203.871
5459	-0.39	-0.40	Rigil Kentaurus (A CEN)	-42.587	239.152
6134	-0.14	-0.15	Antares (A SCO)	- 4.567	249.399
1708	-0.09	-0.10	Captella (A AUR)	22.864	81.494
7001	+0.10	+0.10	Vega (A LYR)	61.734	284.951
1713	+0.16	+0.16	Rigel (B ORI)	-31.126	76.466
1457	+0.21	+0.19	Aldebaran (A TAU)	- 5.469	69.426
2943	+0.31	+0.30	Procyon (A CMI)	-16.015	115.427
4763	+0.51	+0.48	Gacrux (G CRU)	-47.828	216.377
472	+0.52	+0.52	Achernar (A ERI)	-59.377	344.941
5267	+0.59	+0.59	Hadar (B CEN)	-44.134	233.431
8636	+0.64	+0.63	Beta Grus (B GRU)	-35.429	321.624
7557	+0.77	+0.77	Altair (A AQU)	29.304	301.409
4730	+0.81	+0.81	Acrux (A CRU)	-52.876	221.510
2990	+0.92	+0.90	Pollux (B GEM)	6.682	112.857
5056	+0.96	+0.97	Spica (A VIR)	- 2.053	203.478
8728	+1.23	+1.23	Fomalhaut (A PAU)	-21.132	333.494
4853	+1.27	+1.28	*Mimosa (B CRU)	-48.636	221.286
7924	+1.29	+1.29	Deneb (A CYG)	59.907	334.971
337	+1.31	+1.30	Mirach (B AND)	25.944	29.706
3634	+1.43	+1.42	Al Suhail (L VEL)	-55.871	160.495
3982	+1.44	+1.44	*Regulus (A LEO)	0.464	149.468
-	+0.74	+0.74	Saturn***		

Star positions from 1974 American Ephemeris.

Listed in approximate descending order of brightness for silicon detector response as documented in reference 1.5.23.

* Not reliable for spin rates above 30 rpm.
**Yale University Observatory catalog edited by D. Hoffleik.
***Saturn silicon magnitude applies for the spacecraft located at Venus on 8 December 1978.

TABLE 3.3.3.3-4

Offset (=A) and Gain (=B) versus Spin Rate, Bandpass and Selected Star Channel. (Ref: 1.5.23)

Channel	PSI*		PSI*	
Bandwidth (H_z)	2	19	2	19
Rate: 4.7 (RPM)	A= 107.13 B= 3.532	A= 43.35 B= 2.506	A= 43.73 B= 3.651	A= 13.58 B= 2.544
10	87.40 3.925	40.20 3.095	27.50 3.950	8.00 3.105
15	68.11 4.149	36.52 3.279	11.97 4.071	2.71 3.269
20	65.92 4.199	58.50 3.337	16.10 4.066	16.38 3.294
25	59.00 4.050	57.00 3.340	15.70 3.940	11.00 3.280
30	52.40 3.852	55.71 3.256	15.34 3.771	6.46 3.210
35	54.30 3.610	57.30 3.115	20.10 3.535	12.20 3.070
40	56.50 3.320	58.10 2.955	25.00 3.300	18.10 2.885
45	58.81 3.097	59.02 2.735	29.77 3.054	23.68 2.692
50	57.35 2.885	64.67 2.545	41.17 2.824	22.05 2.520
55	50.93 2.697	62.49 2.388	39.61 2.645	21.24 2.372
60	48.71 2.522	56.17 2.248	45.20 2.471	29.90 2.221

TABLE 3.3.3.3-5

Minimum star brightness level (as telemetered) required for time interval measurements versus selected threshold level, bandpass and star channel. (Ref: 1.5.23)

Threshold Level	Pd (Note 1)	PSI* 2Hz 0.993	PSI* 19Hz 0.993	PSI-2* 2Hz 0.993	PSI-2* 19Hz 0.993	PSI* 2Hz 0.999	PSI* 19Hz 0.999	PSI-2* 2Hz 0.999	PSI-2* 19Hz 0.999
8	V(mv)	210	208	168	161	226	223	183	175
7		240	236	196	191	255	252	211	205
6		268	273	228	232	284	288	242	245
5		324	321	267	268	340	336	282	283
4		523	523	450	450	540	540	465	465
3		800	800	715	715	820	820	732	732
2		1235	1235	1100	1100	1255	1255	1117	1117
1		1980	1980	1720	1720	1920	1920	1735	1735

Note 1: Pd = Probability of Detection.

TABLE 3.3.3.4-1 TELEMETRY INDICATIONS FOR ADP STATUS FOR THRUST MANEUVERS

Mnemonic for Pertinent TM	Status Prior to Executing Maneuver Sequence	Status After Completion of Maneuver Sequence	Remarks
ATTM1Z and ATTM2Z	o SRR to SRR equal to expected spin period	o SRR to SRR equal to expected spin period	o Change dependent on magnitude of spin maneuver
	o SRR to PSI 2 equal to 90° ± 2°	o SRR to PSI 2 equal to expected sun aspect angle ± 2°	o Change dependent on magnitude of precession maneuver
AJMAGC	o JCE countdown at commanded count	o JCE countdown at maximum count of 4095	
APULLS	o Pulse width selection as commanded	o Pulse width selection as commanded	
AMAGCS	o Pulse time count selection as commanded	o Pulse time count selection as commanded	
AJETMS	o Pulse fire mode selection as commanded	o Pulse fire mode selection as commanded	
ASDETS	o Spin rate detector enabled	o Spin rate detector enabled	
AJETSS	o Jets selection as commanded	o Jets selection as commanded	
AJCEFS	o JCE fire status disabled	o JCE fire status disabled	o JCE fire status becomes enabled only for actual firing duration
A*GASS	o Star Gate A channel select as commanded	o Star Gate A channel select as commanded	
A*GBSS	o Star Gate B channel select as commanded	o Star Gate B channel select as commanded	

TABLE 3.3.3.4-1 TELEMETRY INDICATIONS FOR ADP STATUS FOR THRUST MANEUVERS (cont'd)

Mnemonic for Pertinent TM	Status Prior to Executing Maneuver Sequence	Status After Completion of Maneuver Sequence	Remarks
ASUNSS	o Sun sensor selection as commanded	o Sun sensor selection as commanded	
ASRRMS	o Sun selected as SRR	o Sun selected as SRR	
A*ACQS	o Star normal mode selected	o Star normal mode selected	
ASPINS	o Spin range selection as commanded	o Spin range selection as commanded	
ALOLES	o PLL loss of lock (per Table 3.3.3.5-2)	o PLL loss of lock selection remains unchanged	
ASUNGS	o Sun gate enable/disable status (per Table 3.3.3.5-2)	o Sun gate enable/disable selection remains unchanged	
ADVANS	o SRR advance set to normal	o SRR advance set to normal	
ARIPAD	o RIP delay as commanded	o RIP delay as commanded	
ALOCKS	o In lock status	o In lock status	With PLL loss of lock detection inhibited, status will still show "in lock", although SRR may be outside Gate A.
AMISSC	o Normally = 0 counts	o Dependent on maneuver effect on PLL operation	
AACSAD	o ACS angle delay as commanded	o ACS angle delay as commanded	

TABLE 3.3.3.4-1 TELEMETRY INDICATIONS FOR ADP STATUS FOR THRUST MANEUVERS (cont'd)

Mnemonic for Pertinent TM	Status Prior to Executing Maneuver Sequence	Status After Completion of Maneuver Sequence	Remarks
ASIMSZ	o PLL spin period as commanded	o PLL spin period as commanded	
AJCE1S	o JCE 1 buffer output status = 1 (ADP 1 ON); = 0 (ADP 2 ON)	o JCE 1 buffer output = 0	
AJCE2S	o JCE 2 buffer output status = 0 (ADP 1 ON); = 1 (ADP 2 ON)	o JCE 2 buffer output = 0	

TABLE 3.3.3.4-2

USE OF PLL LOSS OF LOCK FOR THRUST MANEUVERS

Maneuver Type	PLL Loss of Lock Selection	Sun Gate Selection	Remarks
Continuous Radial ($\Delta W \equiv$ Spin Rate Change).	Disabled	Disabled	PLL response unable to maintain SRR within Gate A during large spin changes. Removal of gate allows reference pulse to be detected once per revolution for proper operation of spin rate detector and preventing erroneous termination.
Continuous Axial ($\Delta V \equiv$ Velocity Change).	Enabled	Enabled	Loss of lock detection not required for maneuver accuracy, but protects against a radial thruster failure.
Pulsed Axial, 1 Per Revolution ($\Delta P \equiv$ Precession).	Enabled	Enabled	Loss of lock detection required for phase angle accuracy; protects against a radial thruster failure.
Pulsed Axial, 2 Pulses per Rev. (ΔV)(180° intervals).	Enabled	Enabled	Loss of lock detection required to maintain 180-degree phase separation between pulses.
Pulsed Radial Pair (ΔV)	Enabled	Enabled	Loss of Lock detection required for phase angle accuracy; protects against a radial thruster failure.
Pulsed Radial (512 ms) (ΔW).	Disabled	Disabled	Loss of lock detection disabled to prevent PLL loss of lock termination.
Pulsed Radial (128 ms) (ΔW).	Enabled	Enabled	PLL can track phase angle change in either normal or alternate fire modes.

TABLE 3.3.3.4-3

TYPICAL MANEUVER SEQUENCE

Step No.	Command Mnemonic	Command Title	Remarks
1.	ATQØ2 or ATQØB	Jet Control	Select jets to be fired, firing mode, and spin rate detection.
2.	ATQØ5 or ATQØE	ACS Angle Delay Magnitude.	Only necessary when using pulse fire mode.
3.	ATQØ4 or ATQØD	Jet Countdown	Select desired pulse count or time count.
4.	ATQØ3 or ATQØC	ADP Mode Select	Only necessary to disable sun gate and PLL loss of logic detection.
5.	VAL19 or VALA9	Latch Valve 1 OPEN	Open Latch Valve No. 1 to fire 3R, 4R, A5 .
	AND/OR:		
	VAL29 or VALB9	Latch Valve 2 OPEN	Open Latch Valve No. 2 to fire 1R, 2R, or A6.
6.	ATQØ9 or ATQØI	JCE Buffer Output ENABLE	Enables JCE Output Buffers.
7.	ATQ11 or ATQØK	Jet Fire Interlock	Enables JCE Firing Logic.
8.	ATQ12 or ATQØL	Jet Fire	Initiates Thruster Firing.
9.	ATQ1Ø or ATQØJ	JCE Power Buffer Output Disable	Command to be sent after a delay compatible with commanded magnitude.
10.	VAL1Ø or VALAØ	Latch Valve 1 CLOSE	As deemed necessary to provide adequate safeguard after completion of maneuver.
	AND/OR:		
	VAL2Ø or VALBØ	Latch Valve 2 CLOSE	
11.	ATQØ3 or ATQØC	ADP Mode Select	Only necessary at this time if selection of different spin range is required.
12.	ATQØ4 or ATQØD	Jet Countdown	Set to zero count, or to account for the next maneuver if maneuver is to be performed immediately.
13.	ATQØ2 or ATQØB	Jet Control	Set all jet select bits to zero or to the jet combination for the next maneuver if a subsequent maneuver is to be performed immediately.
14.	ATQØ3 or ATQØC	ADP Mode Select	Only necessary to re-establish sun gate and PLL loss of lock detection if disabled in Step No. 4. Command to be sent after PLL has obtained lock.

TABLE 3.3.3.5-1

MEASURED PARAMETERS VIA COMBINATIONS OF ATTM MEASUREMENTS

Commanded Measurement:	Measurement is W.R.T. the Following Selected Reference			
	Sun (ψ Only)	(Star Gate A Channel 1(ψ*))	(Star Gate A Channel 2(ψ_2*))	Simulated SRR
1. SRR to SRR	Actual Spin Period	Actual Spin Period	Actual Spin Period	Simulated Spin Period
2. Star - Minor Frame Start	Determination of Azimuth Reference (In TM Clock Time)			
3. SRR - RIP	Science Ref.	Science Ref.	Science Ref.	Science Ref.
SRR - MIP (Invalid on Multiprobe)	Invalid on Multiprobe.			
4A SRR - Gated Star B (Channel 1)	Separation of Star in Gate B W.R.T. the Sun.	Separation of Star in Gate B W.R.T. star in Gate A.	Aspect angle of Star in Gate B W.R.T. star in Gate A, measured in plane parallel to ψ_2* slit.	Simulated Drift Rate
4B SRR - Gated Star B (Channel 2)	Aspect angle of Star in Gate B W.R.T. the Sun.	Aspect angle of Star in Gate B W.R.T. star in Gate A.	Biased aspect angle of Star in Gate B W.R.T. Star in Gate A.	Simulated Drift Rate.
5. SRR - ψ_2	Sun Aspect Angle.	Aspect Angle of Sun W.R.T. Star in Gate A.	Biased Aspect Angle of Sun W.R.T. Star in Gate A.	Simulated Drift Rate.
6. SRR - Major Frame Start	TM Determination of Azimuth Reference (Primarily used when in the Subcom. Mode).			

TABLE 3.3.3.8-1

CONTROL UNIT REDUNDANCY MECHANIZATION

Unit	Interfacing Units	Remarks
ADP	COMs	
ADP	DIMs	
ADP	TM Processors	See Section 3.5 for telemetry clock signals.
ADP	Sun Sensors	Each ADP is handwired to a separate set of sun sensors (mid-range and extended range) (no crosstrapping).
ADP	Star Sensor	Each ADP can operate with either star sensor channel (full crosstrapping).
ADP	Scientific Instruments	Each ADP provides separate outputs to each scientific instrument (full crosstrapping).
Star Sensor	COMs	Any discrete command or quantitative command can be sent to either star sensor channel from either COM 1 or COM 2 (full crosstrapping).
Star Sensor	DIMs	• PSI* and PSI2* intensity measurements are telemetered on redundant subcom channels using DIM 6 and DIM 7. PSI* intensity is telemetered in the minor frame mode only using DIM 6 (not redundant). PSI2* intensity is telemetered in the minor frame mode only using DIM 7 (not redundant). • All other measurements are processed in a non-redundant manner with each measurement assigned to a single TM channel.

TABLE 3.3.4-1

COMMAND RESPONSES FOR CONTROL SUBSYSTEM

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
Star Sensor				
STR19 STRA9	PSI* ON	Provides 28 Vdc switched loads bus pwr to the PSI* channel Electronic Control Unit (ECU Regulator) (Figure 3.3.2.3-1) (vertical slit)	A*1ONS	PSI* ON/OFF Status (Bilevel)= Logical "1": indicates an "ON" condition
			A*1THS	PSI* threshold setting indicates 0 Vdc prior to PSI* ON CMD and 0.28 Vdc minimum thereafter. (There is no preferred threshold state due to PSI* ON cmd.)
			ASTART	Star sensor temperature - Rises
			PBUSLI	Spacecraft Loads Current increases 35 milliamperes. Since the minimum resolution for PBUSLI is 72 milliamperes per count, no TM change for PBUSLI is anticipated.
STR1Ø STRAØ	PSI* OFF	Removes 28 Vdc switched loads bus pwr to the PSI* Channel ECU regulator (Figure 3.3.2.3-1)	A*1ONS	PSI* ON/OFF Status (Bilevel)= Logical "0": indicates an "OFF" condition
			A*1THS	PSI* threshold setting indicates 0 Vdc.
			ASTART	Star sensor temperature - decreases.
			PBUSLI	Spacecraft loads current decreases 35 milliamperes. No TM change anticipated.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
STR 29 STRB9	PSI2* ON	Provides 28 Vdc switched loads bus power to the PSI2* Channel ECU Regulator (Figure 3.3.2.3-1) (canted slit)	A*2ONS	PSI2* ON/OFF Status (bilevel) = Logical "1": indicates an "ON" condition.
			A*2THS	PSI2* threshold setting indicates a minimum of 0.28 Vdc (there is no preferred threshold state due to PSI2* ON cmd).
			ASTART	Star sensor temperature - rises
			PBUSLI	Spacecraft loads current increases 35 milliamperes. No TM change is anticipated.
STR2Ø STRBØ	PSI2*OFF	Removes 28 Vdc switched loads bus power to the PSI2* channel ECU regulator (Figure 3.3.2.3-1)	A*2ONS	PSI2* ON/OFF status (bilevel) = Logical "0": indicates an "OFF" condition.
			A*2THS	PSI*2 threshold setting indicates 0 Vdc
			ASTART	Star sensor temperature - decreases
			PBUSLI	Spacecraft loads current decreases 35 mA. No TM change is anticipated.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
STRQ1, STRQA	Star Sensor Threshold and Bandpass Select (Quantitative Command)			
S2GX	PSI2* Threshold = 1-8	"X" determines the threshold setting, 1 through 8, inserted in a register in (Figure 3.3.2.3-2) for detection of stars from -1.0 mag. to +1.2 mag., respectively, for the PSI2* channel. Bits 34, 35, and 36 are the binary states used as follows: 34 35 36 X 0 0 0 1 1 1 0 4 1 1 1 8	A*2THS	PSI*2 threshold setting from 0 to 5 Vdc representing the desired threshold state (X) selected TM (Vdc) State (X) 3.24 1 2.02 2 1.24 3 0.74 4 0.44 5 0.34 6 0.26 7 0.32 8
S1GX	PSI* Threshold = 1-8	"X" determines the threshold setting, 1 through 8, inserted in a register (Figure 3.3.2.3-2) for detection of stars from -1.0 mag to +1.2 mag, respectivelv, for the PSI* channel. Bits 38, 39, and 40 are the binary states used as follows: 38 39 40 X 0 0 0 1 1 1 0 4 1 1 1 8	A*1THS	PSI* threshold setting from 0 to 5 Vdc, representing the desired threshold state (X) selected. TM (Vdc) State (X) 3.64 1 2.30 2 1.44 3 0.90 4 0.56 5 0.46 6 0.38 7 0.30 8

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
STRQ1 STRQA	Star Sensor Threshold and Bandpass Select (Continued)			
S2BWX	Star 2 BP - Hi/Lo	"X" determines bandpass selection inserted in a register (Figure 3.3.2.3-2) for PSI2* channel. CMD Bit 33, Logical "0" indicates 19 Hz bandpass; "1" indicates 2 Hz bandpass. X = H for 19 Hz X = L for 2 Hz	A*2BPS	PSI2* bandpass state status = 1 for 2 Hz selected; = 0 for 19 Hz selected. (19 Hz recommended for all spin rates for nominal mission).
S1BWX	Star 1 BP = Hi/Lo	"X" determines bandpass selection inserted in a register (Figure 3.3.2.3-2) for PSI * channel. Cmd bit 37, Logical "0" indicates 19 Hz bandpass; "1" indicates 2 Hz bandpass X = H for 19 Hz X = L for 2 Hz	A*1BPS	PSI* bandpass state status =1 for 2 Hz selected; =0 for 19 Hz selected. (19 Hz recommended for all spin rates for nominal mission)

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks

Star Sensor Threshold and Bandpass Select Quantitative Command Summary

CMD: STRQ1, S2GX, S1GX, S2BWX, S1BWX (Bits 25 through 40)

STRQA, S2GX, S1GX, S2BWX, S1BWX (Bits 25 through 40)

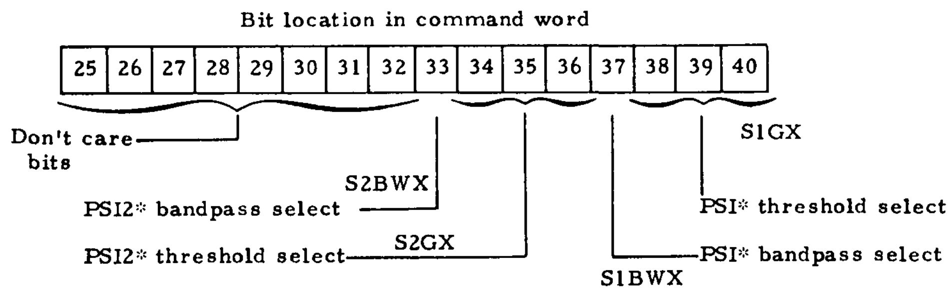


Table 3. 3. 4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATTITUDE DATA PROCESSOR				
ADP19 ADPA9	Attitude Data Processor No. 1 ON	In Figure 3. 3. 2. 3-1, provides spacecraft switched loads bus power to the ECU regulator, thereby activating the onboard ADP electronics except for JCE output buffer and solenoid drivers.	AADPIS	ADP 1 ON/OFF identifies that the ADP 1 is ON when value = 1.
			PBUSLI	Spacecraft loads current increases 250 mA.
ADP1Ø ADPAØ	Attitude Data Processor No. 1 OFF	In Figure 3. 3. 2. 3-1, removes spacecraft switched loads bus power from ADP ECU regulator.	AADPIS	ADP 1 ON/OFF identifies that the ADP 1 is OFF when value = 0.
			PBUSLI	Spacecraft loads current decreases 250 mA.
VAL19 VALA9	Latch Valve No. 1 OPEN	In Figure 3. 3. 2. 3-1, provides propellant to the upstream side of the thruster solenoid assembly for thrusters 3R, 4R, 5A. This thruster combination is a minimum set required to perform spin-up, spin-down, velocity changes, and attitude changes.	VALVIS	Latch valve 1 OPEN/CLOSED identifies that latch valve 1 is OPENED when value = 1.
			PBUSLI	Spacecraft loads current increases 620 mA for 30 milliseconds, then returns to zero.
VAL1Ø VALAØ	Latch Valve No. 1 CLOSED	In Figure 3. 3. 2. 3-1, terminates propellant from to the upstream side of the thruster solenoid assembly for thrusters 3R, 4R, 5A.	VALVIS	Latch valve 1 OPEN/CLOSED identifies latch valve 1 is closed when value = 0.
			PBUSLI	Spacecraft loads current increases 620 mA for 30 milliseconds, then returns to zero.

Table 3. 3. 4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ADP29 ADPB9	Attitude Data Processor No. 2 ON	In Figure 3. 3. 2. 3-1, provides spacecraft switched loads bus power to the ECU regulator thereby activating the onboard ADP electronics except for JCE output buffer and solenoid drivers.	AADP2S	ADP 2 ON/OFF identifies that the ADP 2 is ON when value = 1.
			PBUSLI	Spacecraft loads current increases 250 mA.
ADP2Ø ADPBØ	Attitude Data Processor No. 2 OFF	In Figure 3. 3. 2. 3-1, removes spacecraft switched loads bus power from ADP ECU regulator.	AADP2S	ADP2 ON/OFF identifies that the ADP 2 is OFF when value = 0.
			PBUSLI	Spacecraft loads current decreases 250 mA.
VAL29 VALB9	Latch Valve No. 2 OPEN	In Figure 3. 3. 2. 3-1 provides propellant to the upstream side of the thruster solenoid assembly for thrusters 1R, 2R, and 6A. This thruster combination is minimum set required to perform spin-up, spin-down velocity changes and attitude changes.	VALV2S	Latch valve 2 OPEN/CLOSED identifies that latch valve 2 is opened when value = 1.
			PBUSLI	Spacecraft loads current increases 620 mA for 30 milliseconds, then returns to zero.
VAL2Ø VALBØ	Latch Valve No. 2 CLOSED	In Figure 3. 3. 2. 3-1, terminates propellant fromto the upstream side of the thruster solenoid assembly for thrusters 1R, 2R, and 6A.	VALV2S	Latch valve 2 OPEN/CLOSED identifies that latch valve 2 is closed when value = 0.
			PBUSLI	Spacecraft loads current increases 620 mA for 30 milliseconds, then returns to zero.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ1 ATQØA	Measurement Select		ATTM1Z ATTM2Z	Attitude measurement - these two 8-bit groups (ATTM1Z is the 8 LSB's group; ATTM2Z is the 8MSB's group) combine to form the 16 bit word representing either of the two time measurements selected ahead. The range is 0 to 16.38 seconds in 0.25 ms increments. Measurements alternate between the two measurements selected as A and B (or for 1 and 2). However, if the new measurement is not complete, the last measurement will be read out again.
			ATTMSS	Data measurement = when = 1, that measurement selected for MEA1 is telemetered in ATTM1Z and ATTM2Z. When = 0, that measurement selected for MEA2 is telemetered in ATTM1Z and ATTM2Z.
MEA1= SRR	Measurement A select: SRR-SRR	Start and stop of 16 bit counter selected to be successive occurrences of SRR. The spin period measurement is generated with the selected roll reference (SRR) being either the ψ, ψ^*, ψ_2^* or simulated SRR as the primary input to the phase lock loop (PLL)	AM1ADS	Measurement A address = binary 000: "SRR to SRR" spacecraft spin period measurement selected.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ1 ATQØA	Measurement Select (Continued)			
MEA1= MFS	Measurement A Select: "SRR - Minor F Start"	Start and stop of 16 bit counter selected to be occurrences of SRR and start of telemetry minor frame, respectively.	AM1ADS	Measurement A address = binary 100: "SRR - Minor Frame Start" selected. Time between occurrence of SRR and start of telemetry minor frame. Used with the minor frame mode TM format. Used for inertial determination of the SRR with respect to any and all telemetry bit stream word locations.
MEA1= RIP	Measurement A Select: "SRR - RIP"	Start and stop of 16 bit counter selected to be occurrences of SRR and RIP, respectively.	AM1ADS	Measurement A address = binary 010: "SRR - RIP" selected. Time between occurrence of SRR and occurrence of roll index pulse, (used for spin angle (azimuth) determination of the roll index pulse with respect to the SRR).

Table 3. 3. 4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ1 ATQØA	Measurement Select (Continued)			
MEA1= GSB (Gated Star B)	Measurement A Select: "SRR - Gated Star B"	Start and stop of 16 bit counter selected to be occurrences of SRR and gated star B, respectively.	AM1ADS	Measurement A address = 001: "SRR - Gated Star B" selected. Time between occurrence of SRR and appearance of a star above the commanded threshold setting within $\pm 5.625^{\circ}$ of the center of Gate B. (Used for spin angle determination of the first gated star B with respect to the SRR).
MEA1= S2S (PSI2)	Measurement A Select: "SRR - PSI2"	Start and stop of 16 bit counter selected to be occurrences of SRR and appearance of sun in PSI2 window, respectively.	AM1ADS	Measurement A address = 101: "SRR - PSI2" selected. Time between occurrence of SRR and appearance of sun in PSI2 sensor window. Depending on the selected roll ref., the SRR - PSI2 measurement will represent the following: SRR — Measurement ψ — Sun aspect angle ψ^* — Sun - star separation ψ_2^* — Sun - star separation Sim SRR — Free run mode drift rate and/or sim spin period verification with actual rotor spin rate.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ1 ATQØA	Measurement Select (Continued)			
MEA1= MAF	Measurement A Select: "SRR - Major F Start"	Start and stop of 16 bit counter selected to be occurrence of SRR and start of telemetry major frame, respectively.	AM1ADS	Measurement A address = 011: "SRR - Major Frame Start" selected. Time between occurrence of SRR and occurrence of start of telemetry major frame. Used primarily with the subcom mode format. Used for inertial determination (azimuth) of the SRR with respect to any and all telemetry bit stream word locations.
MEA2= SRR	Measurement B Select: SRR-SRR	Start and stop of 16 bit counter selected to be successive occurrences of SRR. The spin period measurement is generated with the selected roll reference (SRR) being either the ψ, ψ^*, ψ_2 or simulated SRR as the primary input to the phase lock loop (PLL)	AM2ADS	Measurement B address = binary 000: "SRR to SRR" spacecraft spin period measurement selected.
MEA2= MFS	Measurement B Select: "SRR - Minor F Start"	Start and stop of 16 bit counter selected to be occurrences of SRR and start of telemetry minor frame, respectively.	AM2ADS	Measurement B address = binary 100: "SRR - Minor Frame Start" selected. Time between occurrence of SRR and start of telemetry minor frame. Used with the minor frame mode TM format. Used for inertial determination of the SRR with respect to any and all telemetry bit stream word locations.)

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ1 ATQØA	Measurement Select (Continued)			
MEA2= RIP	Measurement B Select: "SRR - RIP"	Start and stop of 16 bit counter selected to be occurrences of SRR and RIP, respectively.	AM2ADS	Measurement B address = binary 010: "SRR - RIP" selected. Time between occurrence of SRR and occurrence of roll index pulse (used for spin angle (azimuth) determination of the roll index pulse with respect to the SRR)
MEA2= GSB (Gated Star B)	Measurement B Select: "SRR - Gated Star B"	Start and stop of 16 bit counter selected to be occurrences of SRR and gated star B, respectively.	AM2ADS	Measurement B address = 001: "SRR - Gated Star B" selected. Time between occurrence of SRR and appearance of a star above the commanded threshold setting within ±5.625° of the center of gate B. (Used for spin angle determination of the first gated star B with respect to the SRR).

Table 3. 3. 4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ1 ATQØA	Measurement Select (Continued)			
MEA2= S2S (PSI2)	Measurement B Select: SRR - PSI2	Start and stop of 16 bit counter selected to be occurrences of SRR and appearance of sun in PSI2 window, respectively.	AM2ADS	Measurement B address = 101: "SRR - PSI2" selected. Time between occurrence of SRR and appearance of sun in PSI2 sensor window. Depending on the selected roll ref., the SRR - PSI2 measurement will represent the following: SRR — Measurement ψ — Sun aspect angle ψ^* — Sun-star separation ψ_2^* — Sun-star separation Sim SRR — Free run mode drift rate and/or sim spin period verification with actual rotor spin rate.
MEA2= MAF	Measurement B Select: "SRR - Major F Start"	Start and stop of 16 bit counter selected to be occurrence of SRR and start of telemetry major frame, respectively.	AM2ADS	Measurement B address = 011: "SRR - Major Frame Start" selected. Time between occurrence of SRR and occurrence of start of telemetry major frame. Used primarily with the sub com mode format. Used for inertial determination (azimuth) of the SRR with respect to any and all telemetry bit stream word locations.)

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
<u>ADP Measurement Select Quantitative Command Summary</u>				

CMD: ATQØ1, MEA1 = XXX, MEA2 = XXX (ADP No. 1 Only)
ATQØA, MEA1 = XXX, MEA2 = XXX (ADP No. 2 Only)

Where XXX = SRR or MFS or RIP or GSB or S2S or MAF

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
Address 0 0 0 0				Don't Care Bits						MEA1 = XXX			MEA2 = XXX		

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ2 ATQØB	Jet Control			
R1X	Jet Select - Jet 1R	X determines the Enabled/Disabled state for radial jet 1R X = E for ENABLE; D for Disable. In Figure 3.3.2.3-1, the 1R driver amplifier is enabled or disabled	AJET1S	Jet select bits: Bit 1 (radial jet 1R status) = binary 1 for enabled, = 0 for disabled.
R2X	Jet Select - Jet 2R	X determines the Enabled/Disabled state for radial jet 2R. X = E for ENABLE; X = D for Disable. In Figure 3.3.2.3-1, the 2R driver amplifier is enabled or disabled.	AJET2S	Jet select bits: Bit 2 (radial jet 2R status) = binary 1 for enabled, = 0 for disabled.
R3X	Jet Select - Jet 3R	X determines the Enabled/Disabled state for radial jet 3R. X = E for ENABLE; X = D for DISABLE. Jet select in Figure 3.3.2.3-1, the 3R driver amplifier is enabled or disabled.	AJET3S	Jet select bits: Bit 3 (radial jet 3R status) = binary 1 for enabled, = 0 for disabled.
R4X	Jet Select - Jet 4R	X determines the Enabled/Disabled state for radial jet 4R. X = E for ENABLE; X = D for DISABLE. In Figure 3.3.2.3-1, the 4R driver amplifier is enabled or disabled.	AJET4S	Jet select bits: Bit 4 (radial jet 4R) status) = binary 1 for enabled, = 0 for disabled.
A5X	Jet Select - Jet 5A	X determines the Enabled/Disabled state for axial jet 5A. X = E for ENABLE; X = D for DISABLE. In Figure 3.3.2.3-1, the 5A driver amplifier is enabled or disabled.	AJET5S	Jet select bits: Bit 5 (axial jet 5A status) = binary 1 for enabled, = 0 for disabled.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ2 ATQØB	Jet Control (Continued)			
A6X	Jet Select - Jet 6A	X determines the Enabled/Disabled state for axial jet 6A. X = E for ENABLE; X = D for DISABLE. In Figure 3.3.2.3-1, the 6A driver amplifier is enabled or disabled.	AJET6S	Jet select bits: Bit 6 (axial jet 6A status) = binary 1 for enabled, = 0 for disabled.
FMX	Normal/Alternate Fire Mode	X determines selection of normal/alternate fire mode for thruster firing. Normal mode provides one thruster pulse per spacecraft spin revolution. Alternate mode provides two thruster pulse firings, separated by 180^o, per spacecraft spin revolution. X = N for Normal; X = A for Alternate. Enable signal is provided to 0^o AND gate only OR to (0^o and 180^o) AND gate in Figure 3.3.2.3-1.	AJETMS	Normal/alternate fire mode = 1 for normal select; = 0 for alternate select.
P/X	Pulse Width Select	Two selectable pulse widths, 128 ms or 512 ms, determined by X selection as follows: X — Pulse Duration 2 — 512 ms = 1/2 sec 8 — 128 ms = 1/8 sec Enable signal is provided to 128 ms AND gate OR to 512 ms AND gate in Figure 3.3.2.3-1.	APULLS	Pulse width select 512/128 ms = 1 for 512 ms selected; = 0 for 128 ms selected.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ2 ATQØB	Jet Control (Continued)			
CX	Pulse/Time Count Select	The firing duration is controlled by either counting time or the number of thruster pulse firings. This selection is independent of pulse/continuous firing mode selection, i.e., pulse Firing mode and time count can both be selected. X = P for pulse count selected; X = T for time count selected. Enable signal is provided to pulse count AND gate OR time count AND gate in Figure 3.3.2.3-1.	AMAGCS	Pulse/time count select = 1 for pulse count selected; = 0 for time count selected.
FX	Continuous/Pulse Fire Select	The thruster is fired either continuously or as a pulse. Pulse firing selection is further predicated on normal/alternate mode selection and pulse duration selection. X = T for continuous fire select; X = P for pulse fire select. Enable signal is provided to PULSE FIRE AND gate OR to a TIME FIRE AND gate in Figure 3.3.2.3-1.	AJETMS	Continuous/pulse fire select = 1 for continuous selected; = 0 for pulse selected.
SRDX	Spin Rate Detector Enable/Inhibit	Provides automatic thruster turn-off, if enabled, upon detection of spin speeds less than or equal to 3.7 rpm, and greater than or equal to 67 rpm. X = E for ENABLE selected; X = D for INHIBIT selected. Enable signal is or is not provided to spin rate detector in Figure 3.3.2.3-1.	ASDETS	Spin rate detector Enable/Inhibit = 1 for Enabled; = 0 for Inhibited.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ADP JET CONTROL QUANTITATIVE COMMAND SUMMARY CMD: ATQØ2, R1X, R2X, R3X, R4X, A5X, A6X, FMX, P/X, CX, FX, SRDX (ADP No. 1 Only) ATQØB, R1X, R2X, R3X, R4X, A5X, A6X, FMX, P/X, CX, FX, SRDX (ADP No. 2 Only) Bits 25 26 27 28: Address 1 0 0 0 Bits 29 30 31 32 33 34 35: Thruster Selection R1X, R2X, R3X, R4X, A5X, A6X, Bit 36: Normal/Alternate Fire Mode, FMX Bit 37: Pulse Width Select, P/X Bit 38: Pulse/Time Count, CX Bit 39: Continuous/Pulse, FX Bit 40: Spin Rate detector, SRDX				
ATQØ3 ATQØC	ADP Mode Select			
SRXX	PLL Spin Range - Select	Provides for wide PLL dynamic range in four discrete steps determined by XX selection as follows: XX — Range Selected 04 — 1) 4-9 rpm 08 — 2) 8-18 rpm 16 — 3) 16-35 rpm 32 — 4) 32-71 rpm Selected range must encompass actual spacecraft rotor rate for proper acquisition and tracking. One of four countdown durations is selected in a variable countdown circuit in Figure 3.3.2.3-1.	ASPINS	PLL spin range select: identifies selected spin range as follows: State of TM word 5, bits (0, 1) — Range (rpm) 0 0 — 4-9 1 0 — 8-18 0 1 — 16-35 1 1 — 32-71

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ3 ATQØC	ADP Mode Select (Continued)			
PLX	PLL Loss of Lock Enable/Inhibit	In Figure 3.3.2.3-1, selecting the PLL loss of lock enabled provides for automatic transfer of SRR to simulated SRR and JCE turn-off, upon detection of two consecutive missing SRR pulses. Enabled, the PLL loss of lock prevents the PLL frequency from being updated if the SRR pulse is outside gate A. Inhibited, the loss of lock detection circuit is forced to an in-lock condition independent of whether or not the SRR pulse is within gate A. If the SRR pulse occurs within gate A the PLL will maintain frequency and phase to keep the SRR pulse in the center of gate A. If a pulse occurs outside gate A when in the gated mode (star or gated sun mode) the loop will not be corrected and will maintain phase and frequency to within the drift characteristics of the PLL circuitry. If a pulse occurs outside gate A in the ungated sun mode the loop performs as in the sun acquisition mode and will slew in frequency and phase to lock onto detected pulse. A count of the number of missing SRR pulses is recorded. X = E for enabled; X = D for inhibited.	ALOLES	PLL loss of lock enable/inhibit = 1 if enabled; = 0 if inhibited.
			ALOCKS	PLL loss of lock: With PLL loss of lock enabled, bit will = 1 if SRR is the PLL input; bit will = 0 if input is transferred to SIM SRR. With PLL loss of lock disabled, bit will always be = 1.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ3 ATQØC	ADP Mode Select (Continued)			
SGAX	Star Gate A Channel Select	In Figure 3.3.2.3-1, selects either the ψ^* channel or ψ_2^* channel output as the Gated A star output. If star acquisition CMD (STX) and SRR advance CMD (SRAX) are also commanded then gate A is centered over first star in the spin revolution following receipt of star advance CMD that is above the selected threshold setting. Gate A is 45^{o} until star normal is commanded, when it becomes 11.25^{o}. X = 1 for ψ^* channel selected; = 2 for ψ_2^* channel selected.	A*GASS	Star Gate A; channel 1/2 select = 1 if ψ^* channel is selected; = 0 if ψ_2^* channel is selected.
SGBX	Star Gate B Channel Select	In Figure 3.3.2.3-1, selects either the ψ^* channel for ψ_2^* channel as the gated B star output for attitude measurements. X = 1 for ψ^* channel select. = 2 for ψ_2^* channel selected.	A*GBSS	Star Gate B; channel 1/2 select = 1 if ψ^* channel is selected; = 0 if ψ_2^* channel is selected
SSRX	Sun Sensor (Range) Select	In Figure 3.3.2.3-1, provides for selection of one of the three sun sensor viewing ranges as follows: Viewing Range / X / State of CMD Bits 34, 35 Mid / M / 00 Upper / U / 10 Lower / L / 01	ASUNSS	Sun Sensor Select - State of TM Word 4, Bits 3 & 4 / Indicates Selected State 00 / Mid 10 / Upper 01 / Lower

Table 3. 3. 4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ3 ATQØC	ADP Mode Select (Continued)			
REFXXX	Sun/Star/Simulated Select	In Figure 3. 3. 2. 3-1, provides for selection of one of three roll references (sun, star, or simulated SRR) as the primary input to PLL. XXX = SUN OR = STR (STAR) OR = SIM (SIMULATED)	ASRRMS	Identifies reference selection as follows: Ref — State of TM Word 4, Bits 5 and 6 Sun — 1 1 Star — 0 1 Simulated SRR — (1 or 0) 0
STX	Star Acquisition/ Normal	In Figure 3. 3. 2. 3-1, provides for PLL frequency and phase locking to any star in the spin revolution that is capable of being detected by the star sensor threshold setting. The normal (11. 25°) gate A is expanded to 45° when star acquisition is selected, to allow for the frequency difference between PLL and the actual rotor rate to be slightly in excess of ±1. 5%. This CMD is used in conjunction with the SRR advance CMD (SRAX) X = A for acquisition selected; = N for normal selected.	A*ACQS	Star acquisition/normal = 1 if star acquisition is selected; = 0 if normal is selected
SUGX	Sun Gate Enable/Disable	In Figure 3. 3. 2. 3-1, selecting the enabled state provides a "gate A" to be centered over the sun SRR input. This allows the detection of the number of missing sun pulses, as well as automatic switching to simulated SRR upon detection of two consecutive missing sun SRRs. If Disabled state is selected, the	ASUNGS	Enable/Disable Sun Gate = 1 if enabled; = 0 if disabled.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ3 ATQØC	ADP Mode Select (Continued)			
SUGX (cont)		PLL is forced to frequency search and phase lock to the sun at all times. x = E for Enable selected; = D for Disable selected.		
SRAXXX	SRR Advance	In Figure 3.3.2.3-1, allows PLL reference switching from sun to star, or from star to star, or from star to sun or from simulated SRR to sun; and, in rare cases when switching from simulated SRR to star (called star acquisition when sun is not available.). XXX = ADV if advance state is selected; = NRM, if normal state is selected.	ADVANS	•SRR Advance Status = 1 if Advance state is selected = 0 if normal state is selected • If Advance state had been commanded, the state changes automatically from Advance to Normal once a new SRR pulse is detected and the phase is reset.

Table 3.3.4-1 (continued) ADP MODE SELECT QUANTITATIVE COMMAND SUMMARY

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks

ADP MODE SELECT QUANTITATIVE COMMAND SUMMARY

CMD: ATQØ3, SRXX, PLX, SGAX, SGBX, SSRX, REFXXX, STX, SUGX, SRAXXX. (ADP No. 1 Only)

ATQØC, SRXX, PLX, SGAX, SGBX, SSRX, REFXXX, STX, SUGX, SRAXXX. (ADP No. 2 Only)

Bit location in command word

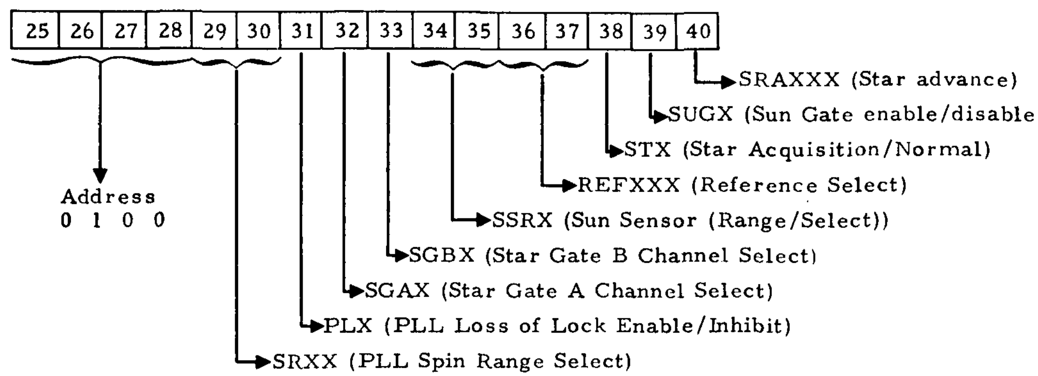


Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ4 ATQØD	Jet Countdown	In Figure 3.3.2.3-1, provides firing duration of selected thrusters in terms of time or number of pulse firings as follows (12 bits): 0 - 2096.64 sec in 0.512 sec steps <u>or</u> 0 - 4095 pulses.	AJMAGC	JCE Countdown (12 bits TM word) - identifies the Jet Countdown Command Register contents at all times. (Stops at maximum count of all "1"s)
ATQØ5 ATQØE	ACS Angle Delay Magnitude	In Figure 3.3.2.3-1, provides a phase angle delay with respect to the SRR for gated Star B control and Jet Fire Angle Control. Angle delay capability from 0.352° to 360.0° in increments of 0.3515625° is provided. <u>Actual ACS Angle Delay is 0.352° or 1 count larger than the commanded value.</u>	AACSAD	ACS Angle Delay Magnitude (10 bits) TM word) - identifies the ACS Angle Delay countdown register contents at all times. (Same range and increment as for command.)
ATQØ6 ATQØF	Roll Index Delay Magnitude	In Figure 3.3.2.3-1, provides a phase angle delay with respect to the SRR for Science and for Small Probes' release. <u>Actual Delay equals the number of Magnitude Counts Commanded.</u> Spin Angle delay capability of 0° to 359.65° in 0.351° steps is provided.	ARIPAD	Roll Index Delay Magnitude (10 bits TM word) - identifies the selected RIP Phase Angle delay. (Same range and increment as for command.)
ATQØ7 ATQØG	PLL Spin Period, Magnitude (MSBs)	In Figure 3.3.2.3-1, controls the eight most significant bits of a 16 bit counter that represents the onboard free-run facsimile of actual spacecraft rotor rate. Commandable period is 0 to 16.32 sec in increments of 0.064 sec.	ASIMSZ	PLL Spin Period Magnitude (8 MSBs): Identifies the 8 MSBs of the Selected Spin Period Control.

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQØ8 ATQØH	PLL Spin Period, Magnitude (LSBs)	In Figure 3.3.2.3-1, controls the eight least significant bits of a 16 bit counter that represents the onboard free run facsimile of the actual spacecraft rotor rate. Commandable period is 0 to 0.06375 sec in 0.25 ms increments.	ASIMSZ	PLL Spin Period Magnitude (8 LSBs): identifies the 8 LSBs of the selected Spin Period Control.
ATQØ9 ATQØI	JCE Buffer Output Enable	In Figure 3.3.2.3-1, provides +15V to the JCE buffers that drive the solenoid drivers. JCE buffer enable is a necessary condition for any thruster firing. Receipt of the JET fire interlock and JET fire CMD in that order, without an intervening JCE buffer disable CMD, is the second necessary condition for any thruster firing.	AJCE1S	Jet Control Electronics 1 Buffer Output State
			AJCE2S	Jet Control Electronics 2 Buffer Output State
				Each above identifies (1=ON) JCE output power is "ON."
ATQ1Ø ATQØJ	JCE Power Buffer Output Disable	In Figure 3.3.2.3-1, removes +15V from the JCE buffers that drive the solenoid driver. JCE buffer disable or more appropriately called JCE "OFF", prevents any thruster firing after CMD receipt. This command also resets the JET fire interlock latch and the JET fire enable latch.	AJCE1S	Jet Control Electronics 1 Buffer Output State
			AJCE2S	Jet Control Electronics 2 Buffer Output State
				Each above identifies (0=OFF) JCE output power is "OFF"
			AJCEFS	Jet Fire enable status = Disabled (= 0).

Table 3. 3. 4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
ATQ11 ATQØK	Jet Fire Interlock	In Figure 3. 3. 2. 3-1, sets the Jet Fire interlock latch. This interlock logic does not allow the jets to fire until both the jet fire interlock CMD and the jet fire CMD have been received in that order without an intervening JCE Buffer Disable CMD. There is no time lapse requirement between the jet fire interlock CMD and the jet fire CMD.		There is no TM that directly indicates the state of the Jet Fire Interlock.
ATQ12 ATQØL	Jet Fire	In Figure 3. 3. 2. 3-1, sets the jet fire enable latch. In the continuous fire mode, receipt of the jet fire CMD (assuming interlock sequence is complete) starts thruster firing. In the pulse mode, the leading edge of the first ACS delay pulse after receipt of jet fire CMD starts the thruster firing.	AJCEFS	Jet fire enable status: Identifies thruster firing sequence is activated when value = 1.
				Each thruster valve requires a maximum of 21 watts to open. This power consumption may be detected via PLIMTI (bus voltage limiter current) when the bus limiters are dissipating excess spacecraft power; or may be detected via PBUSLI (Spacecraft Loads Current).

ADP ATQØ4 THROUGH ATQ12 QUANTITATIVE COMMAND SUMMARY

CMD: ATQØ4 (ADP No. 1 Only)
ATQØD (ADP No. 2 Only)

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40

Address (bits 25–28): 1 1 0 0

Magnitude Information (bits 29–40): Jet Countdown

Table 3. 3. 4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks

ADP ATQØ4 THROUGH ATQ12 QUANTITATIVE COMMAND SUMMARY

CMD: ATQØ5 (ADP No. 1 Only)
ATQØE (ADP No. 2 Only)

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
Address 0 1 0 0				Magnitude Information ACS Delay										Don't Care	

CMD: ATQØ6 (ADP No. 1 Only)
ATQØF (ADP No. 2 Only)

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
Address 1010				Magnitude Information Roll Index Delay										Don't Care	

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks

ADP ATQØ4 THROUGH ATQ12 QUANTITATIVE COMMAND SUMMARY

CMD: ATQØ7 (ADP No. 1 Only)
ATQØG (ADP No. 2 Only)

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
Address 0 1 1 0				Magnitude Information PLL Spin Period, MSBs								Don't Care			

CMD: ATQØ8 (ADP No. 1 Only)
ATQØH (ADP No. 2 Only)

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
Address 1 1 1 0				Magnitude Information PLS Spin Period, LSBs								Don't Care			

CMD: ATQØ9 (ADP No. 1 Only)
ATQØI (ADP No. 2 Only)

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
Address, JCE Buffer Enable 0 0 0 1				Don't Care											

Table 3.3.4-1 (continued)

CMD	Command Title	Command Description & Subsystem Internal Response	TM Mnemonic	TM Title and/or Remarks
	ADP ATQØ4 THROUGH ATQ12 QUANTITATIVE COMMAND SUMMARY			
CMD:	ATQ10 (ADP No. 1 Only) ATQØJ (ADP No. 2 Only)			

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
Address, JCE Buffer Disable				Don't Care											
1	0	0	1												

CMD	Command Title			
CMD:	ATQ11 (ADP No. 1 Only) ATQØK (ADP No. 2 Only)			

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
Address, JCE Fire Interlock				Don't Care											
0	1	0	1												

CMD	Command Title			
CMD:	ATQ12 (ADP No. 1 Only) ATQØL (ADP No. 2 Only)			

Bit location in command word

25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40
Address, Jet Fire				Don't Care											
1	1	0	1												

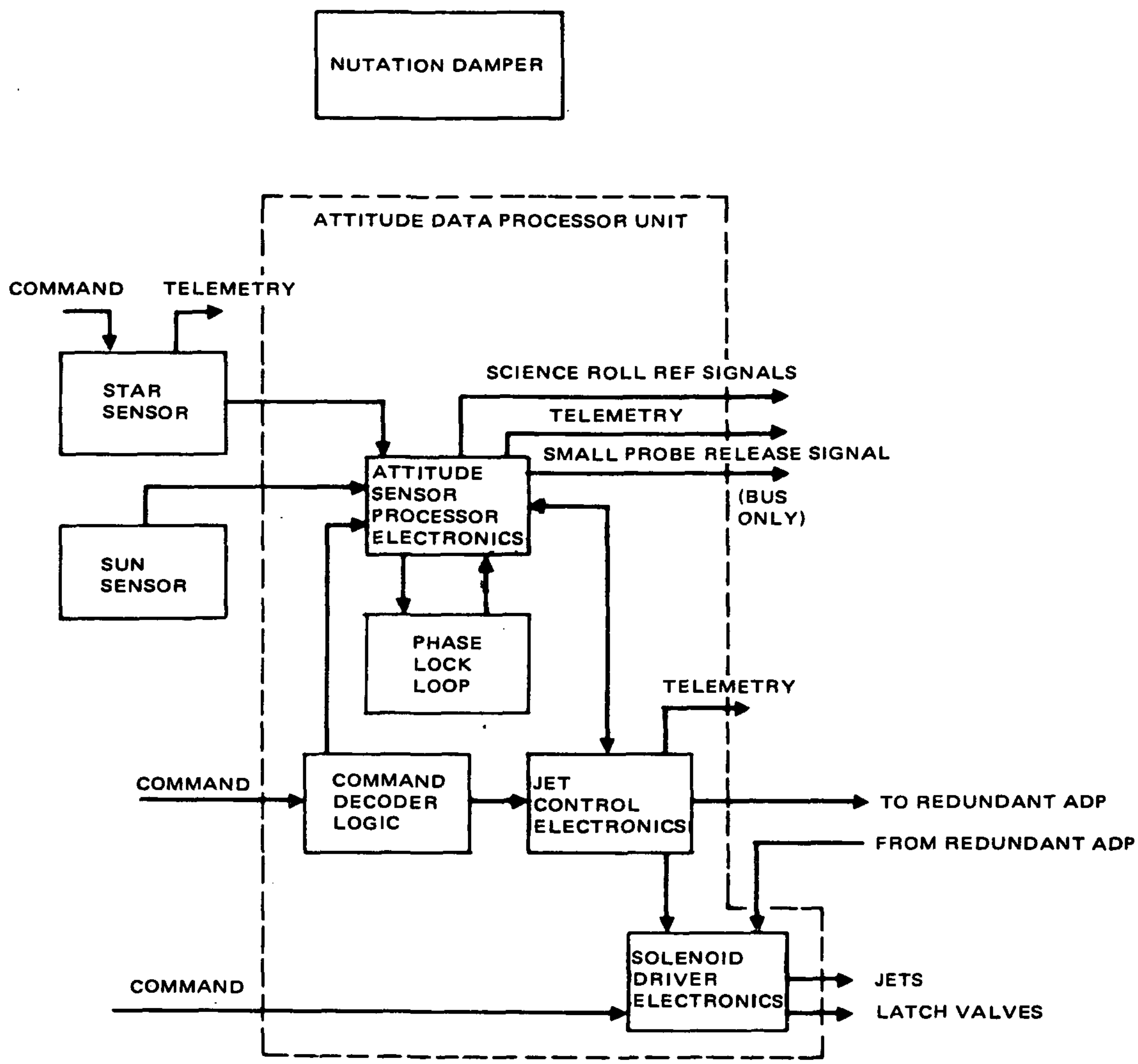
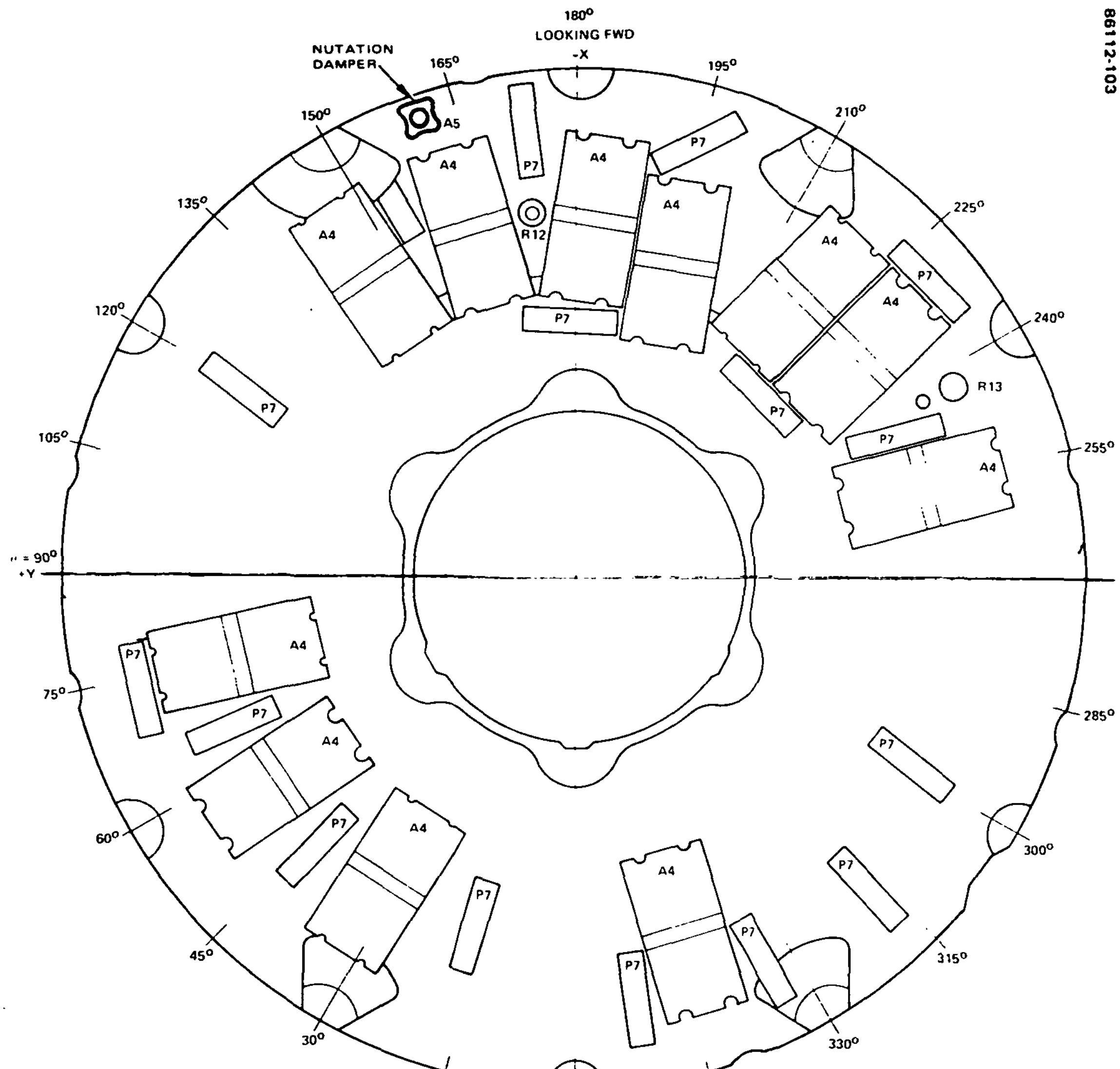
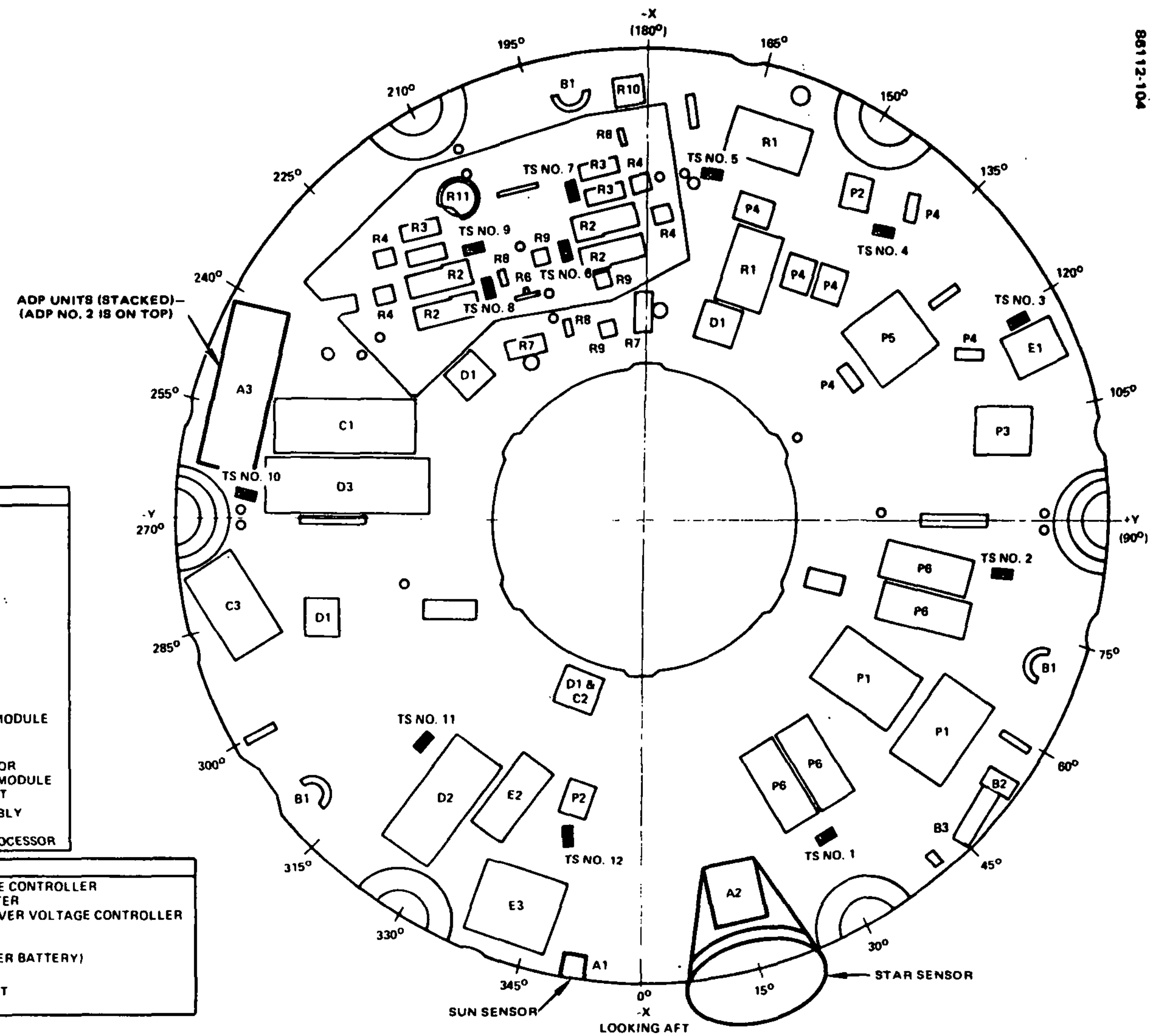


FIGURE 3.3.1-1. CONTROLS SUBSYSTEM BLOCK DIAGRAM



CODE	NOMENCLATURE
A4	THERMAL LOUVERS
A5	NUTATION DAMPER
P7	HEATERS
R12	AFT OMNI MAST
R13	MEDIUM GAIN HORN SUPPORT

FIGURE 3.3.1-2. BUS SYSTEM–SHELF LOCATION OF NUTATION DAMPER



NO.	MNEMONIC
TS1	SHL01T
2	SHL02T
3	SHL03T
4	SHL04T
5	SHL05T
6	SHL06T
7	SHL07T
8	SHL08T
9	SHL09T
10	SHL10T
11	SHL11T
12	SHL12T

CODE	NOMENCLATURE
E1	BIMS SENSOR
E2	BNMS ELECTRONICS
E3	BUMS SENSOR
R1	TRANSPONDER
R2	POWER AMP
R3	DRIVER AMP
R4	ISOLATOR
R5	6 DB HYBRID
R6	3 DB HYBRID
R7	DIPLEXER
R8	S-SWITCH
R9	C-SWITCH
R10	SWITCH DRIVER
R11	FWD OMNI MAST
D1	DUAL DATA INPUT MODULE
D2	PCM ENCODER
D3	TM PROCESSOR
C1	COMMAND PROCESSOR
C2	COMMAND OUTPUT MODULE
C3	PYRO CONTROL UNIT
A1	SUN SENSOR ASSEMBLY
A2	STAR SENSOR
A3	ATTITUDE DATA PROCESSOR

CODE	NOMENCLATURE
P1	CHARGE/DISCHARGE CONTROLLER
P2	BUS VOLTAGE LIMITER
P3	UNDER VOLTAGE/OVER VOLTAGE CONTROLLER
P4	CURRENT SENSOR
P5	POWER INTERFACE
P6	BATTERY PACK (2 PER BATTERY)
B1	IFD BRACKET
B2	UMBILICAL BRACKET
B3	OEP/BSP BRACKET

FIGURE 3.3.1-3. BUS SYSTEM—SHELF LAYOUT OF SOME CONTROLS SUBSYSTEM UNITS

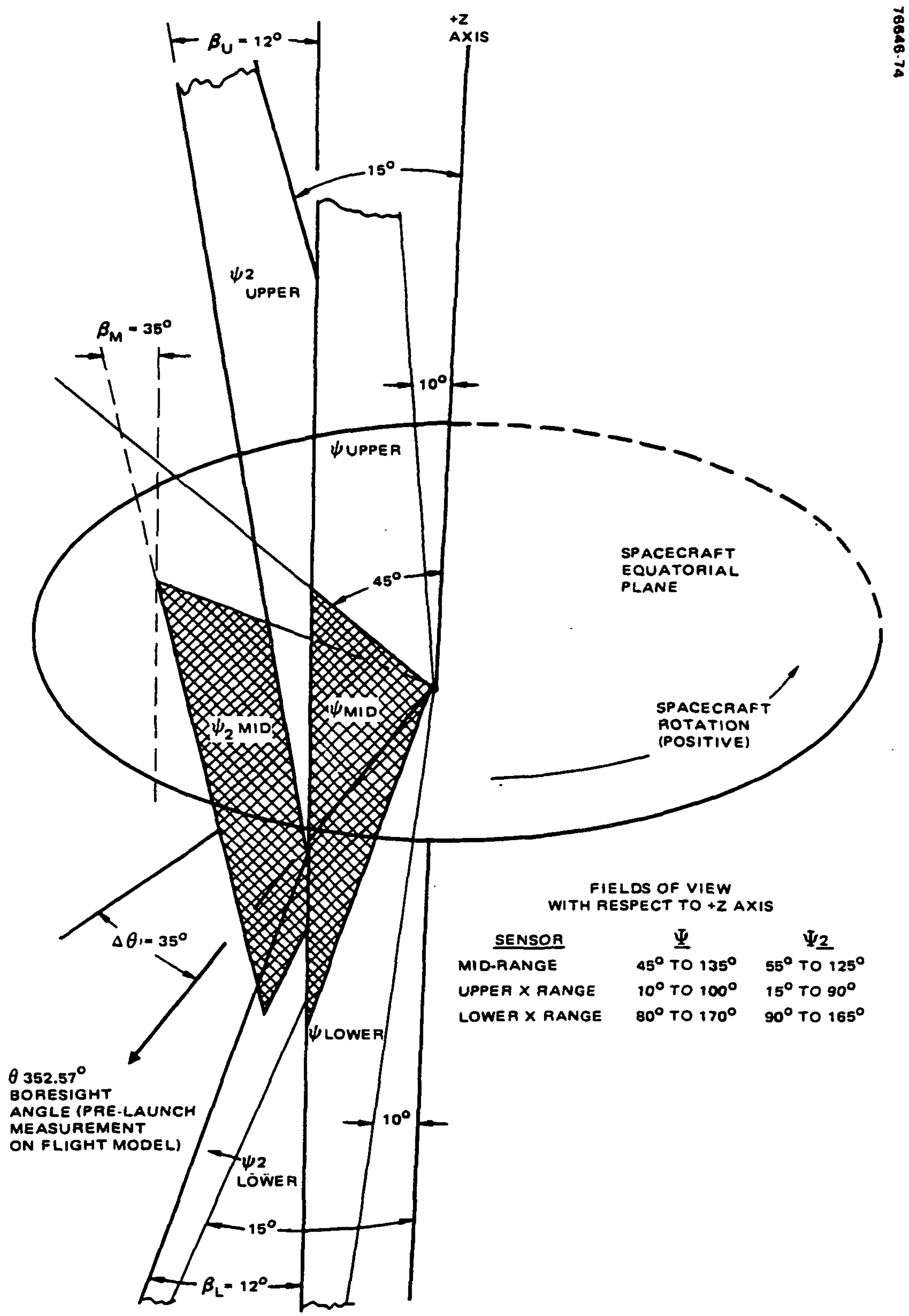
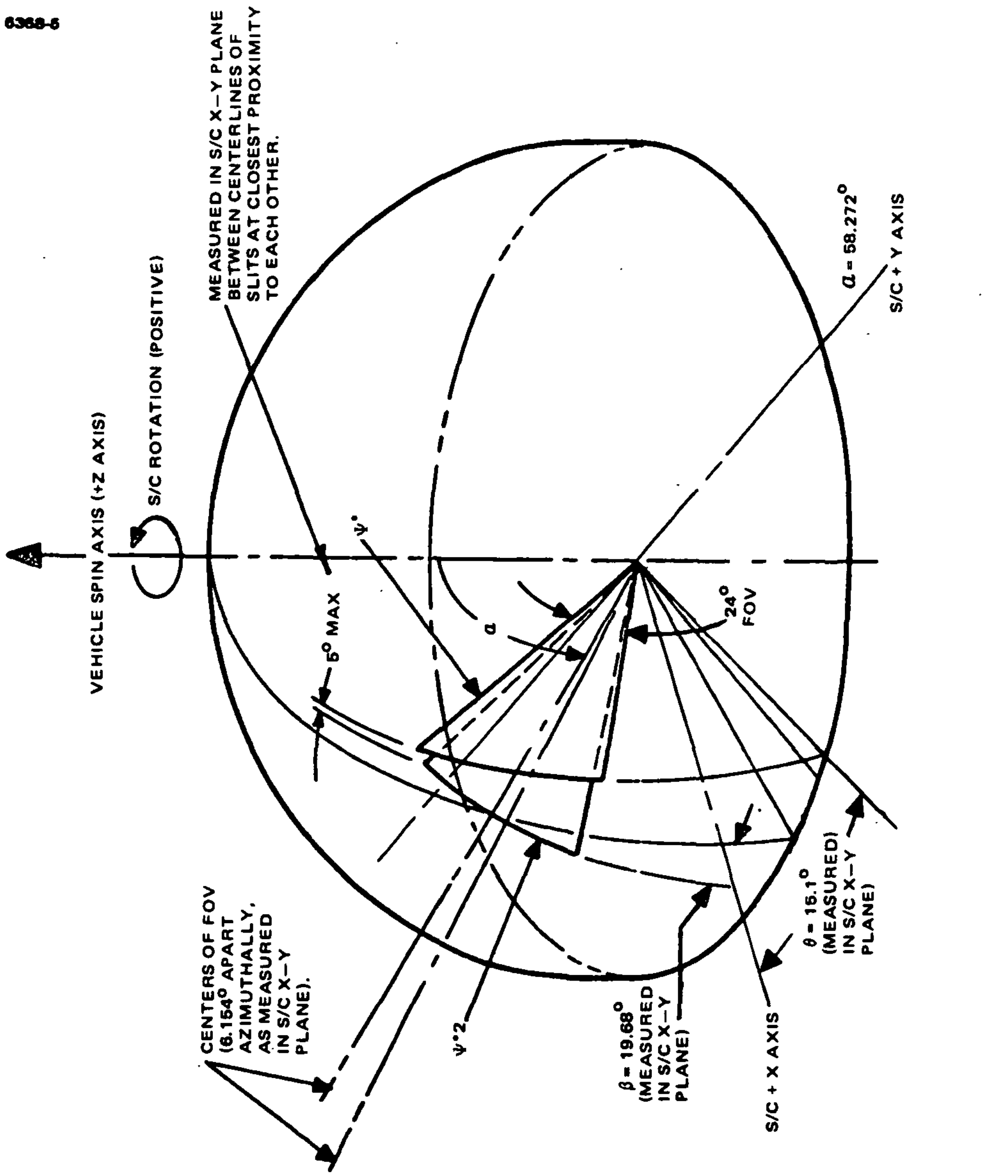
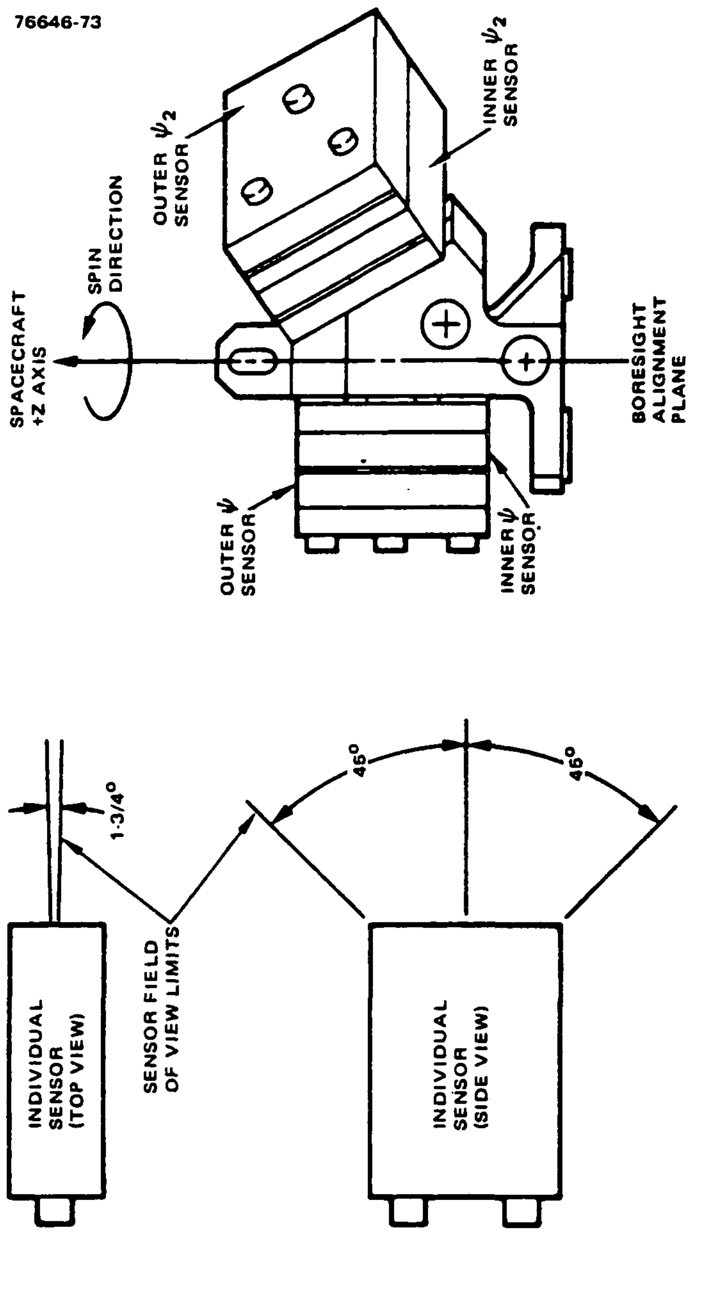


FIGURE 3.3.1-4. SUN SENSORS FIELD OF VIEWS



BORESIGHT OF STAR SENSOR IS TRAILING EDGE OF PSI 2° SLIT AT THE CENTER OF ELEVATION FOV.

FIGURE 3.3.1-5. STAR SENSOR FIELD-OF-VIEW

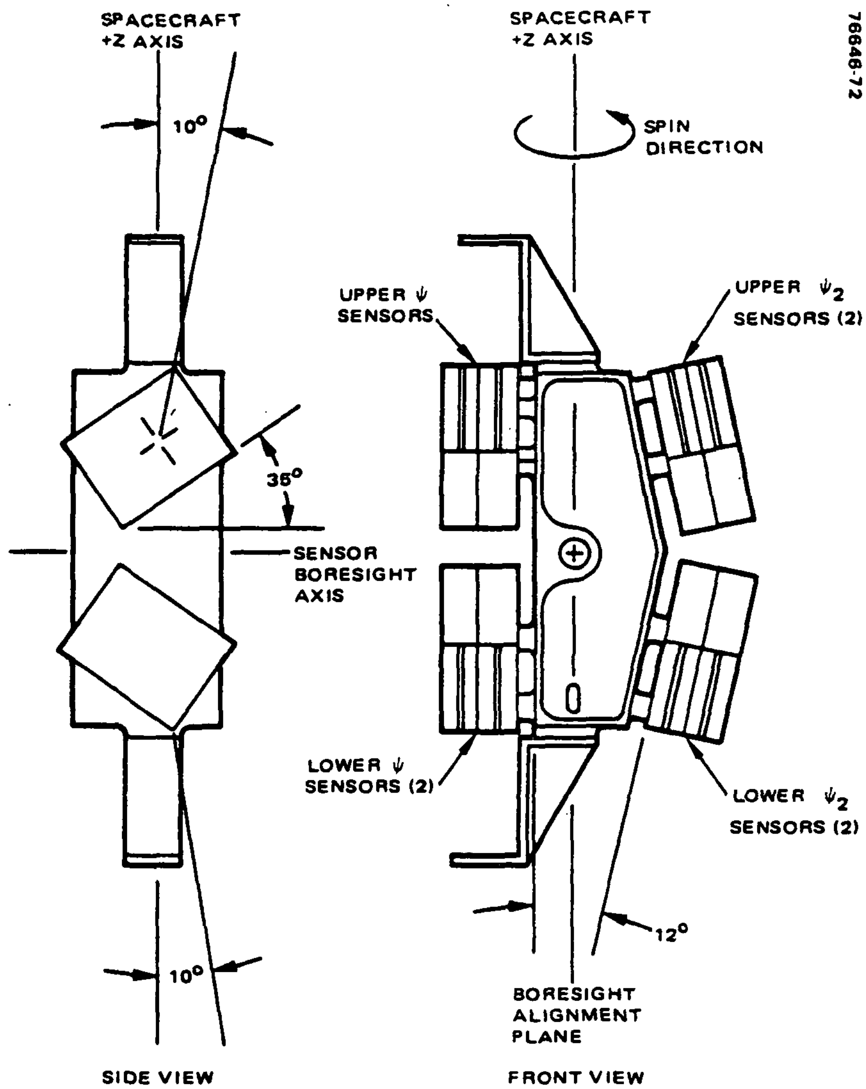


"$\psi - \psi_2$" = $\sin^{-1}$ [$\cot \gamma \tan 35^\circ$] + 35°

WHERE "$\psi - \psi_2$" EQUALS THE NOMINAL SPIN ANGLE BETWEEN ψ AND ψ_2 PULSES.

(35° TERM REFLECTS 35° BIAS SEEN FOR MID-RANGE ψ_2 IN FIGURE 3.3.2.1-5.)

FIGURE 3.3.2.1-1. MID-RANGE SUN SENSOR GEOMETRY



FOR UPPER SENSORS:

"$\psi - \psi_2$" = $\sin^{-1}$ [$\cot \gamma \tan 12°$]

FOR LOWER SENSORS:

"$\psi - \psi_2$" = $-\sin^{-1}$ [$\cot \gamma \tan 12°$]

WHERE "$\psi - \psi_2$" EQUALS NOMINAL SPIN ANGLE BETWEEN ψ AND ψ_2 PULSES

FIGURE 3.3.2.1-2. EXTENDED RANGE SUN SENSOR GEOMETRY

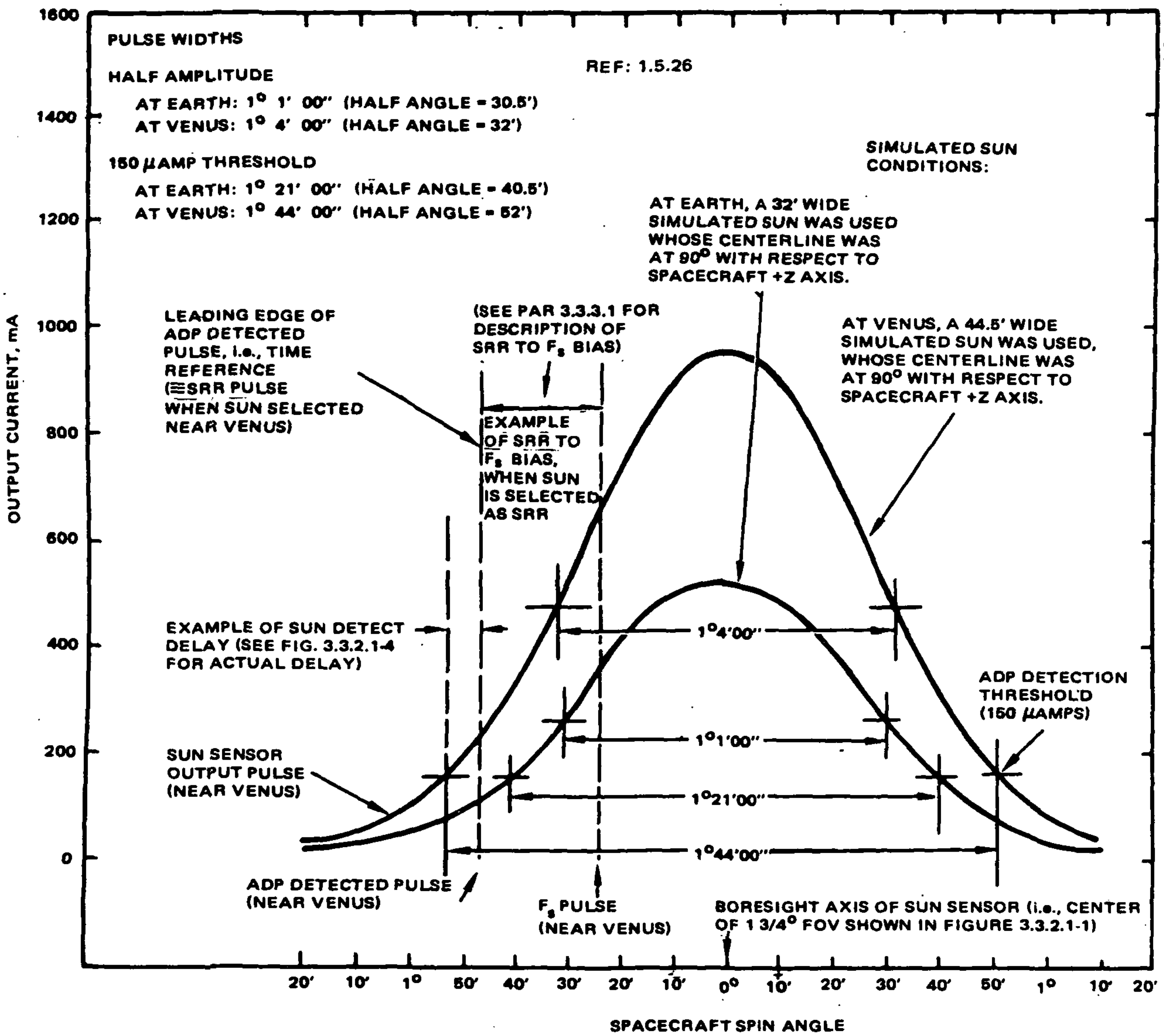


FIGURE 3.3.2.1-3. SUN SENSOR PULSE VARIATION AND TIME RELATIONSHIP TO SRR PULSE AND F_S PULSE

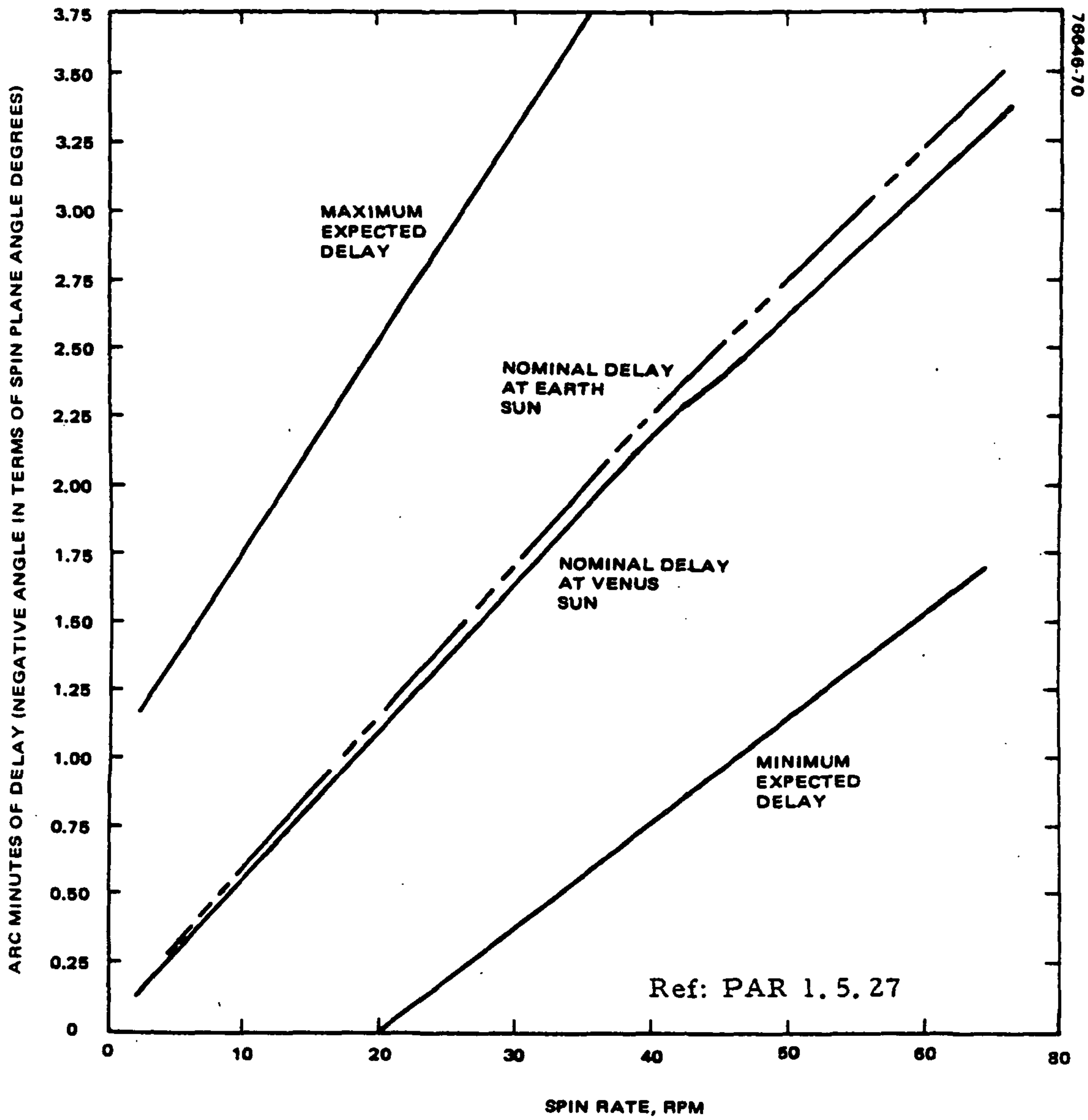


FIGURE 3.3.2.1-4. SUN DETECT DELAY IN SUN SENSOR OUTPUT PULSE vs SPACECRAFT SPIN RATE

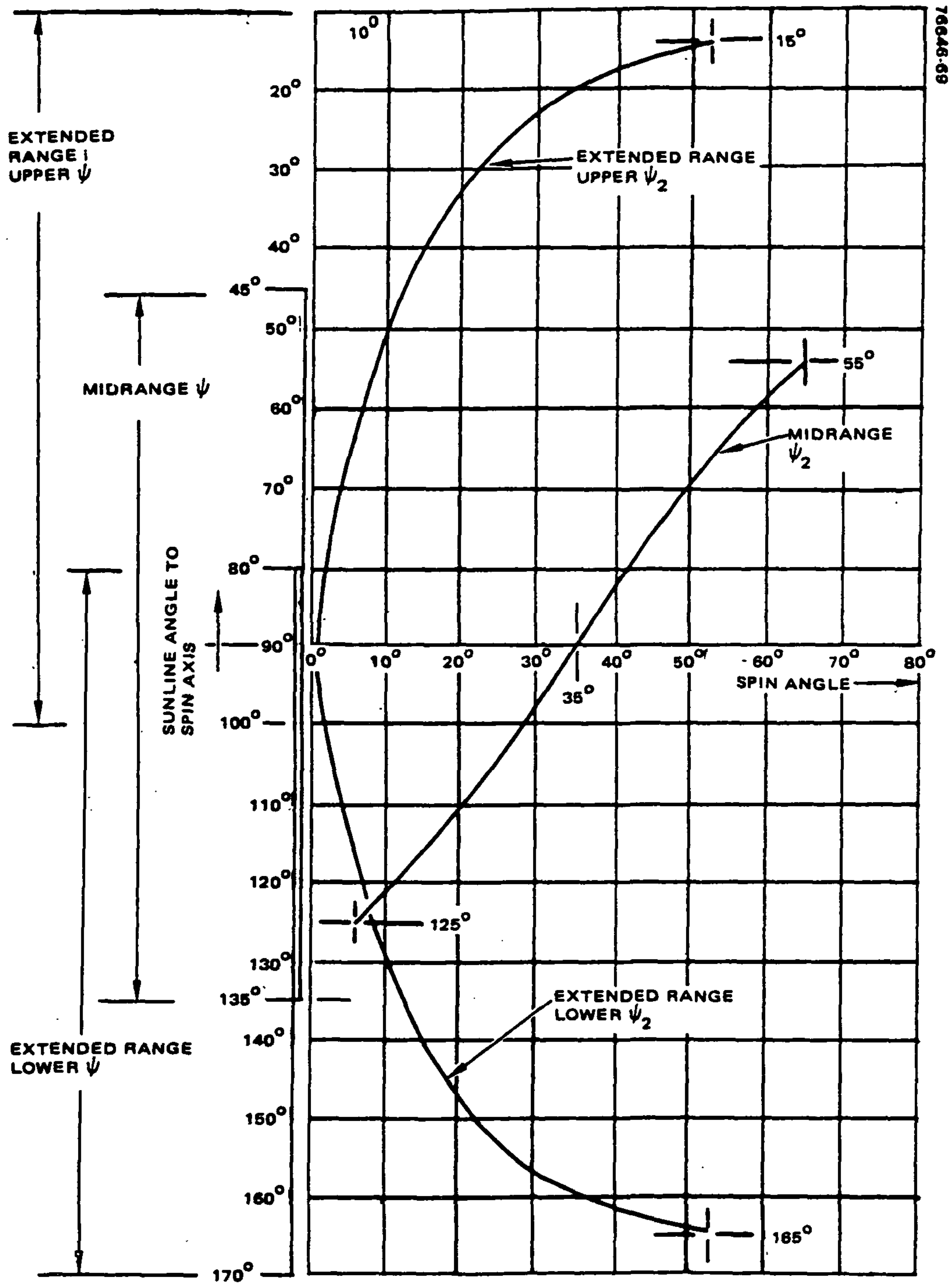


FIGURE 3.3.2.1-5. NOMINAL PSI_2 SPIN ANGLE DELAY vs SUNLINE ANGLE

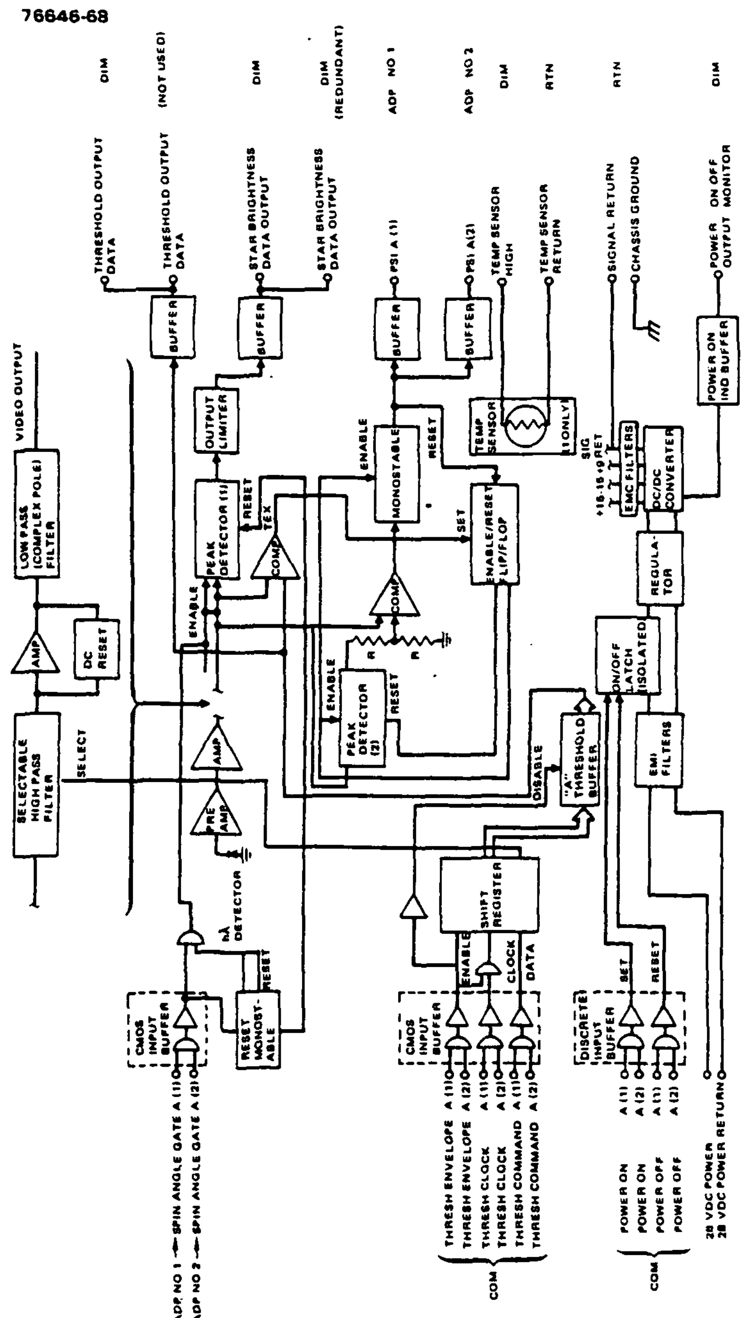
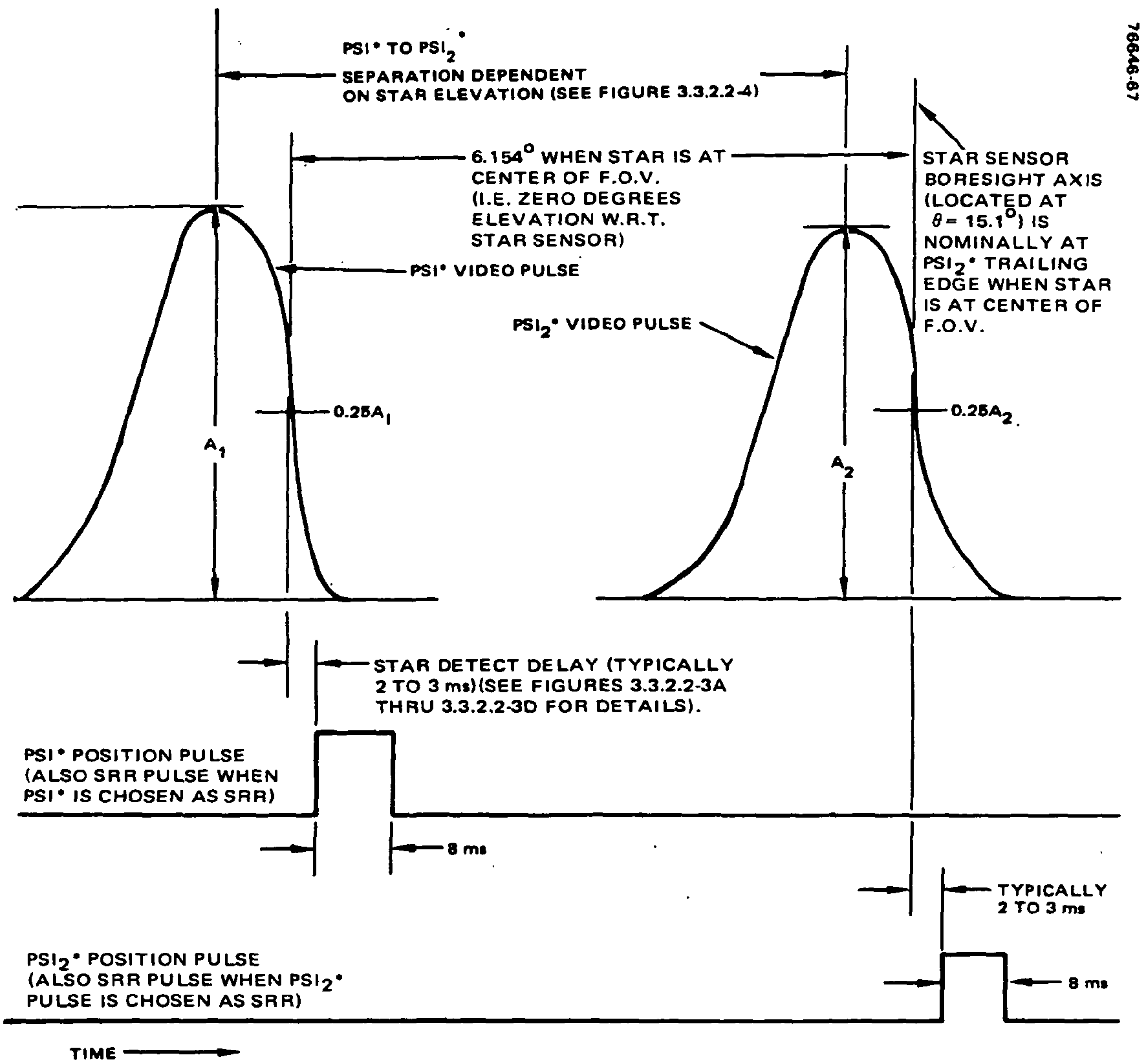


FIGURE 3.3.2.2-1. STAR SENSOR BLOCK DIAGRAM (ONE OF TWO CHANNELS)



LEADING EDGE OF PSI* PULSE OR PSI2* PULSE REPRESENTS EVENT TIME

FIGURE 3.3.2.2-2. STAR SENSOR VIDEO PULSE AND POSITION PULSE RELATIONSHIPS

FIGURES LISTED BELOW — TO BE SUPPLIED LATER —

Figure 3.3.2.2-3A. PSI* to PSI2* Position Pulse Delay Angles

Figure 3.3.2.2-3B.

Figure 3.3.2.2-3C.

Figure 3.3.2.2-3D.

FIGURES LISTED BELOW — TO BE SUPPLIED LATER —

Figure 3.3.2.2-4A. PSI* to PSI2* Delay Angle

Figure 3.3.2.2-4B.

Revision

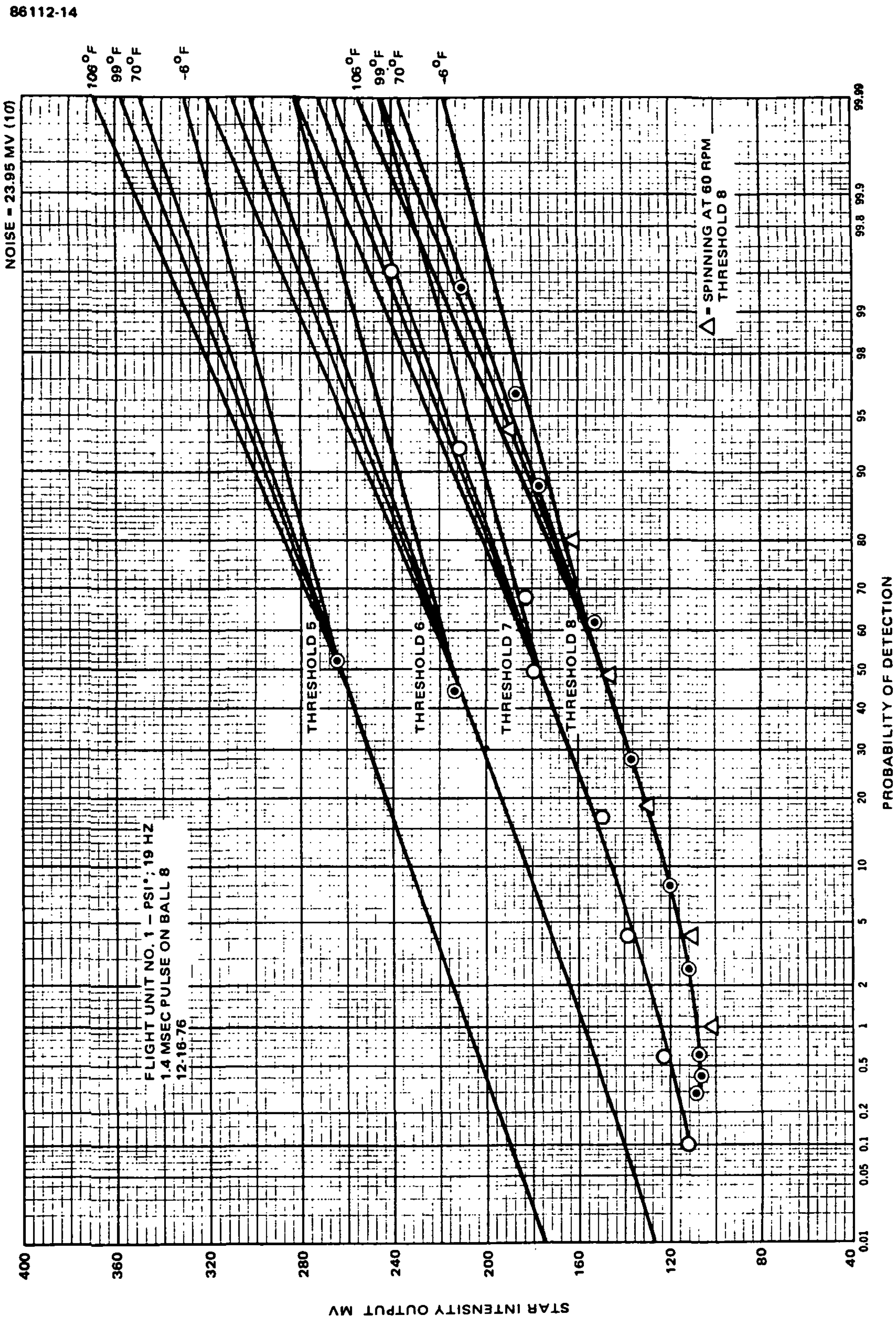


FIGURE 3.3.2.2-5A. EFFECT OF TEMPERATURE ON P_D OF A STAR; PSI CHANNEL AND 19 Hz BANDPASS IN USE

Revision

86112-15

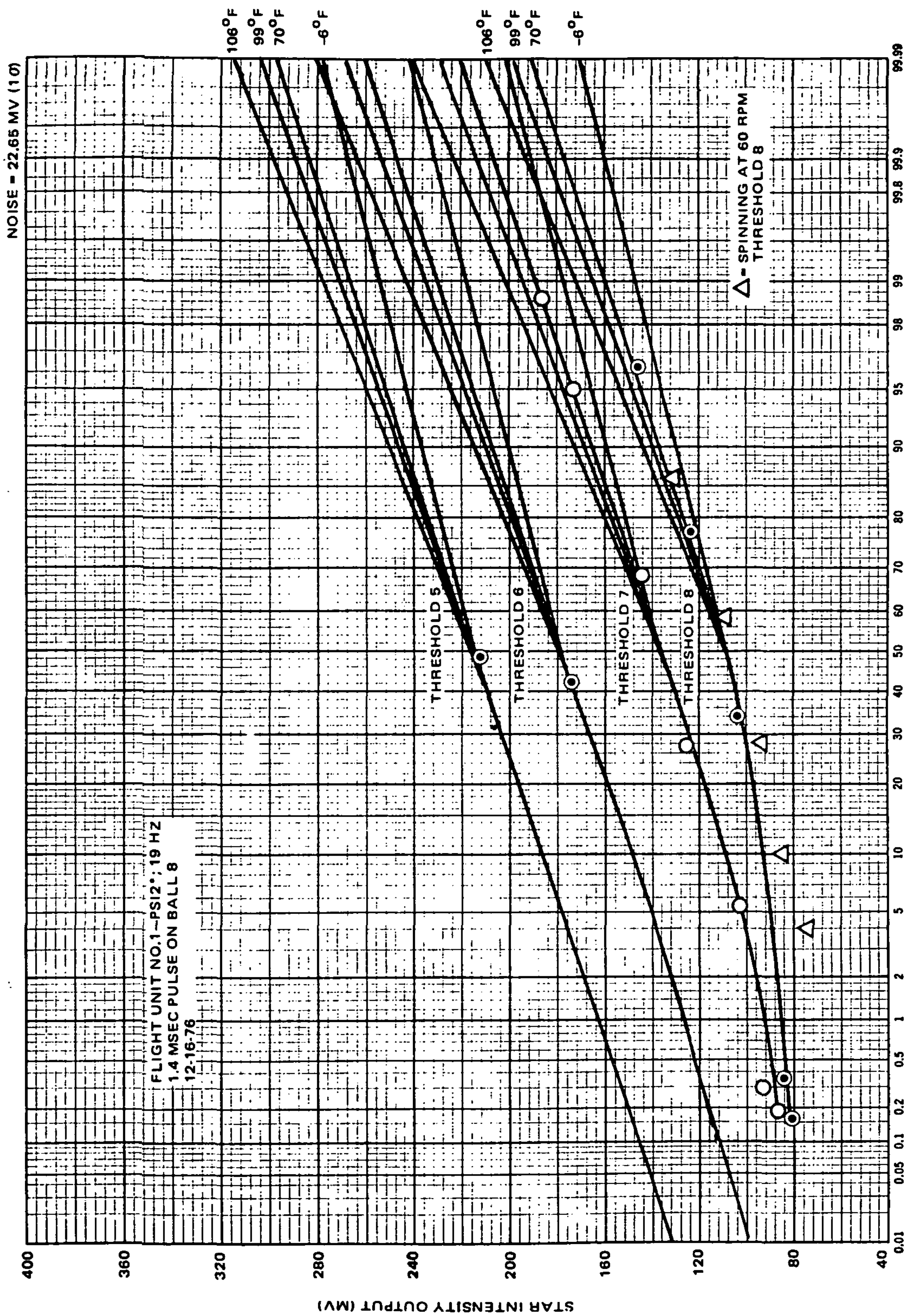


FIGURE 3.3.2.2-5B. EFFECT OF TEMPERATURE ON P_D OF A STAR; PSI2* CHANNEL AND 19 Hz BANDPASS IN USE

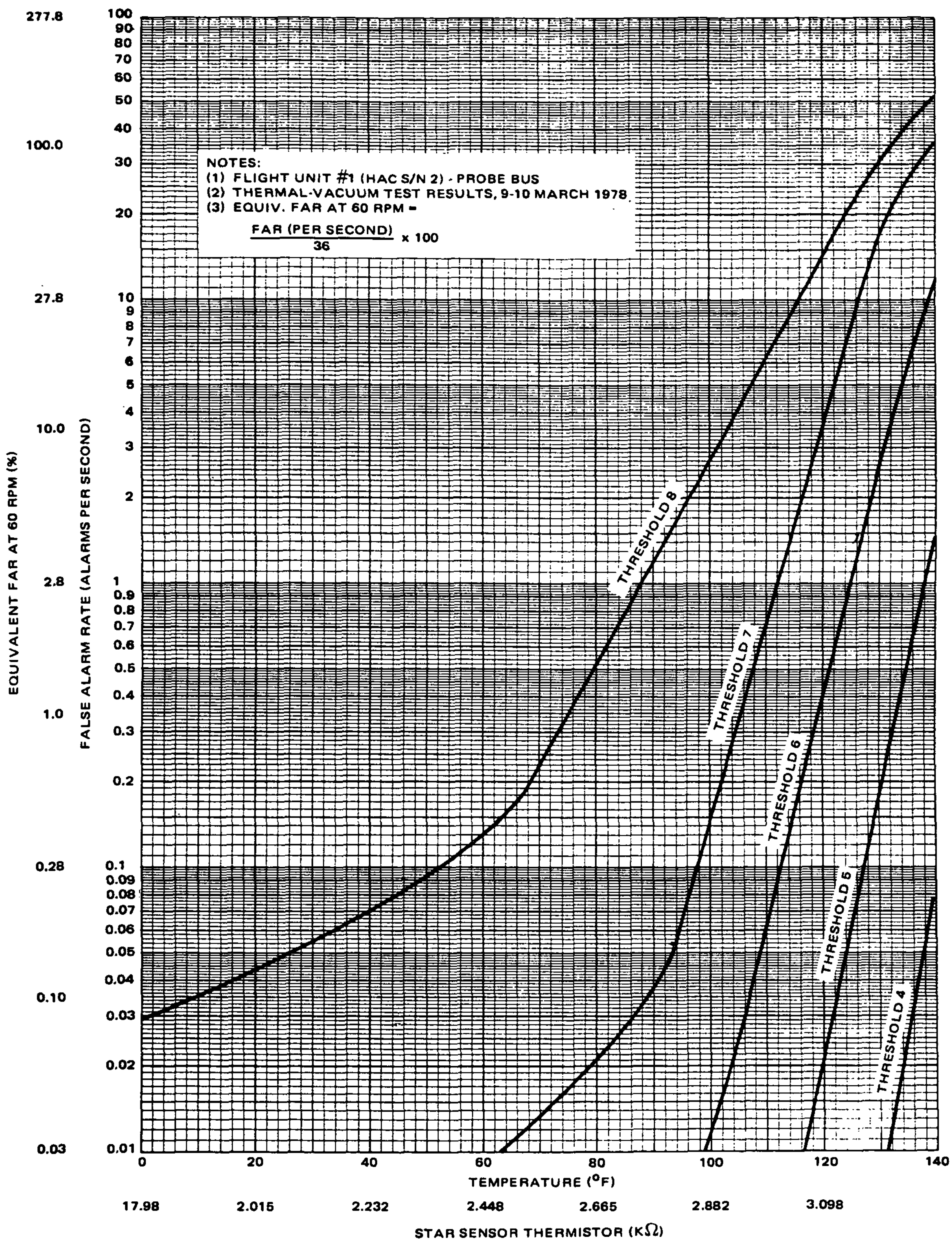


FIGURE 3.3.2.2-5C. FALSE ALARM RATE (STAR DETECTION) VERSUS TEMPERATURE & SELECTED THRESHOLD: PSI*, 2Hz, 60 RPM.

Revision

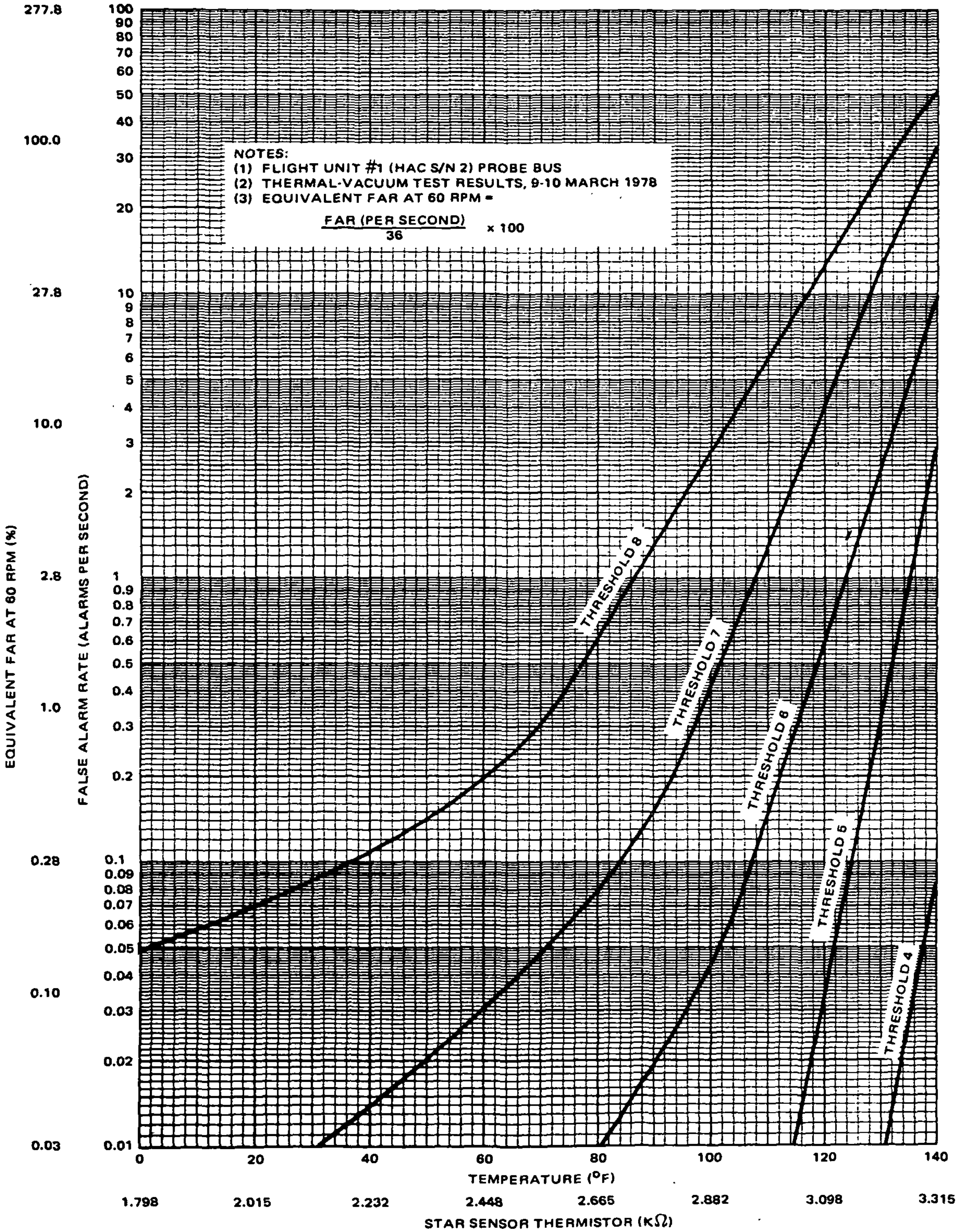


FIGURE 3.3.2.2-5D. FALSE ALARM RATE (STAR DETECTION) VERSUS TEMPERATURE & SELECTED THRESHOLD: PSI*, 19Hz, 60 RPM.

Revision

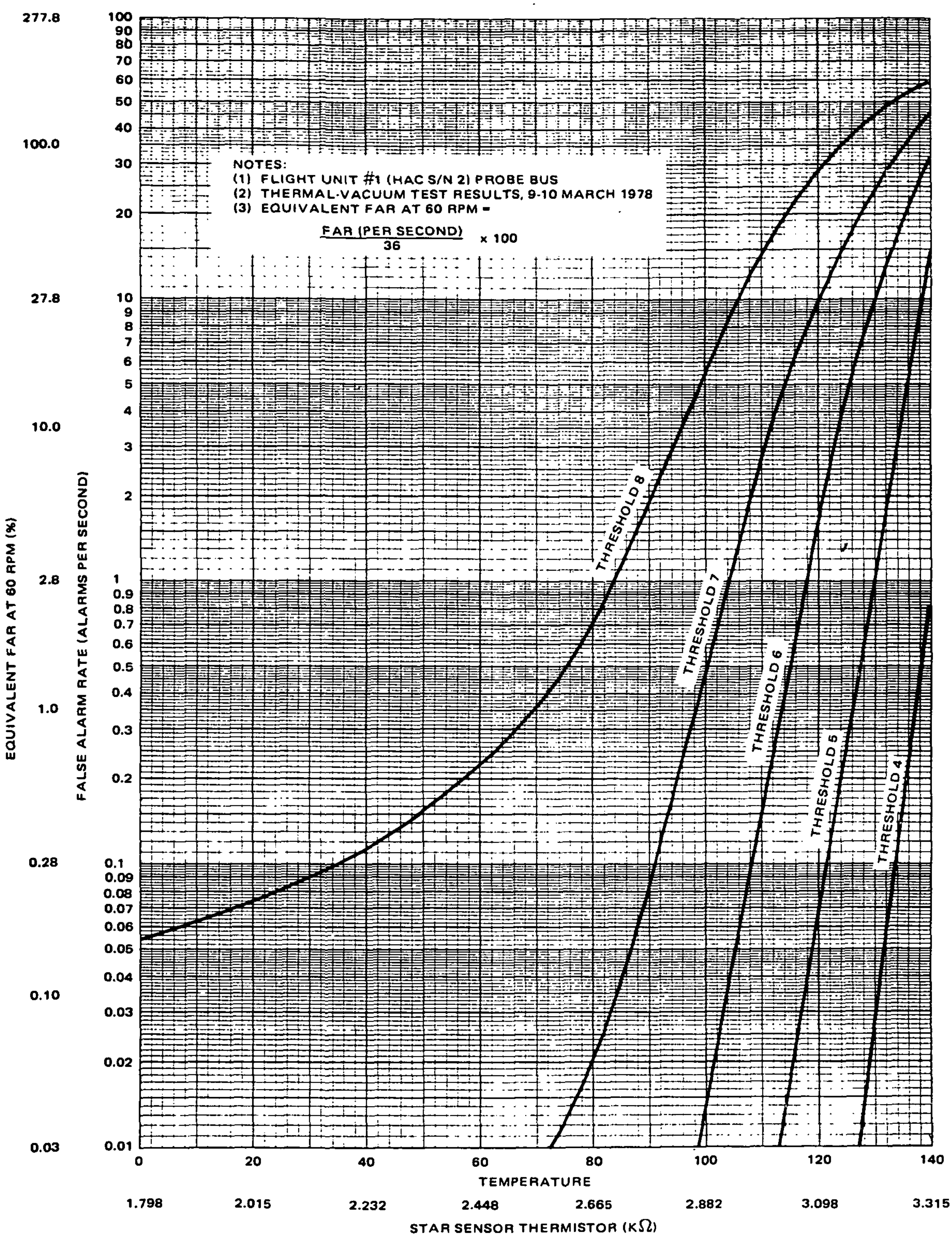


FIGURE 3.3.2.2-5E. FALSE ALARM RATE (STAR DETECTION) VERSUS TEMPERATURE & SELECTED THRESHOLD: PSI2*, 2Hz, 60 RPM.

Revision

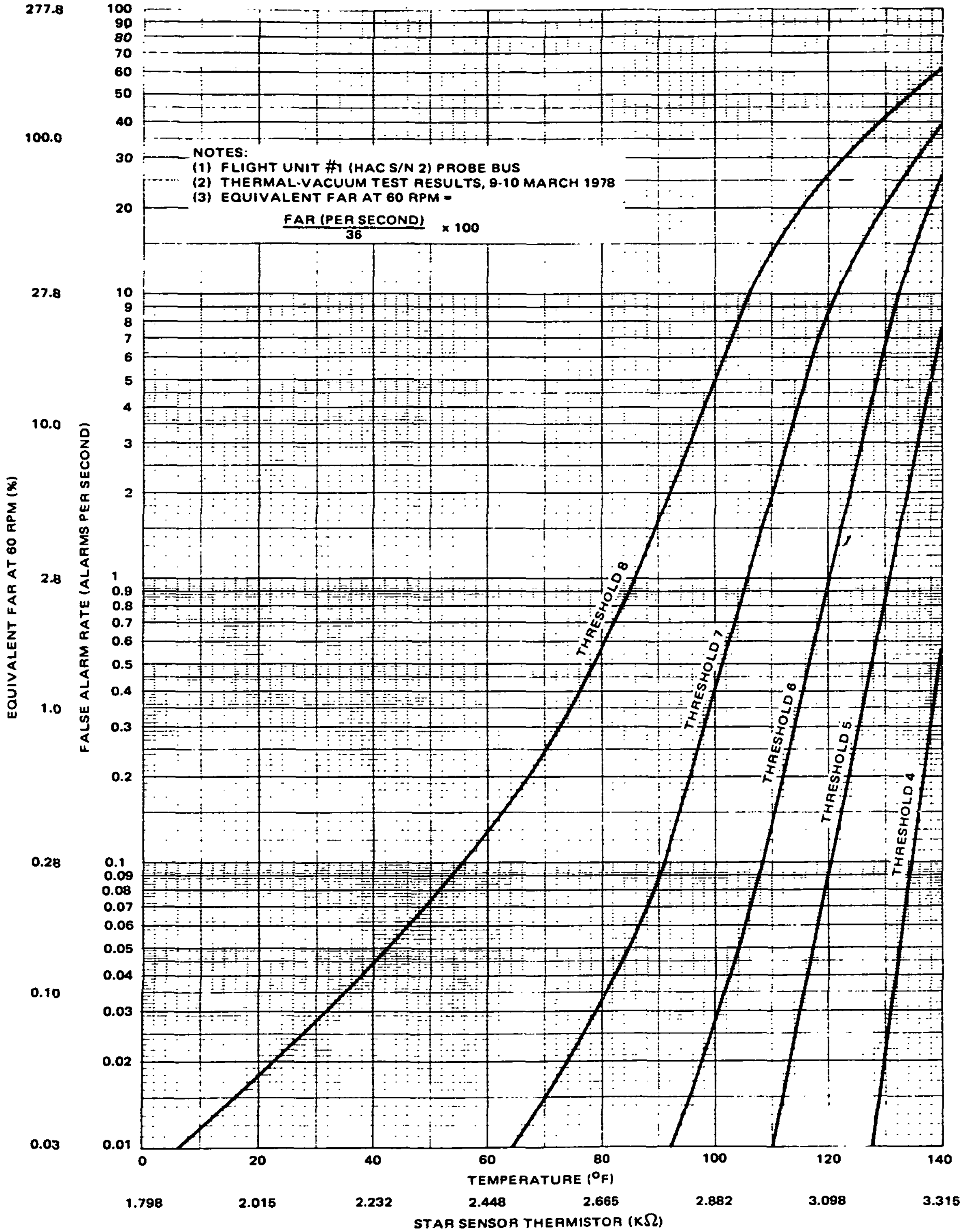


FIGURE 3.3.2.2-5F. FALSE ALARM RATE (STAR DETECTION) VERSUS TEMPERATURE & SELECTED THRESHOLD: PSI2*, 19Hz, 60 RPM.

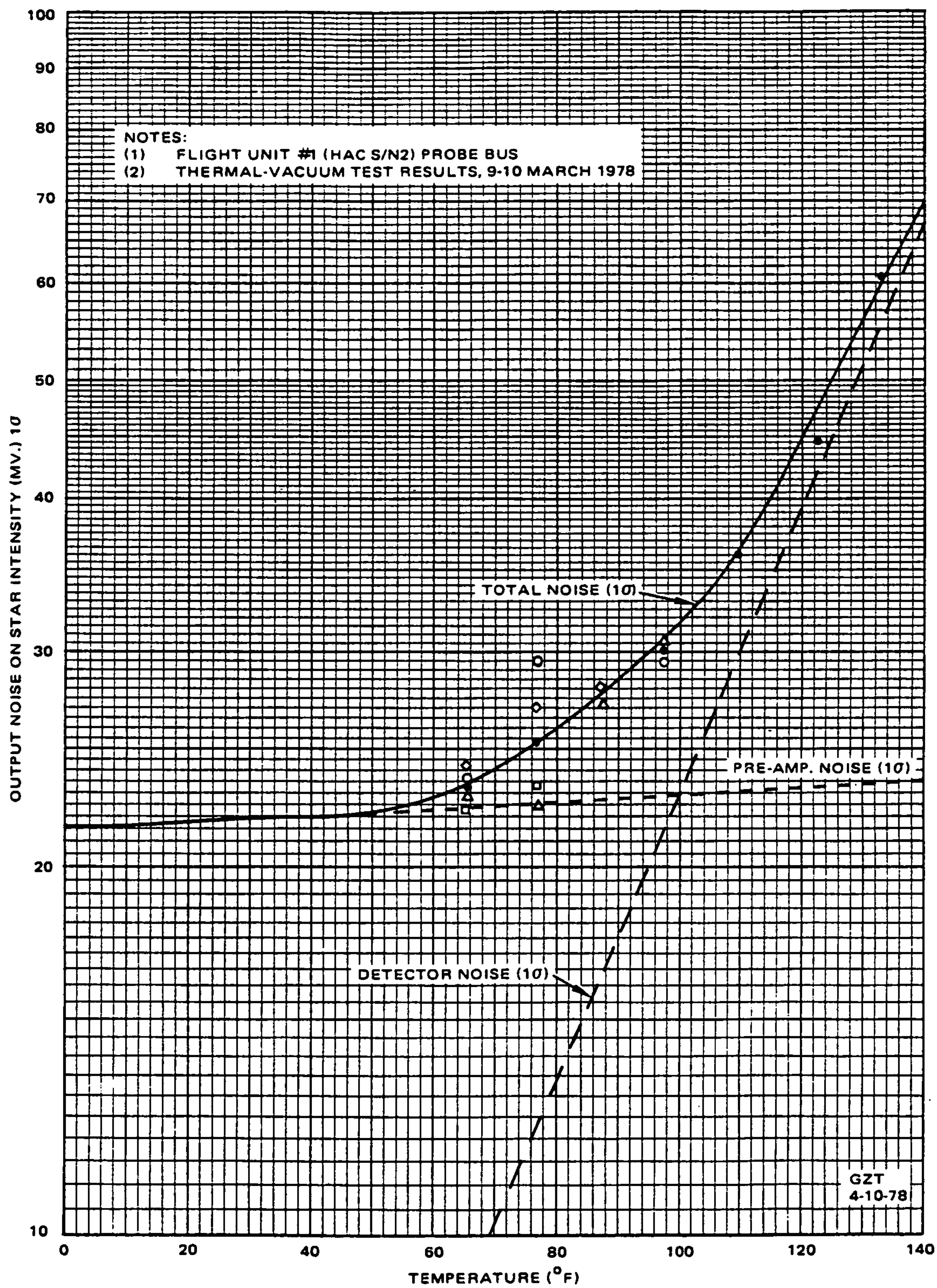


FIGURE 3.3.2.2-5G. STAR INTENSITY NOISE (1σ) AT STAR SENSOR OUTPUT VERSUS TEMPERATURE

Revision

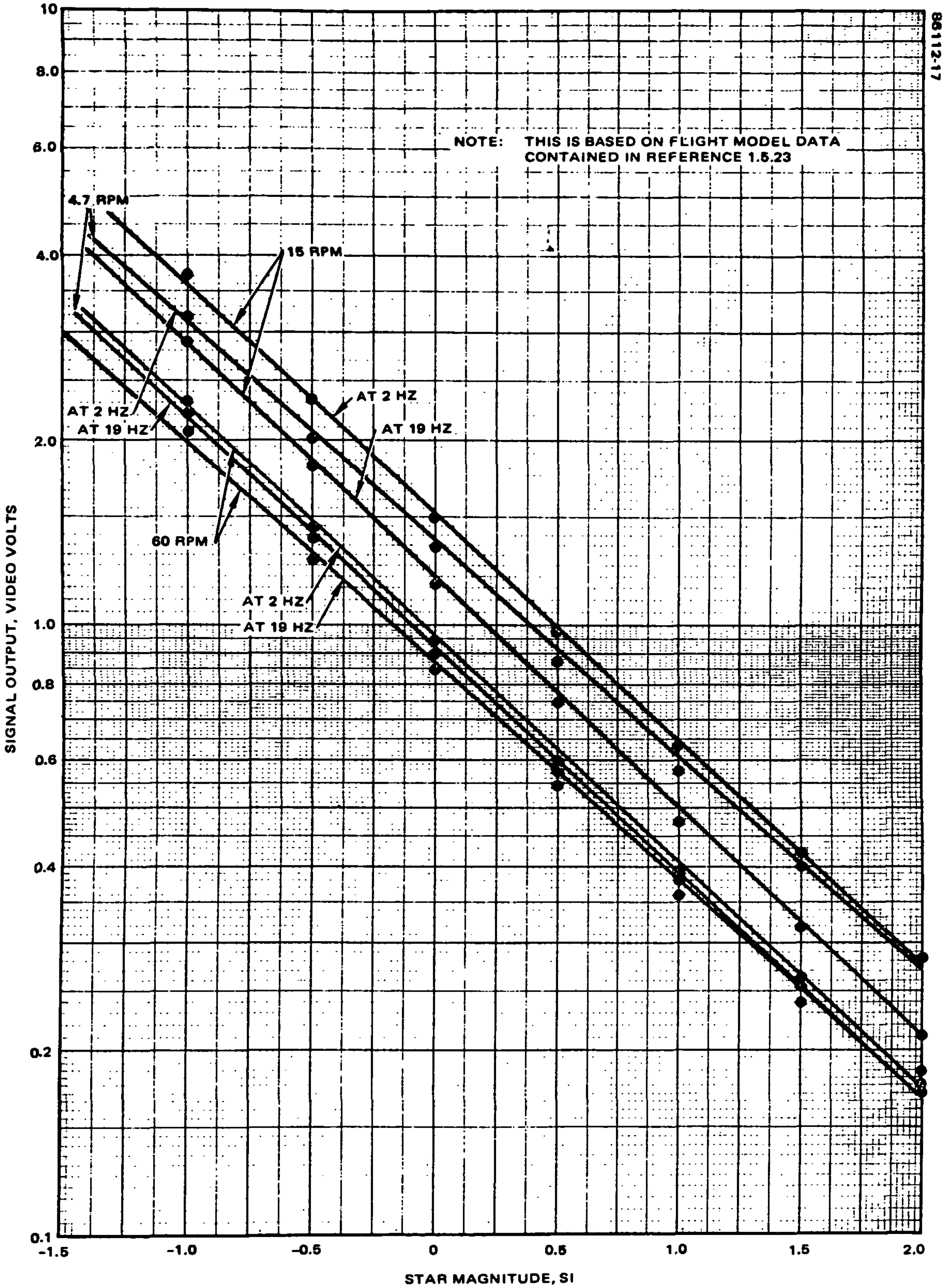


FIGURE 3.3.2.2-6. VARIATION OF STAR BRIGHTNESS WITH SPIN RATE

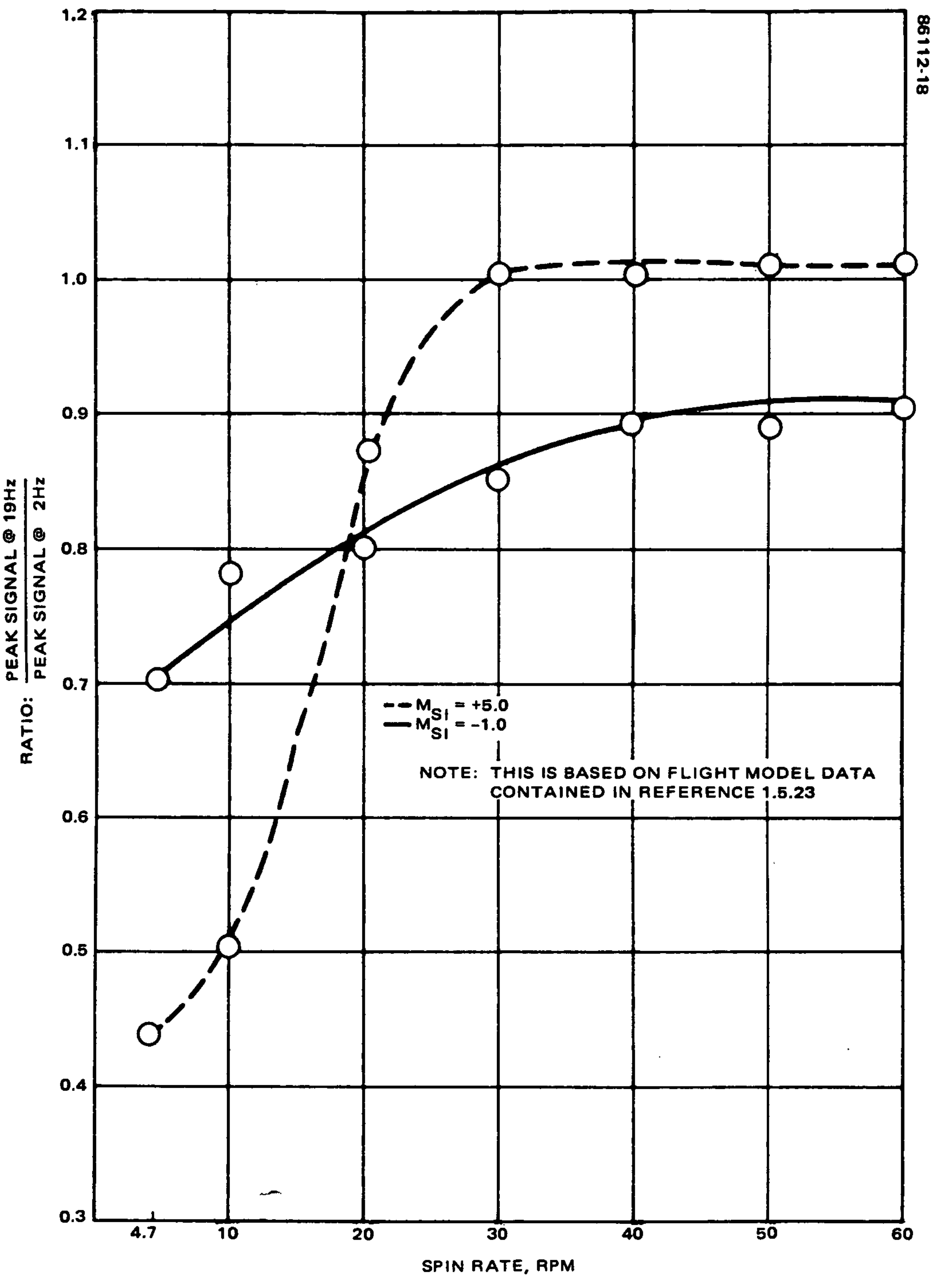


FIGURE 3.3.2.2-7. BANDPASS SELECTION EFFECT ON STAR BRIGHTNESS

Revision

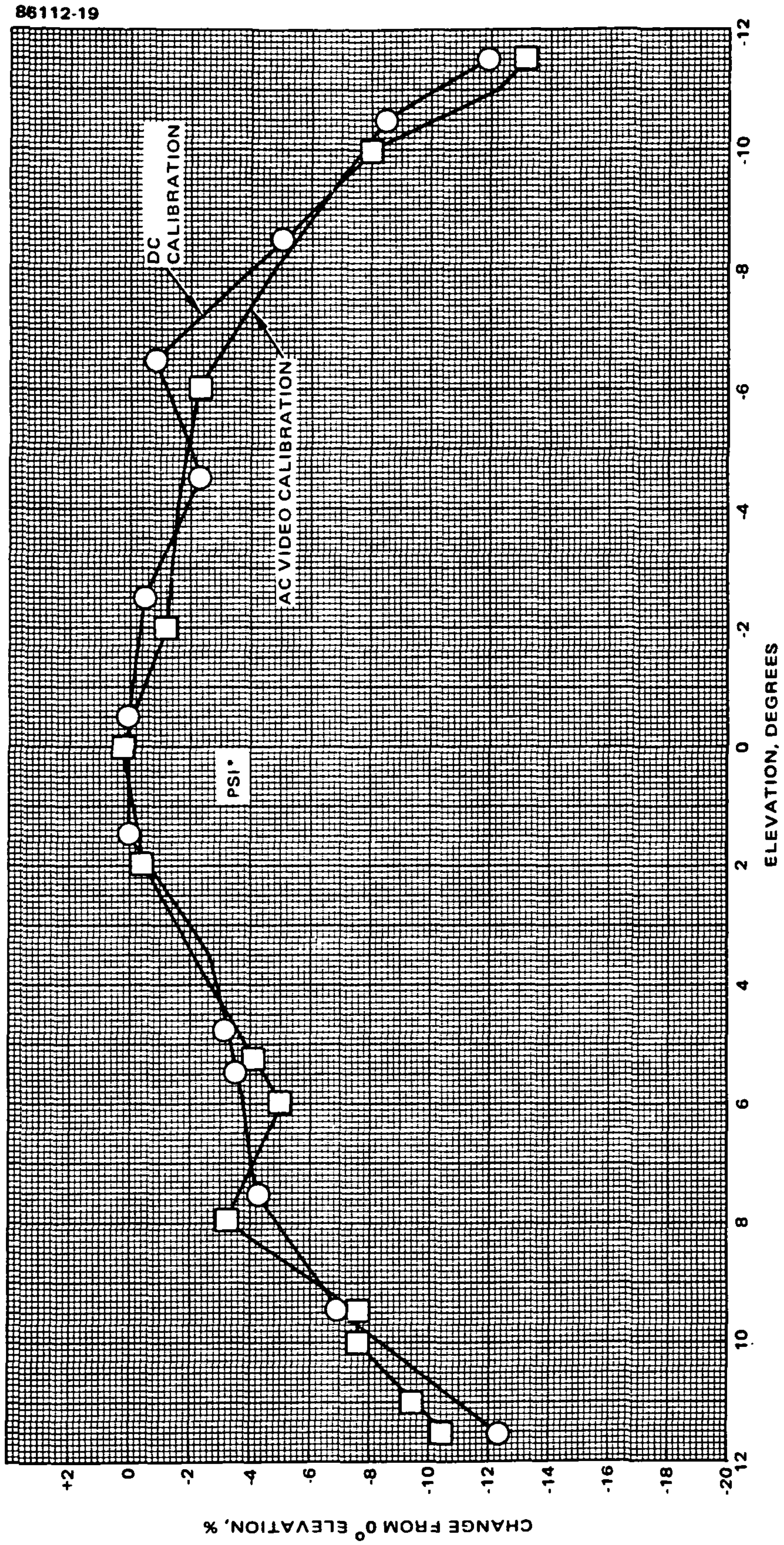


FIGURE 3.3.2.2-8A. VARIATION OF STAR BRIGHTNESS (PSI* CHANNEL) VERSUS ELEVATION FOV WRT STAR SENSOR CENTER FOV LOCATED 56° FROM SPACECRAFT (+Z AXIS)

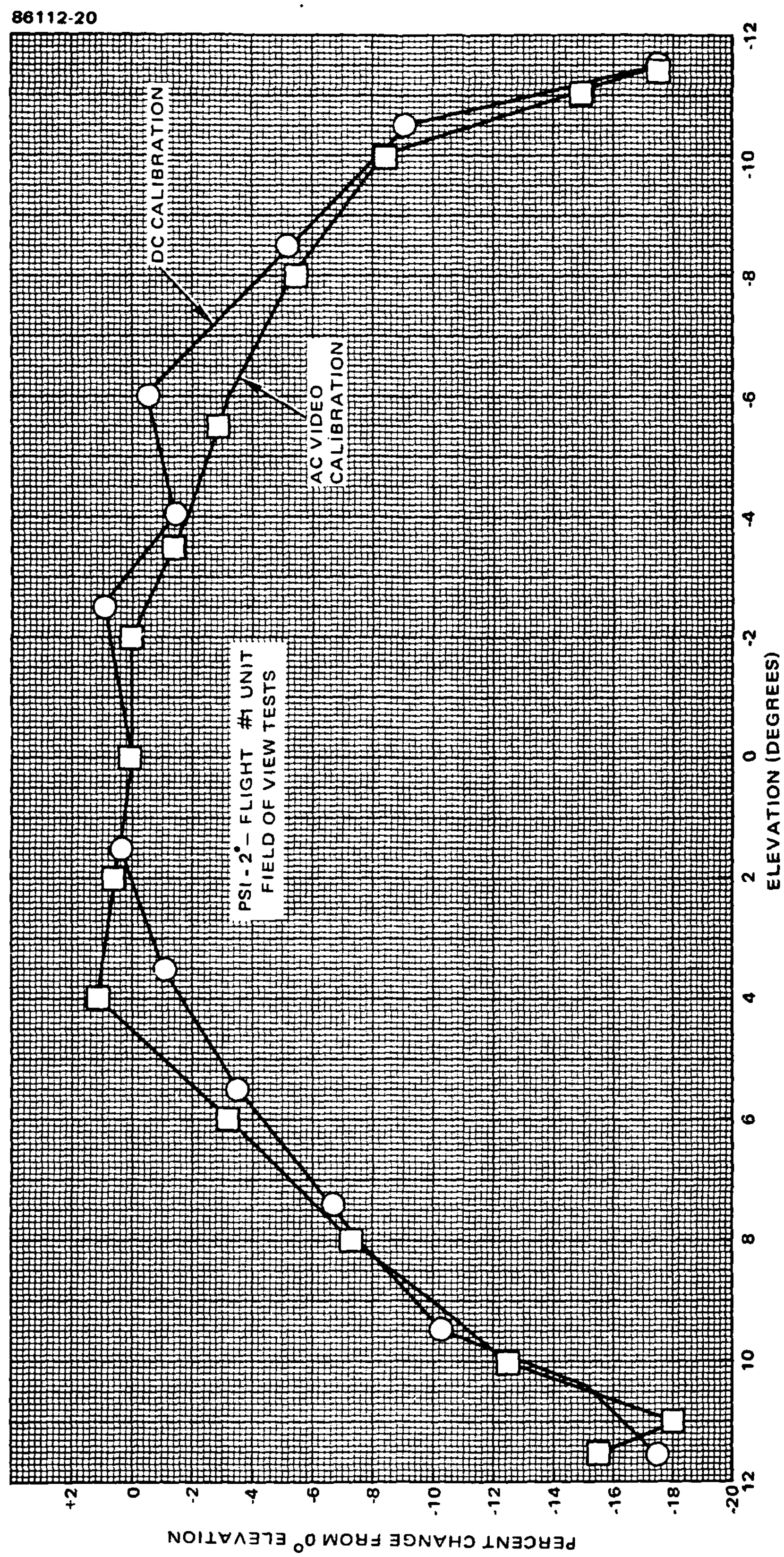


FIGURE 3.3.2.2-8B. VARIATION OF STAR BRIGHTNESS (PSI2* CHANNEL) VERSUS ELEVATION FOV WRT STAR SENSOR FOV (LOCATED 56° FROM SPACECRAFT +Z AXIS)

Revision

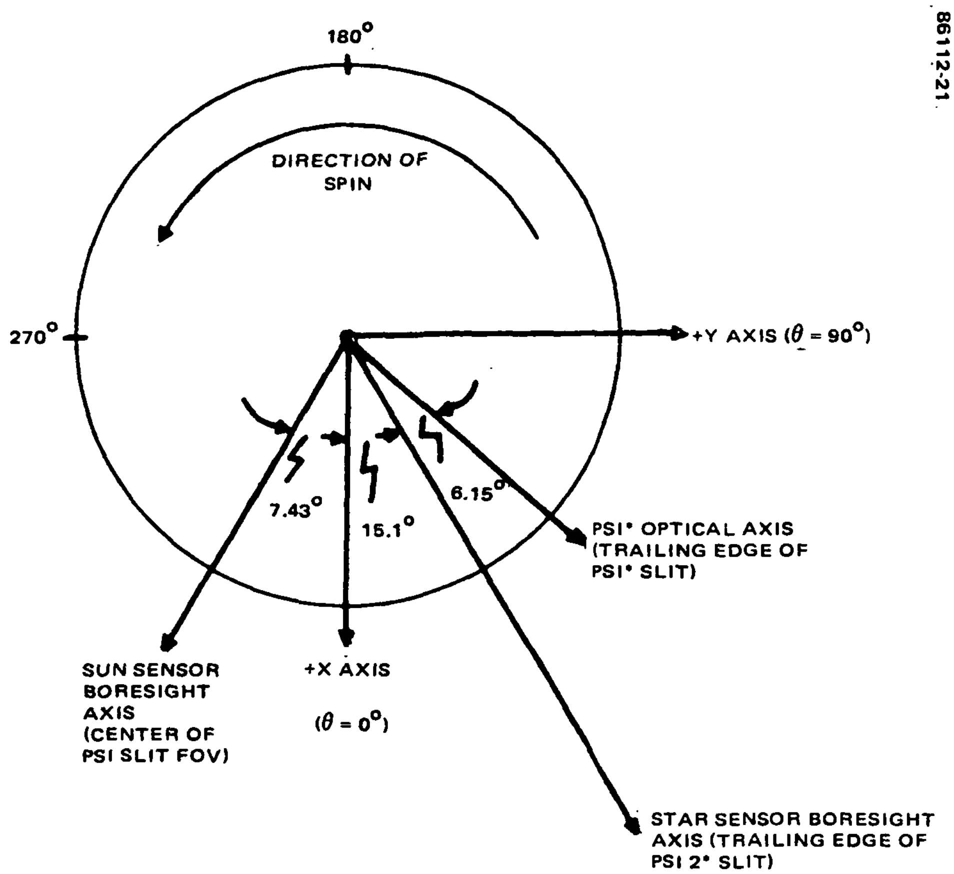


FIGURE 3.3.2.2.6-1. SUN/STAR SENSOR GEOMETRY

****
****
****   ****
****   This Figure is a Foldout.   ****
****   ****
****   See APPENDIX C   ****
****   ****
****
****

Figure 3.3.2.3-1

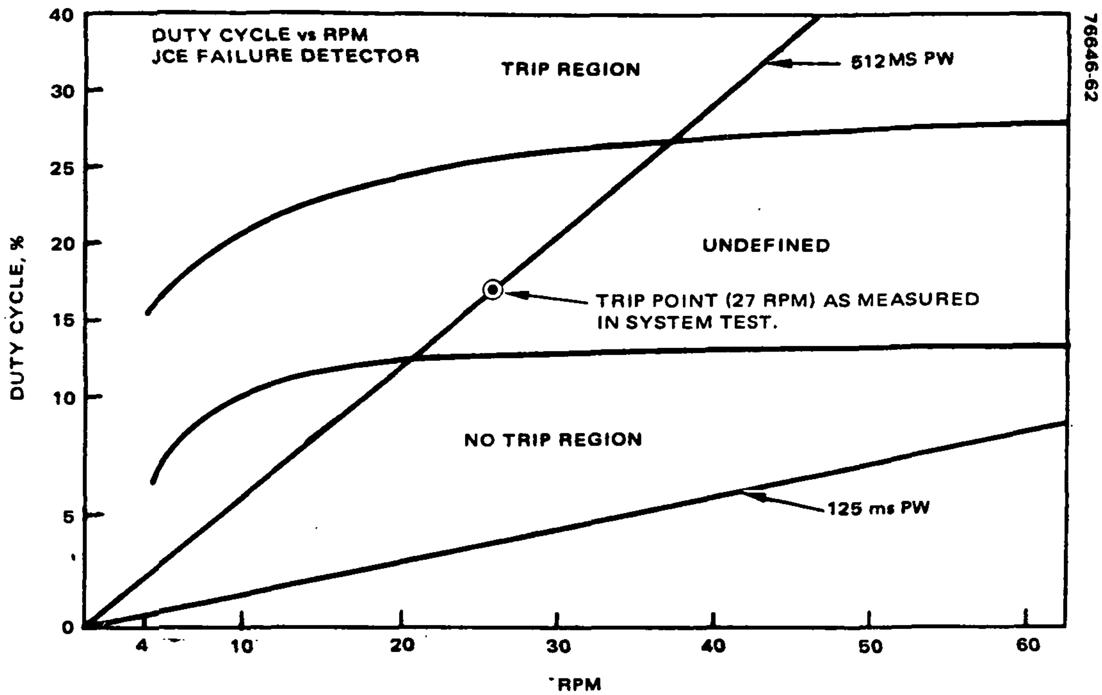


FIGURE 3.3.2.3-2. JCE DUTY CYCLE FAILURE DETECTION

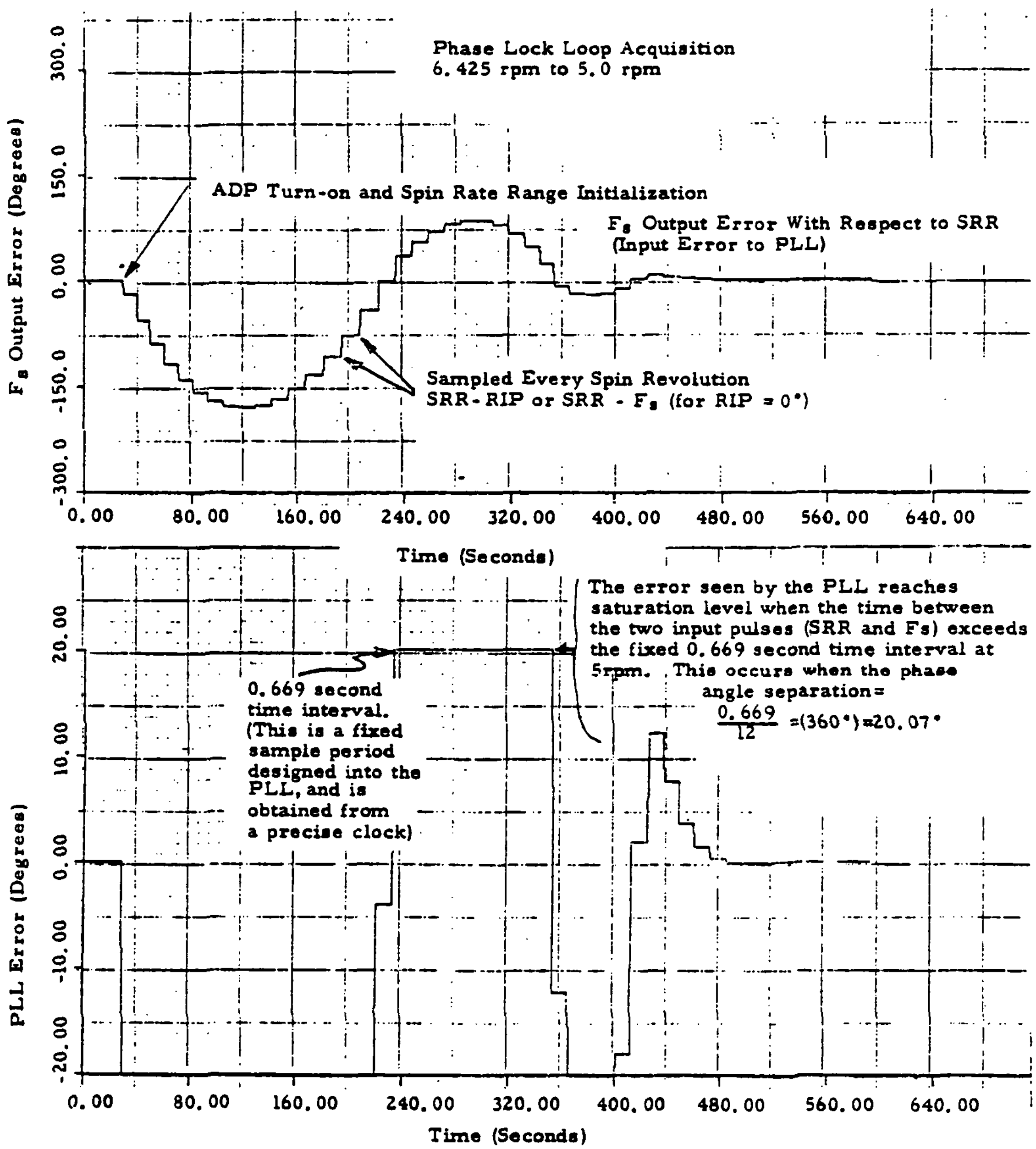


Figure 3.3.3.1-1. Phase Lock Loop Acquisition (5 rpm)

ORIGINAL PAGE IS
OF POOR QUALITY

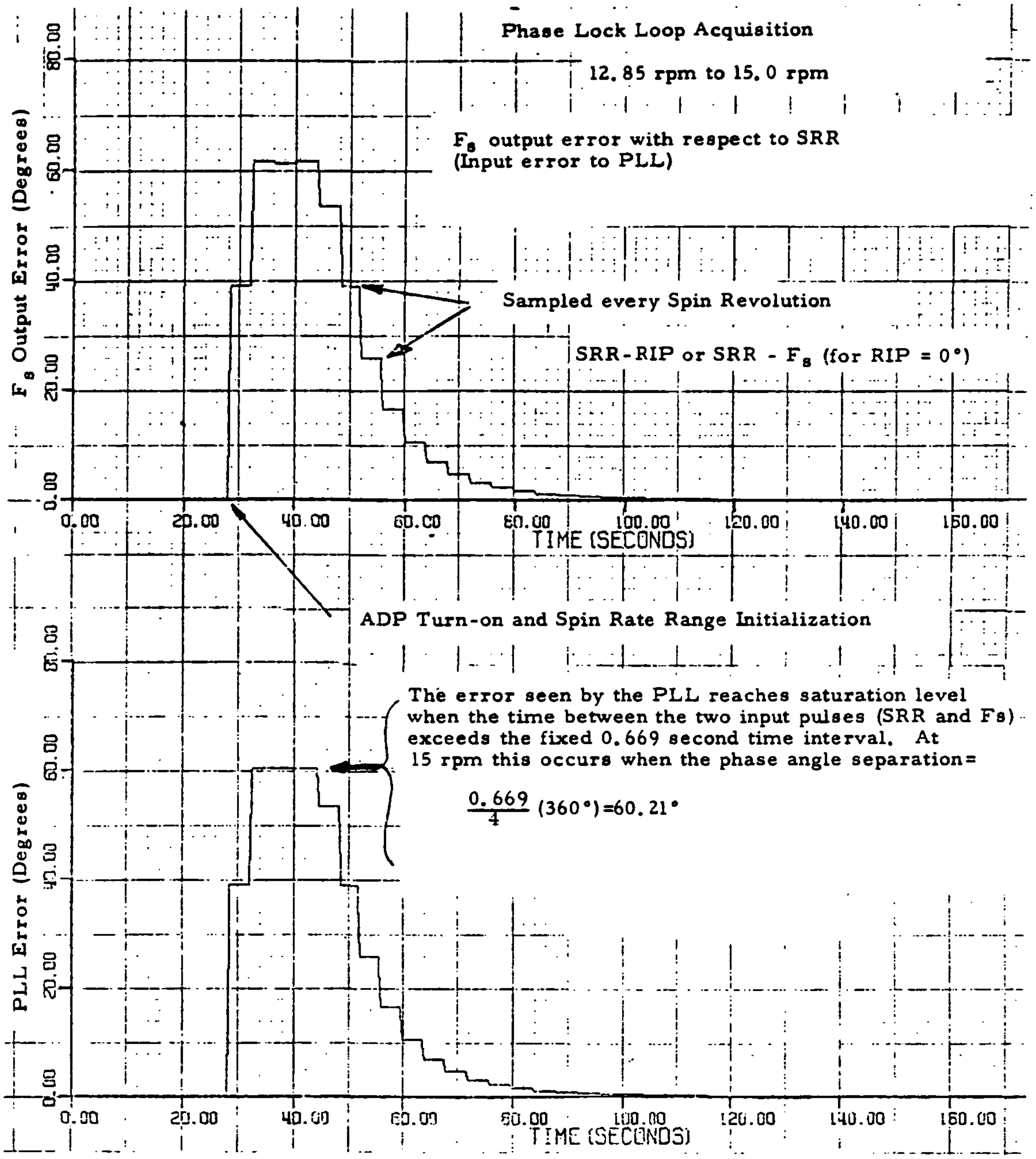


Figure 3.3.3.1-2. Phase Lock Loop Acquisition (15 rpm)

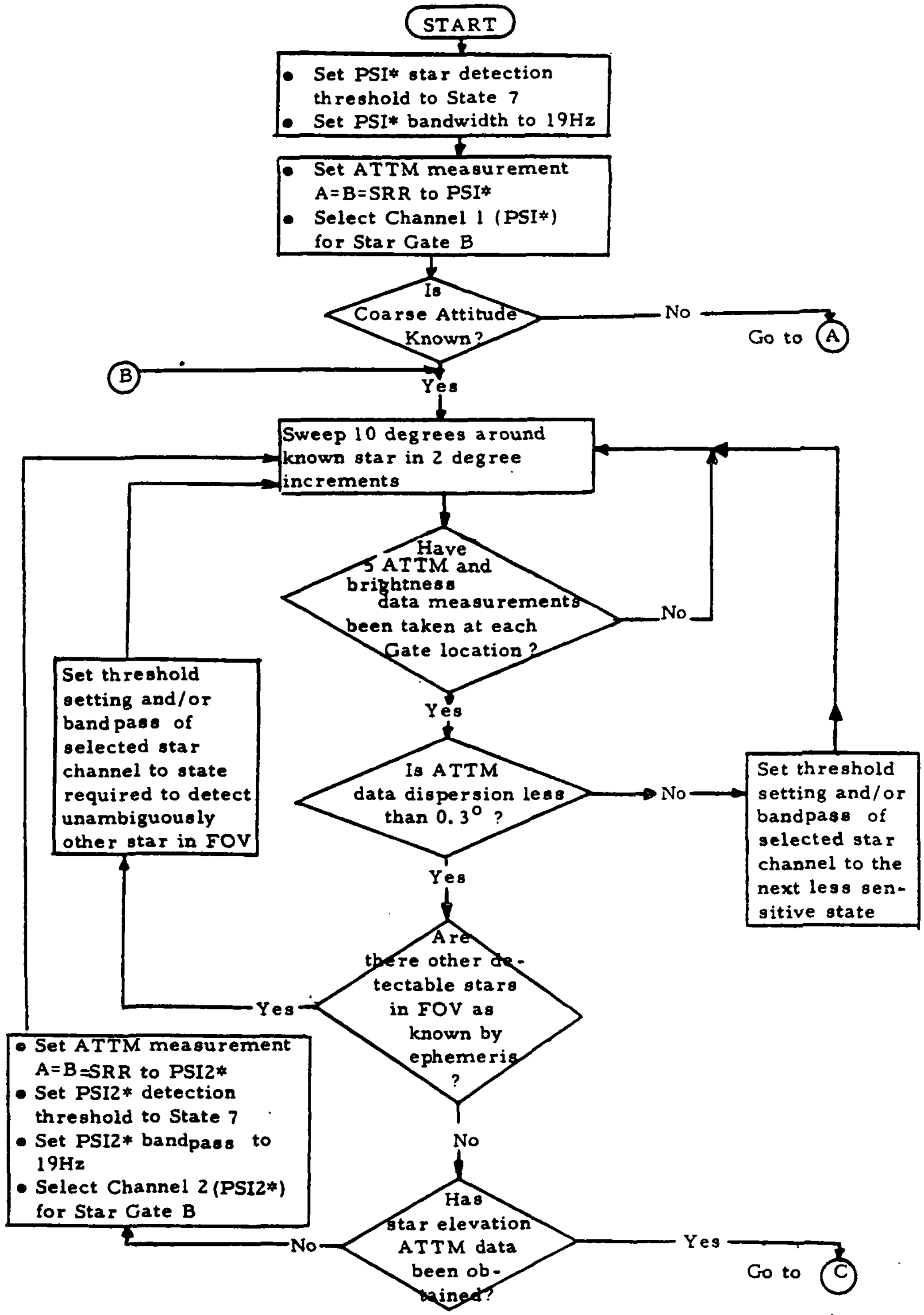


Figure 3.3.3.3-1. Star Mapping Flow Chart

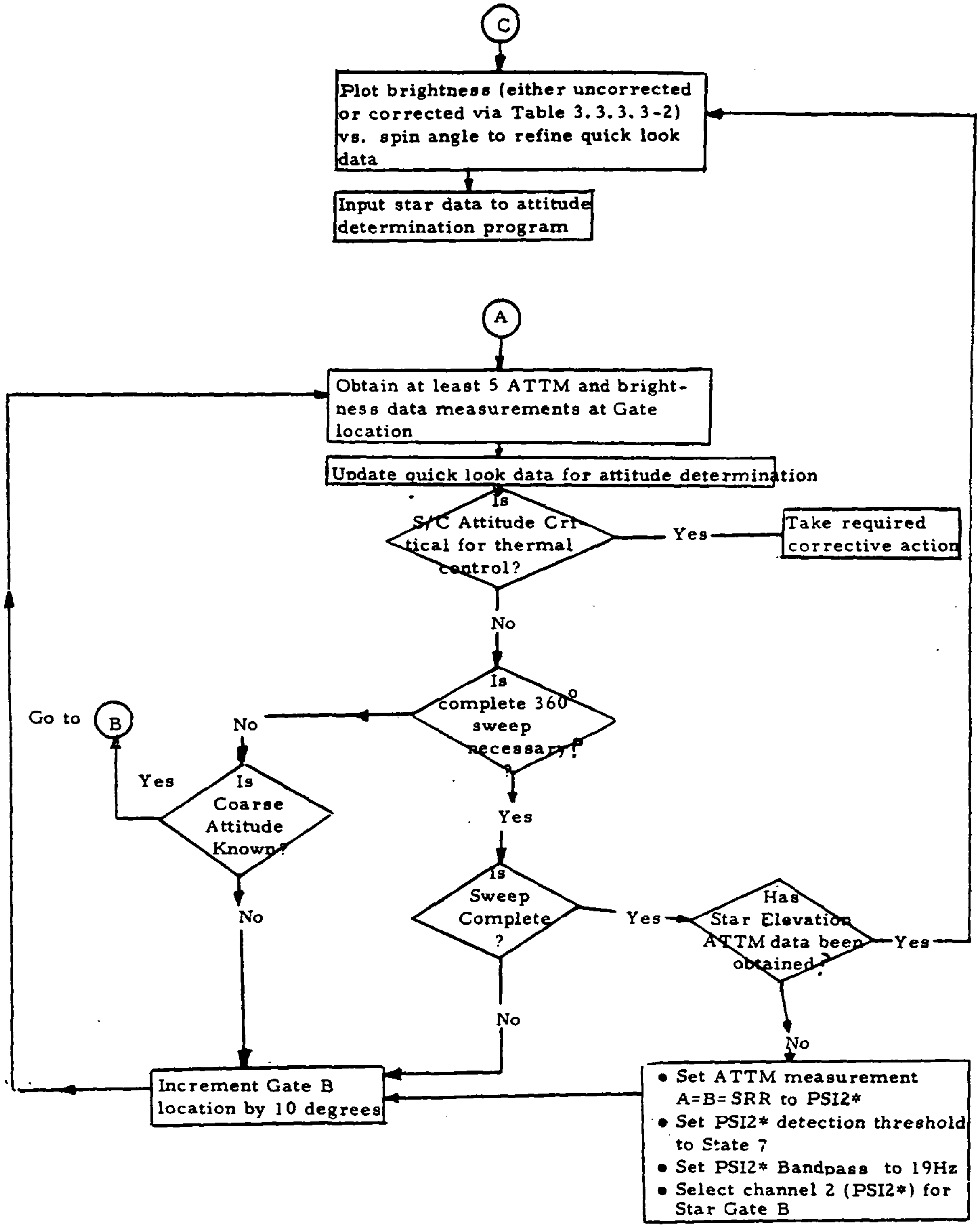


Figure 3.3.3.3-1. Star Mapping Flow Chart (Continued)

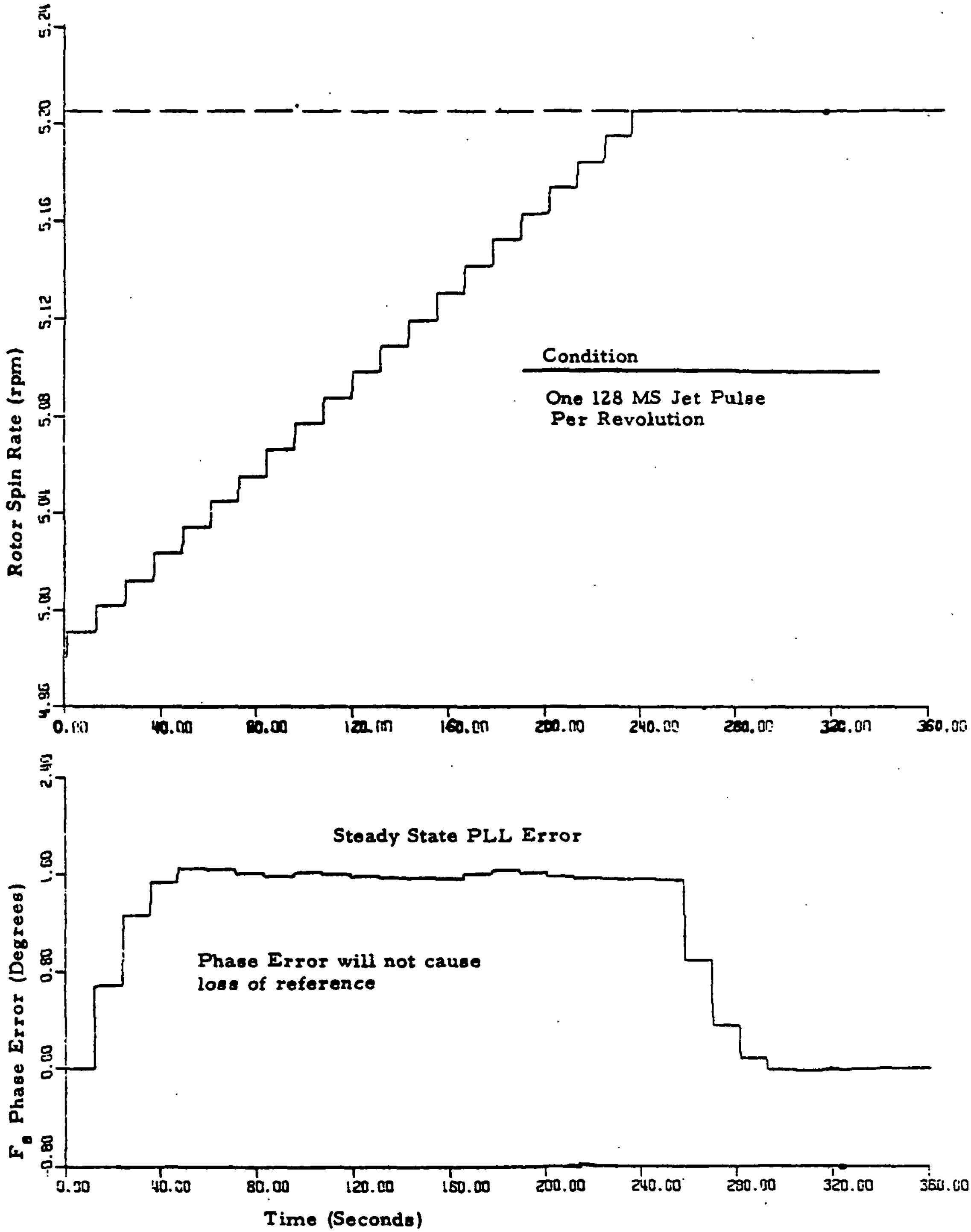


Figure 3.3.3.4-1. F_S Phase Error During Single 128 MS Pulse Per Revolution

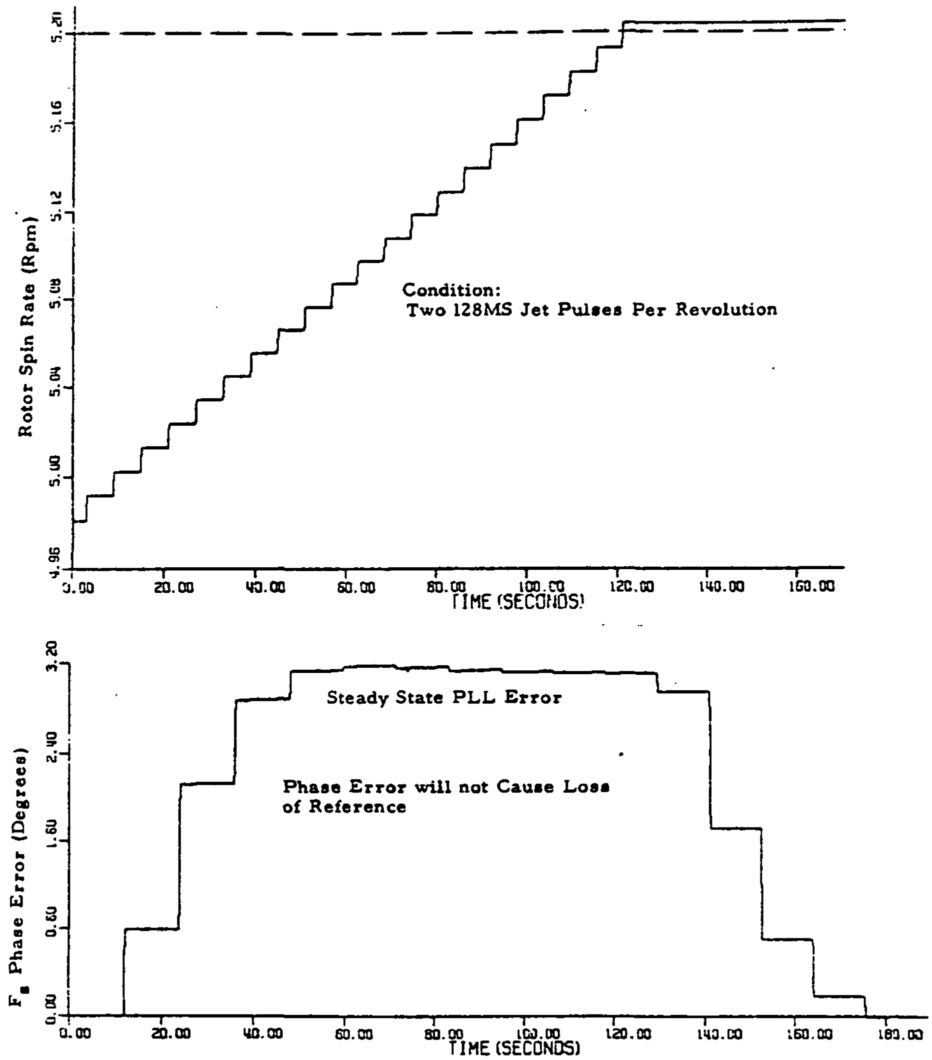


Figure 3.3.3.4-2. F_s Phase Error During Two 128 MS Pulses Per Revolution

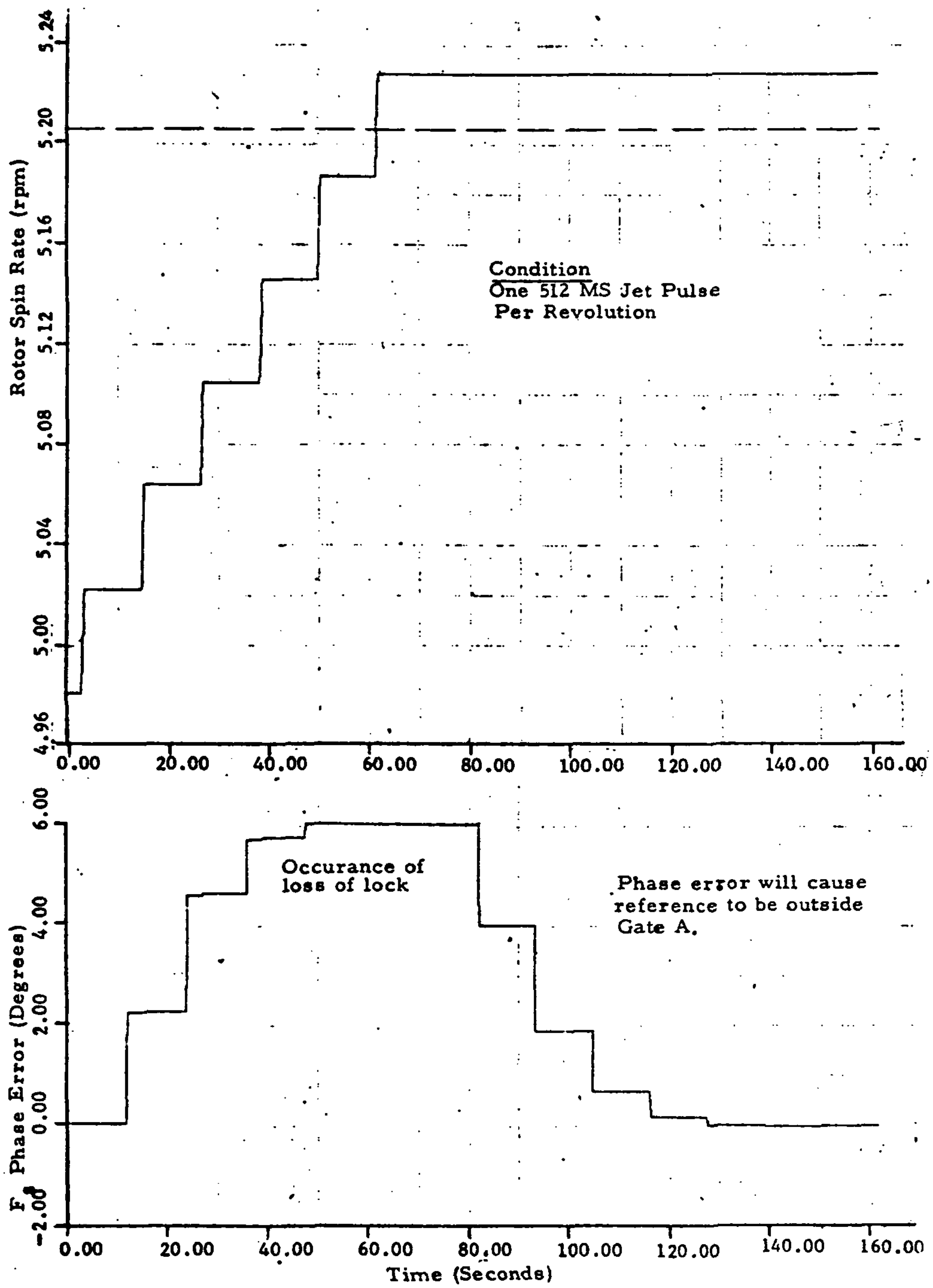


Figure 3.3.3.4-3. F_s Phase Error During Single 512 MS Pulse Per Revolution

Revision

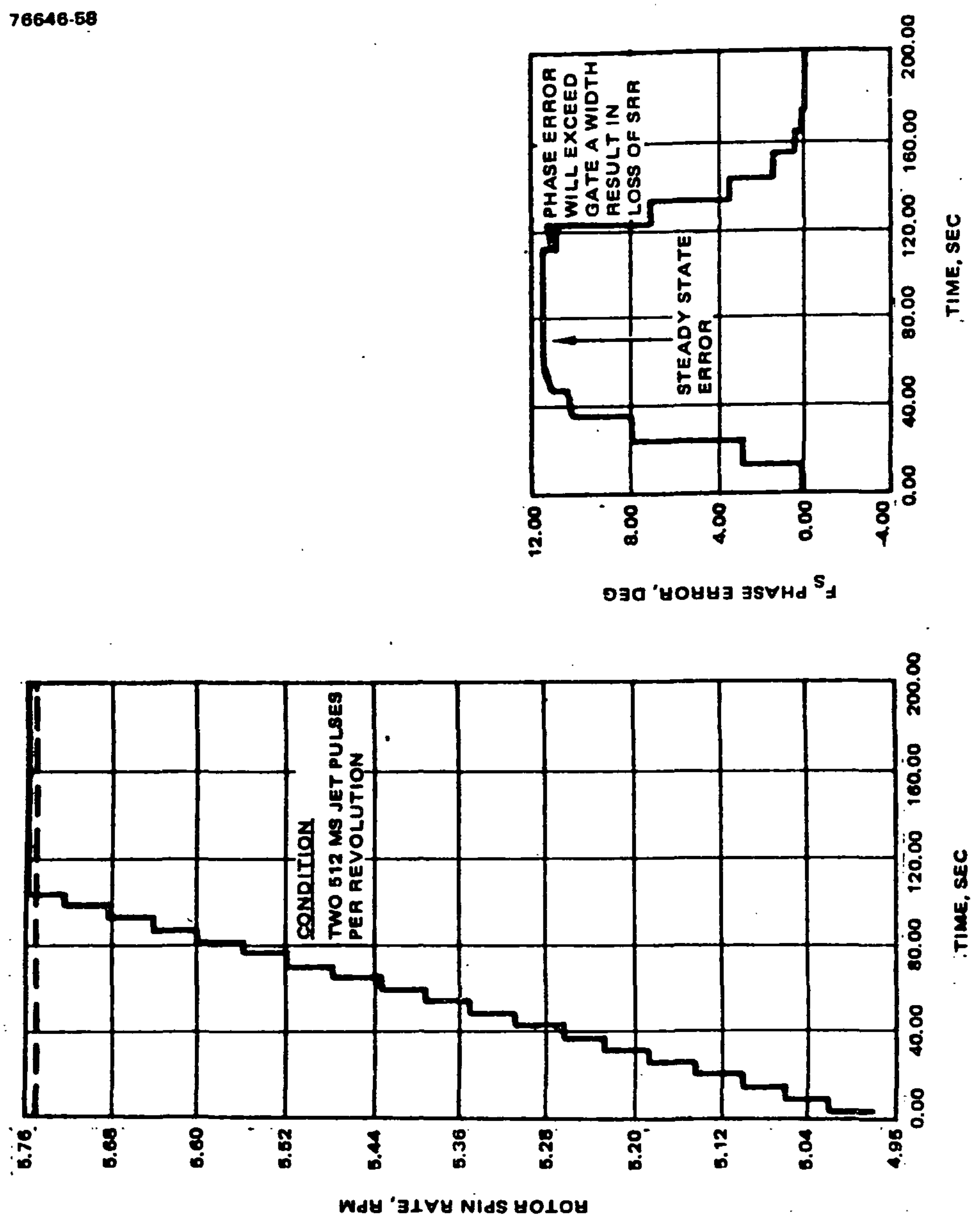


Figure 3.3.3.4-4. F_S Phase Error During Two 512 MS Pulses per Revolution

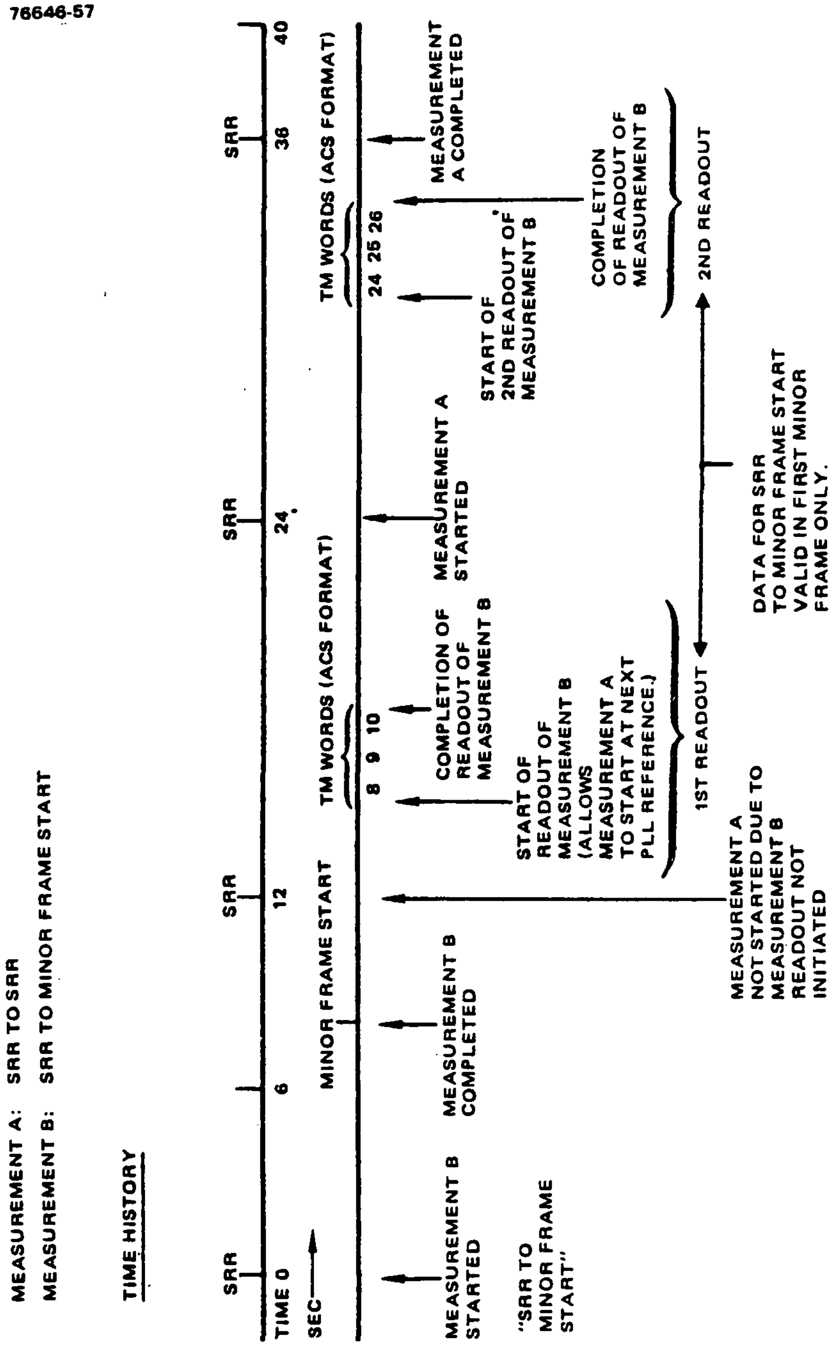


FIGURE 3.3.5-1. ATTM TELEMETRY SAMPLED AT 8 BPS IN MINOR FRAME MODE (ACS FORMAT) AT 5 RPM

Revision

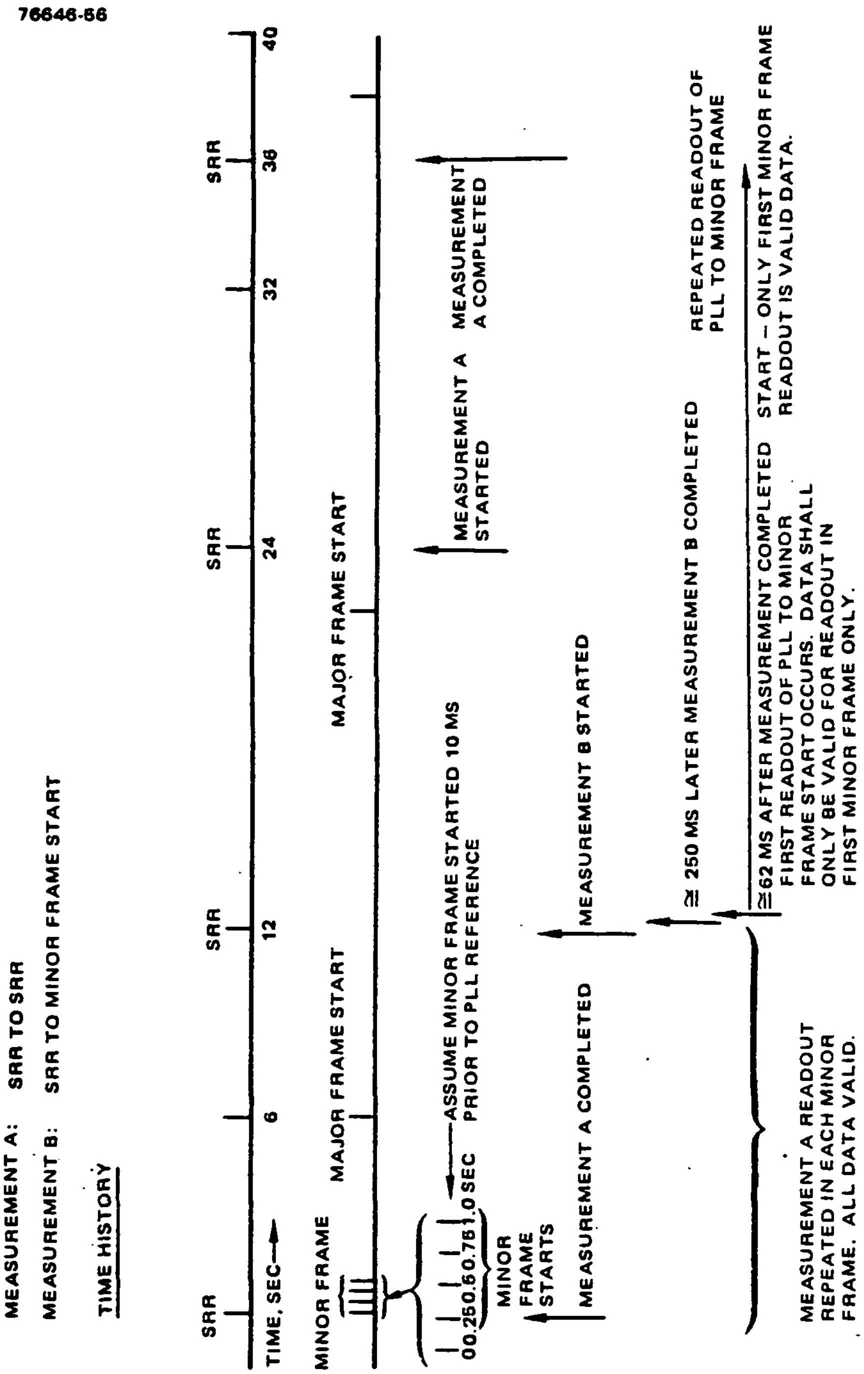


Figure 3.3.3.5-2. ATTM TELEMETRY SAMPLED AT 2048 BPS IN MINOR FRAME MODE (ACS FORMAT) AT 5 RPM

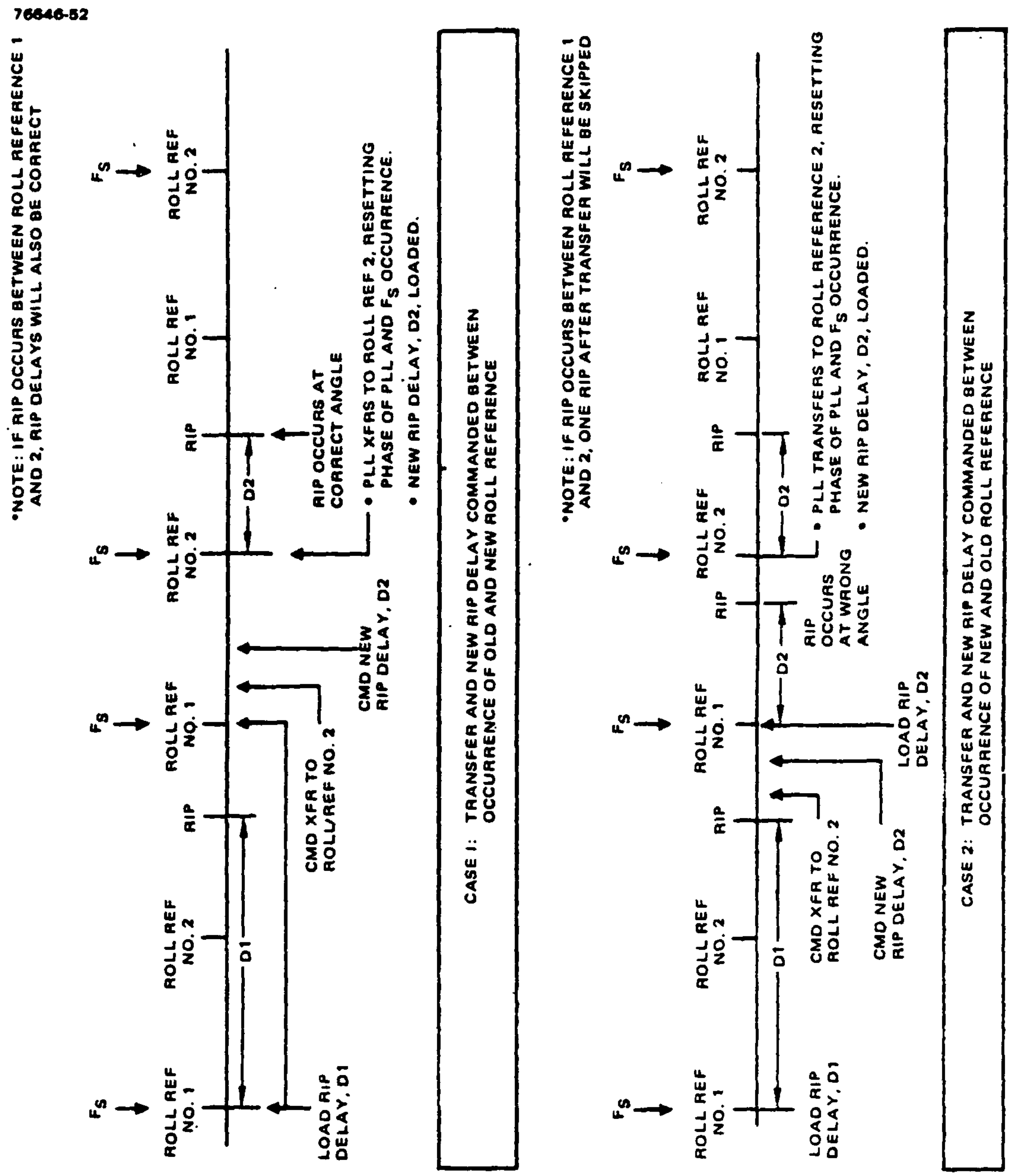


FIGURE 3.3.37-1. RIP GENERATION AT ROLL REFERENCE TRANSFER

This Page Intentionally Left Blank.

3.4 PROPULSION SUBSYSTEM

3.4.1 Subsystem Description. The propulsion subsystem provides the propulsive capability to perform all spin rate, velocity and attitude changes required of the Multiprobe during its mission. These maneuvers are performed by electrically opening the appropriate latch valves and thruster valves and thus allowing pressurized hydrazine, a monopropellant, to flow to the thruster where it is catalytically decomposed to produce thrust. The subsystem operates in the blowdown mode; i.e., the subsystem volume is constant and the pressure is unregulated so that the subsystem pressure drops as propellant is used. Helium is used as the pressurant gas.

The propulsion subsystem, shown schematically in Figure 3.2.2.1.1-1, consists of the following units and components:

- 2 Tanks for propellant storage
- 6 Thruster assemblies for providing thrust (1 lb_f nominal) in specific directions
- 2 Latch valves to isolate the thrusters into two redundant groups
- 4 Filters to prevent thruster valve contamination
- 2 Reservoirs for initial spin-up propellant storage
- 3 Fill/drain valves for subsystem loading and pressurization prior to launch
- 1 Pressure transducer to measure and telemeter subsystem pressure
- Propellant lines to interconnect the various components
- Heaters to maintain propellant temperature greater than 40°F (5°F above freezing)
- 12 Temperature sensors to measure and telemeter subsystem temperature (plus 2 on the OIM and 1 on the S&A).

The six thrusters are separated into two fully redundant groups by the latch valves. Although most normal maneuvers use thrusters from both groups simultaneously, all maneuvers can be performed with only the thrusters in either group. Each thruster group consists of two radial thrusters and one axial thruster; thrusters R1, R2 and A6 are in one group, and thrusters R3, R4 and A5 are in the other. The two radial thrusters in each group are aligned so that their thrust vectors are parallel to each other. The plane containing these two thrust vectors is canted in order to pass through the spacecraft center of mass. The radial thrusters of the two different groups are canted through different points corresponding to the spacecraft center of mass during different phases of the mission. For the Multiprobe, radial thrusters R3 and R4 are canted through the Multiprobe center of meass at the time of the first trajectory correction maneuver while radial thrusters R1 and R2 are canted through the Multiprobe center of mass after Large Probe separation. This design is dictated by the large shift in center of mass due to Large Probe separation usage and the requirement to minimize cross coupling. Table 3.4.1-1 and Figure 3.4.1-1 give the thruster locations, thrust directions, moment arms and general thruster uses.

The physical layout of all subsystem units and components near the equipment shelf is shown in Figure 3.4.1-2.

3.4.2 Units Descriptions

3.4.2.1 Thruster Assembly. The thruster assembly shown in Figure 3.4.2.1-1 consists of a thruster valve to control propellant flow and a thrust chamber with catalyst beds and an expansion nozzle. A cross sectional view of the valve is shown in Figure 3.4.2.1-2.

Valve operation is controlled by application of electrical power from the valve drivers in the ADP of the controls subsystem to redundant coils actuating a single torque motor. The opening and closing response times are 15 ±1 ms and 10 ±1 ms, respectively. The valve is normally closed and is capable of pulsed or continuous operation by controlling the duration of the power application. Command selectable pulse on times of 128 ms and 512 ms are used for Pioneer Venus. The peak power required is 21 watts per valve.

Series redundant propellant shutoff is provided by utilizing dual in-line tungsten carbide poppets and seats. The downstream poppet is actuated by a flapper which is an integral part of the armature; the upstream poppet is mechanically opened by a pin connected to the flapper. A 0.004 inch gap between the pin and the upstream poppet permits positive leak free propellant shutoff by one poppet even if the other poppet is held open by contamination. The interstage cavity between the two valve seats is 0.68 cc; hydrazine trapped in this volume can be subjected to increasing pressure. If so, it relieves back into the system across the upstream poppet which acts as a relief valve. The valve has a cycle life of 350,000 cycles per formal qualification tests; however, Hughes has placed well over one million cycles on a number of the valves without performance degradation. The leakage rate allowable per seat is 2 scc GN_2 per hour for a new valve and 3 scc GN_2 per hour after exposure to hydrazine. Based on analysis and test (Refer to Hughes' IDC 4115.10.2/742, dated 30 September 1976), a GN_2 leak rate of 2 scc per hour is equivalent to less than 10^{-2} gm per year of hydrazine using Knudsen scaling. If effects of kinematic viscosity are considered for a gas leak rate of 2 scc per hour, then the equivalent hydrazine leakage is equal to or less than

$$10^{-1} \frac{gm/yr}{scc/hr}$$

The possibility of a closed failure of this valve is remote. The design of the seat and poppet precludes mechanical interference. The independently wound electrical coils are fully redundant in that either coil is capable of opening the valve in case of failure of the other.

The thrust chamber, the second part of the thruster assembly, is a completely mechanical component that consists of a catalyst chamber where the propellant is decomposed and an expansion nozzle through which the hot gasses are expanded. The thrust chamber is constructed chiefly of cobalt base alloys capable of withstanding the high temperatures associated with hydrazine decomposition. The nozzle has a removable conical extension added to it so that the thruster plume clears the solar panel substrate. With this extension, the nozzle expansion ratio is increased from its normal 40 to 1 ratio to 563 to 1.

Hydrazine flow through the thruster is controlled by a trim orifice. The required orifice size for a given thruster is determined during thruster assembly where water flow tests are made of the injector and candidate orifices. Computation for the orifice pressure drop required to produce one pound of thrust at 200 psia supply pressure takes into account the chamber pressure, the injector and inlet tube assembly pressure drop, the catalyst bed pressure drop and the throat diameter of the nozzle assigned to the thruster. Should hot firing the thruster reveal performance is out of limits, the thruster is re-orificed until the desired performance is achieved.

During the qualification tests of the thruster for Telesat, Canada, 700 cold starts were demonstrated. Presented below is the relation between the mass flow per pulse at high and low supply pressure:

P_S, Psia	On-Time, Ms	Per Pulse Flow, Lb.
300	128	0.00076
300	512	0.00317
75	128	0.00029
75	512	0.00117

Thruster performance varies as a function of propellant supply pressure and, when in the pulse mode, duty cycle and cumulative number of pulses. Performance of the Bus thrusters is shown in

Figure 3.4.2.1-3, Figure 3.4.2.1-4, and in Table 3.4.2.1-1.

3.4.2.2 __Latch Valve__. The two bistable latch valves are used to ensure that propellant is available in each reservoir for spacecraft spinup and to seal off the two redundant thruster groups. The latch valve changes position (open or closed) upon receipt of the appropriate electrical control signal from the ADP of the controls subsystem. The valve contains a single hard seat with an elastomeric poppet. Permanent magnets are used to latch the valve in the desired position after removal of the control signal from either the opening or closing coils of the torque motor. Allowable leakage is 5.0 scc per hour GN_2. Moving parts are isolated from propellant contact by a welded metal torque tube. The valve has been qualified to a cycle life of 1,000 cycles. A cross-sectional view of the valve is shown in Figure 3.4.2.2-1.

The position of the latch valve is indicated by a telemetry status bit (VALV1S and VALV2S) with a "1" indicating the open position.

3.4.2.3 __Propellant Supply Components and Units__

3.4.2.3.1 __Propellant Tank__. Each titanium propellant tank consists of a 12.8 inch diameter hemisphere welded to an 86° cone (Figure 3.4.2.3.1-1) and is designed for an operating pressure of 400 psia with a safety factor of four.

Each tank has two outlet ports, with the propellant outlet at the apex of the conical section, and the second part, diametrically opposite the propellant outlet, used to equalize the gas pressure with the other tank in the subsystem. Total volume of the propulsion subsystem is 2265 in^3.

The propellant/pressurant interface within the tank is determined by the centrifugal forces associated with spacecraft rotation during the mission. Complete drainage of the tank is assured by the orientation of the outlet.

The nominal tank volume is 1100 cubic inches and will be used in the blowdown mode with up to 35 pounds of propellant per tank. Actual volume of each tank is measured during acceptance test by closely measuring the amount of water it holds at room ambient conditions.

3.4.2.3.2 Propellant Lines. The propellant lines are 0.25 inch O.D. by 0.020 wall titanium tubing. With a minimum burst strength of 25,000 psi and all welded joints, the chance of leakage or breakage of the propellant lines is minimal.

3.4.2.3.3 Propellant Reservoir. The propellant reservoir is a 7.1 cubic inch titanium cylinder that is used to ensure that sufficient propellant is available for initial spin up even if the propellant in the tank has settled away from the tank outlet during zero g coast. One reservoir is used in the feed line for each group of thrusters. After initial spin up, the reservoir serves as part of the propellant line system from the tanks to the thrusters.

3.4.2.3.4 Filters. Four filters, two for each thruster group, are provided to ensure that particulate contamination does not reach the thruster valves where it could cause a failed open condition. The filter, with a 10 micron absolute filter rating is a passive device consisting of electrochemically etched discs mounted concentrically inside the housing. Contaminating particles are trapped and held in the discs.

3.4.2.3.5 Fill/Drain Valves. Three fill/drain valves are provided for ground servicing of the propulsion subsystem. Two valves are used for propellant loading and off loading and one for pressurant loading and off loading. The fill/drain valves, which feature triple redundant seals are closed prior to spacecraft encapsulation and launch. During the mission they serve only as terminations for the lines.

3.4.2.4 Heaters. All elements of the propulsion subsystem except the semicircular run of pressurant line are heated to maintain the propellant temperature at 40°F or higher. The mechanically redundant components and units which are located downstream of the latch valves have non-redundant heaters while the non-redundant components and units upstream of the latch valves have redundant heaters. The heater design and oeprational characteristics are given in Section 3.2, Thermal Design.

3.4.2.5 Engineering Instrumentation

3.4.2.5.1 Pressure Transducer. The pressure transducer is a potentiometer type transducer with a nominal maximum resistance of 5120 ohms. A constant current of one milliamp is supplied to the pressure transducer by the PCM Encoder through the DIM, both of which are part of the data handling subsystem. The voltage drop across the potentiometer (0 to 5.12 vdc) is used as the analog representation of the subsystem pressure (0 to 400 psia). Refer to Appendix A for more information about the telemetry. A cross sectional view of the pressure transducer is shown in Figure 3.4.2.5.1-1.

The main uses of the pressure transducer are in determining the expected thruster performance using the performance versus supply pressure graphs presented in Section 3.4.2.1 and, along with the pressurant temperature sensors, in determining the amount of propellant remaining. The determination of remaining propellant is discussed in Section 3.4.3.2.3.

3.4.2.5.2 Temperature Sensors. Twelve temperature sensors are used on the Bus propulsion subsystem. Table 3.4.2.5.2-1 lists each telemetry channel and sensor location. One temperature sensor is mounted on each of the six thruster valve covers. These sensors measure thermal soak back during thruster firing and thus verify thruster operation in the pulse mode and following continuous mode shutdown. Thruster valve temperatures should be between 40°F and 140°F non-operating and up to 225°F during and shortly after operation. One temperature sensor is mounted on the surface of each propellant tank near the top (hemispherical) end of the tank; these sensors are used to indicate pressurant gas temperature. Using this information and the subsystem pressure from the pressure transducer, the propellant remaining can be determined (see Section 3.4.3.2.3). Pressurant temperature should be between 40°F and 140°F.

Four temperature sensors are placed on the propellant lines at locations determined to be representative of the entire subsystem. These locations are shown in Figure 3.4.1-3. Line temperatures should be between 40°F and 140°F.

All of the temperature sensors are positive coefficient thermal resistors (thermistors). The tank and line sensors are all identical (type 908631-32) with a nominal resistance of 2700 ohms. Because of the higher temperatures of the thruster valves the temperature sensors mounted on these components are slightly different (type 908683-4); their nominal resistance is 3100 ohms.

It should be emphasized that at no time during the mission should any of the propellant temperature sensors read less than 40°F. The freezing point of hydrazine is 35°F; should the hydrazine freeze, catastrophic damage to the propulsion subsystem could occur.

3.4.2.5.3 Latch Valve Status. Each latch valve contains a SPDT status switch that is operated by the valve mechanism. The "valve closed" contact of the switch is grounded and the "valve open" contact is open. A one milliamp constant current is supplied to the center contact of the switch by the PCM Encoder through the DIM both of which are part of the data handling subsystem. When the latch valve is moved to the closed position, the current source is grounded and the input voltage drops below 2.56 vdc. This is interpreted by the data handling subsystem as a "0" bit. When the latch valve is moved to the open position, the current source is ungrounded and the input voltage rises above 2.56 vdc. This is interpreted by the data handling subsystem as a "1" bit.

3.4.2.5.4 Heater Status. Heater status telemetry is identified and discussed in detail in Section 3.8.3.8.

3.4.3 Propulsion Subsystem Operations

3.4.3.1 Initial Conditions. The latch valves and the thruster valves are the only commandable units in the propulsion subsystem. (Propulsion heaters, including those that are commandable, are discussed in Section 3.2, Thermal Design). Initial spacecraft spin up is performed automatically as part of the preprogrammed spacecraft separation sequence (Section 4.2.1). When initial ground station acquisition of the spacecraft occurs sometime after the completion of this sequence, the propulsion subsystem telemetry should indicate that all thruster valves and latch valves are closed.

If operation of the propulsion subsystem is interrupted later in the mission, the thruster valves will return to their normally closed position while the latch valves will remain latched in their position prior to the interruption.

3.4.3.2 Normal Operating Modes

3.4.3.2.1 Latch Valve Usage. The functions of the latch valves are to ensure that propellant is available for initial spin up by trapping it in the reservoirs and to isolate the thruster groups to provide redundancy in the event of a failure. Propellant supply for initial spin up is assured by launching the spacecraft with the latch valves closed and then opening them by the preprogrammed spin up sequence just prior to spin up.

In order to prevent catastrophic loss of propellant in the event of a failure downstream of the latch valves, the latch valves should be closed when the spacecraft is not in communication with the ground. The latch valves should also be closed immediately after any maneuver that uses the radial thrusters in order to prevent large spin speed changes in the event of a thruster valve open failure. During extended maneuvers with the axial thrusters when spacecraft telemetry is being monitored, it is recommended that the latch valves be left open. This recommendation is based on the non-redundant design of the latch valve.

Prior to any maneuver the appropriate latch valve must be opened; i.e., latch valve 1 must be opened for any maneuver requiring thruster R3, R4, A5 and latch valve 2 must be opened for any maneuver requiring thrusters R1, R2 and A6. The appropriate latch valve must be open for thrust to be produced and to prevent possible damage to the thruster valve (see Section 3.4.3.4).

Latch valve position status can be determined directly from telemetry as discussed in Section 3.4.2.5.3.

3.4.3.2.2 Thruster Valve Usage. Since the thruster valves control propellant flow to the thrust chamber, they are the main operational elements of the propulsion subsystem. Thrusters are selected as required for the particular maneuver planned (Section 4.3.1), and the controls subsystem supplies the valve control power at the appropriate times in the spacecraft spin period. The thruster valves can be operated in the continuous fire mode as would be done during spin speed changes and axial ΔV maneuvers and in the pulsed mode as would be done during precession and ΔV maneuvers. In the pulse mode, pulse widths of either 128 ms or 512 ms are supplied by the controls subsystem. Table 3.4.3.2.2-1 details the pulse mode duty cycles at the normal spacecraft spin speeds. The 512 ms pulse is not usable (conservatively) at spin rates greater than 20 rpm because of the large angles transversed during the 512 ms pulse width at higher spin rates results in inefficient propellant usage and the duty cycle detector in the Controls Subsystem precludes any pulsed firing. At spin rates below 20 rpm either pulse width can be used; the 128 ms pulse width results in more exact maneuvers while the 512 ms pulse width results in shorter maneuvers.

Thruster valve operation cannot be directly verified in spacecraft telemetry. The first indirect indication is a decrease in subsystem pressure following the propellant remaining curves given in Figure 3.4.3.2.3-1. A second indirect indication of thruster operation is the thruster valve temperature sensor. During thruster valve operation the temperature increases due to power dissipation in the coils and due to thermal soakback from the hydrazine decomposition. The magnitude and rate of temperature increase depends on initial temperature, duty cycle and propellant supply pressure. In general, the temperature tends to rise slowly; temperature rises of 20°F to 50°F after 10 minutes of pulse mode operation or 100 seconds of continuous operation are typical. After completion of thruster operation, the temperature drops several degrees in the first minute after shutdown and then stabilizes due to thermal soakback. Typical thruster valve temperature profiles are shown in Figures 3.4.3.2.2-1 (pulsed firing) and 3.4.3.2.2-2 (continuous firing).

Continuous operation of a thruster valve can also be detected by its power consumption of 21 watts maximum. The Bus loads current is telemetered in the ACS format that will be used during maneuvers. Assuming the normal case where the Bus loads is dissipating nominal spacecraft power, the Bus loads current should increase during valve operation. The only reason this method cannot be used to verify pulse mode operation is the low sampling rate in relation to the pulse on time. Pulsed operations are verifiable via use of 2048 bps rate or use of the programmable format.

The final method of verifying thruster operation is through changes in spacecraft spin rate, attitude and/or velocity.

3.4.3.2.3 Propellant Remaining. The current plan calls for loading 60 pounds of propellant into the Bus propellant tanks. The amount of propellant remaining can be determined from the pressure transducer telemetry signal (VTANKP), the pressurant temperature telemetry (VTNK1T and VTNK2T), and the propellant remaining curves given in Figure 3.4.3.2.3-1. Curves are given for the minimum, maximum and nominal temperatures. For other temperatures, the following equation may be used:

$$\text{Propellant Remaining} = 0.0364 \left\{ 2265 - \left[\left(\frac{119{,}673}{P} \right) \times \left(\frac{459.6+T}{529.6} \right) \right] \right\}$$

where

P = Tank pressure, psia

T = Tank temperature, °F

3.4.3.3 Abnormal Operating Modes

3.4.3.3.1 Possible Failure Modes. Table 3.4.3.3.1-1 presents a list of general failure modes and corrective actions that are possible for the propulsion subsystem.

3.4.3.3.2 Use of Redundancy. As can be seen from a review of Table 3.4.3.3.1-1, the primary corrective action for a non-catastrophic failure is to use the redundant thruster group. If the failure is anything but a closed failure of a single thruster valve, the appropriate latch valve must be closed thus preventing use of any of the thrusters in that particular group.

In general, the use of only one thruster group causes less efficient use of propellant and more complex maneuvers due to additional cross coupling. Typical maneuvers that can be performed with a single thruster group include:

- Spin speed changes using a single radial thruster for each direction
- Attitude precession maneuvers using a single axial thruster or the radial thruster pair not canted through the c.g. at the time of the maneuver.

- Velocity changes with an axial thruster that may require a large precession maneuver to properly align the thrust vector.

Each of these maneuvers is discussed in detail in Section 4.3, (Maneuvers).

3.4.3.4 Operational Restrictions. The following operational restrictions had been determined to be necessary for the longevity and proper operation of the propulsion subsystem.

- The appropriate latch valve must be open before the operation of any thruster valve. If the thruster valve is subjected to continuous power for approximately 60 seconds without propellant flow, permanent damage to the thruster valve coils can result.
- The thruster valve temperature must be 40°F or higher prior to operation to prevent damage to the thruster. This minimum temperature should be maintained automatically by the subsystem heaters.
- The subsystem temperature must be 40°F or higher at all times to prevent propellant freezing which would rupture subsystem components. This minimum temperature should be maintained automatically by the subsystem heaters.
- Prior to opening a latch valve, telemetry should be checked to assure all heaters downstream of the valve are operating. This will determine if it is safe to open the latch valve in the event a heater has failed or a line or component has ruptured.

TABLE 3.4.1-1

BUS THRUSTER GEOMETRY

Thruster	Location	Thrust Plume Direction*	Usage
R1	$\theta = 82^\circ$ STA = 68.75	$\theta = 52^\circ 30'$ Canted 18.68° Aft	+ Spin; ΔV & Precession (Post L. P. Separation)
R2	$\theta = 22^\circ$ STA = 68.75	$\theta = 52^\circ 30'$ Canted 18.68° Aft	- Spin; ΔV & Precession (Post L. P. Separation)
R3	$\theta = 262^\circ$ STA = 68.75	$\theta = 232^\circ 30'$ Canted 28.50° Aft	+ Spin; ΔV & Precession (Cruise)
R4	$\theta = 202^\circ$ STA = 68.75	$\theta = 232^\circ 30'$ Canted 28.50° Aft	- Spin; ΔV & Precession (Cruise)
A5	$\theta = 105^\circ$ R = 46 in.	Forward from STA 82.4	ΔV; Precession
A6	$\theta = 285^\circ$ R = 42 in.	Aft from STA 63.5	ΔV; Precession

*Direction of Spacecraft Motion is opposite.

TABLE 3.4.2.1-1

CUMULATIVE TOTAL IMPULSE TABLE

Pulse No.	CUMULATIVE TOTAL IMPULSE (LB-SEC) AT INDICATED DUTY CYCLE, RPM, AND SUPPLY PRESSURE							
Duty Cycle	128/3872	512/3488	128/1872	128/3872	512/3488	128/872	128/11872	512/11488
RPM	15	15	30	15	15	60	5	5
P_s, PSIA	300	300	225	200	200	200	75	75
1	0.08	0.47	0.06	0.06	0.34	0.05	0.03	0.16
2	0.19	1.05	0.14	0.14	0.77	0.13	0.08	0.35
3	0.31	1.66	0.24	0.23	1.21	0.21	0.12	0.55
4	0.44	2.29	0.34	0.32	1.67	0.29	0.16	0.76
5	0.57	2.94	0.45	0.42	2.15	0.39	0.20	0.97
6	0.70	3.59	0.57	0.52	2.63	0.49	0.25	1.19
7	0.84	4.24	0.69	0.62	3.12	0.59	0.29	1.41
8	0.98	4.90	0.82	0.72	3.61	0.69	0.34	1.63
9	1.13	5.56	0.94	0.83	4.11	0.80	0.38	1.86
10	1.27	6.22	1.07	0.94	4.60	0.91	0.43	2.09
15	2.02	9.56	1.72	1.50	7.09	1.46	0.66	3.23
20	2.80	12.91	2.40	2.09	9.61	2.03	0.91	4.37
30	4.39	19.63	3.78	3.29	14.64	3.22	1.41	6.65
50	7.63	33.08	6.62	5.73	24.70	5.65	2.44	11.13
100	15.76	66.82	13.78	11.84	49.93	11.80	4.94	21.76

TABLE 3.4.2.5.2-1

TEMPERATURE SENSOR LOCATIONS

VTNK1T	Tank 1 Temperature, $\theta = 105^{\circ}$	
VTNK2T	Tank 2 Temperature, $\theta = 285^{\circ}$	
VLIN1T	Propellant Line	See Figure 3.2.1-1.
VLIN2T	Propellant Line	See Figure 3.2.1-1.
VLIN3T	Propellant Line	See Figure 3.2.1-1.
VLIN3T	Propellant Line	See Figure 3.2.1-1.
VJET1T	R1 Thruster Valve Cover, $\theta = 82^{\circ}$	
VJET2T	R2 Thruster Valve Cover, $\theta = 22^{\circ}$	
VJET3T	R3 Thruster Valve Cover, $\theta = 262^{\circ}$	
VJET4T	R4 Thruster Valve Cover, $\theta = 202^{\circ}$	
VJET5T	R5 Thruster Valve Cover, $\theta = 105^{\circ}$	
VJET6T	R6 Thruster Valve Cover, $\theta = 285^{\circ}$	

TABLE 3.4.3.2.2-1

PROPULSION SUBSYSTEM
PULSE MODE DUTY CYCLES

SPACECRAFT RPM	DUTY CYCLE
5	128 ms ON, 11,872 ms OFF 512 ms ON, 11,488 ms OFF
10	128 ms ON, 5872 ms OFF 512 ms ON, 5488 ms OFF
15	128 ms ON, 3872 ms OFF 512 ms ON, 3488 ms OFF
30	128 ms ON, 1872 ms OFF
60	128 ms ON, 872 ms OFF

TABLE 3.4.3.3.1-1

POSSIBLE FAILURE MODES

Possible Failure Mode	Method of Recognition	Corrective Action
1. Thruster valve fails open.	With no input to thruster valve and L/V open, T/V temperature high; pressure decreasing.	Close appropriate L/V; use redundant thruster group.
2. Failure of non-redundant heater downstream of L/V.	Temperature sensor telemetry.	Close appropriate L/V; use redundant thruster group.
3. Leakage failure downstream of L/V.	Only when L/V open S/S pressure decreases T/V temperature normal.	Close appropriate L/V; use redundant thruster group.
4. Latch valve fails closed.	Telemetry signal, thrusters inoperable.	Use redundant thruster group.
5. Thruster valve fails closed.	T/V temperature remains normal or decreases prematurely. S/S pressure remains normal during "operation" of ADP control.	Use other thrusters.
6. Leakage failure upstream of L/V.	With L/V closed S/S pressure decreases while tank temperature remains constant.	None.
7. Failure of redundant heaters upstream of L/V.	Temperature sensor telemetry.	If possible, maintain temperature by shelf heaters and/or sun angle change.
8. Failure of both thruster group non-redundant heaters.	Temperature sensor telemetry.	Same as 7.

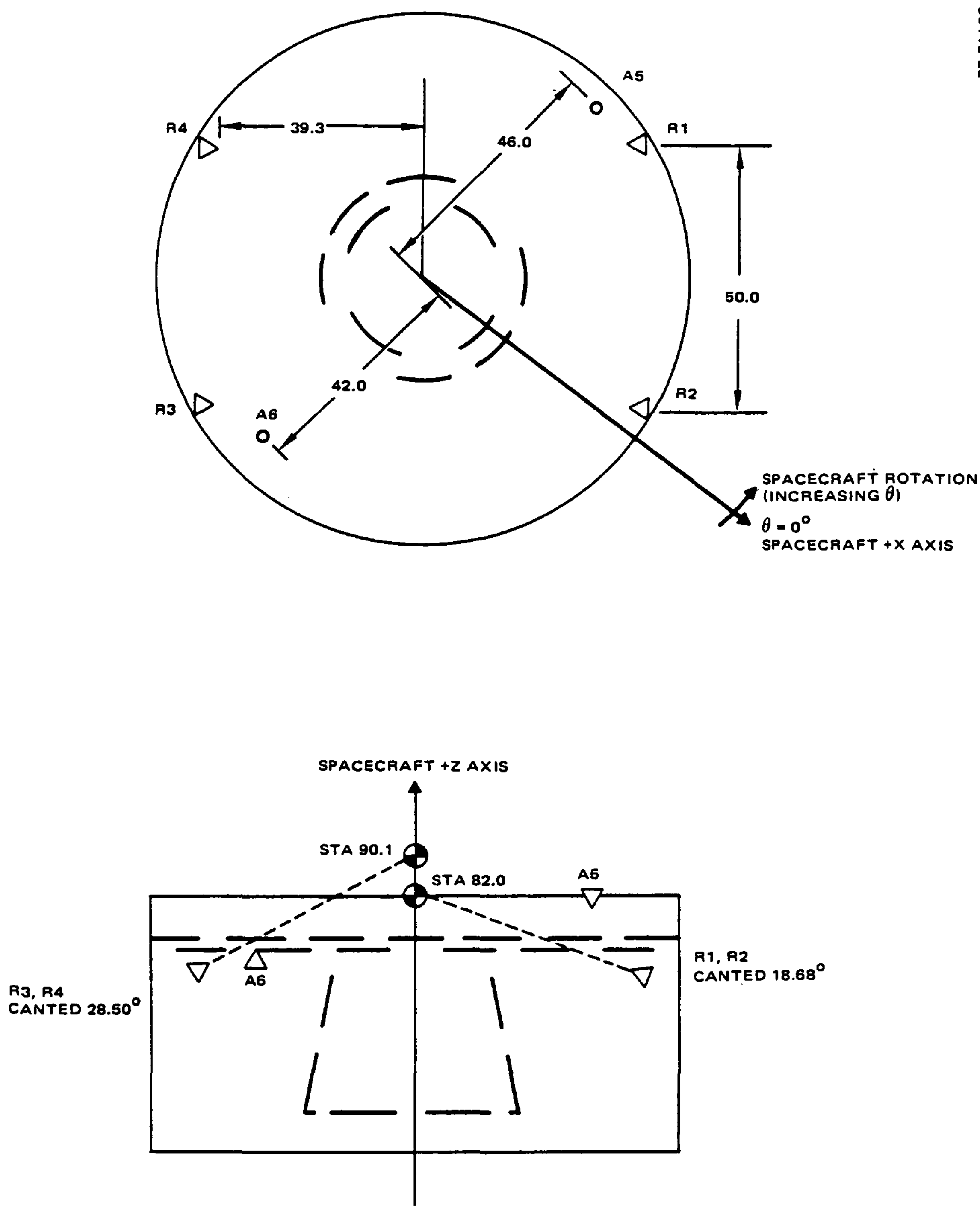


FIGURE 3.4.1-1. MULTIPROBE THRUSTER LAYOUT

Revision

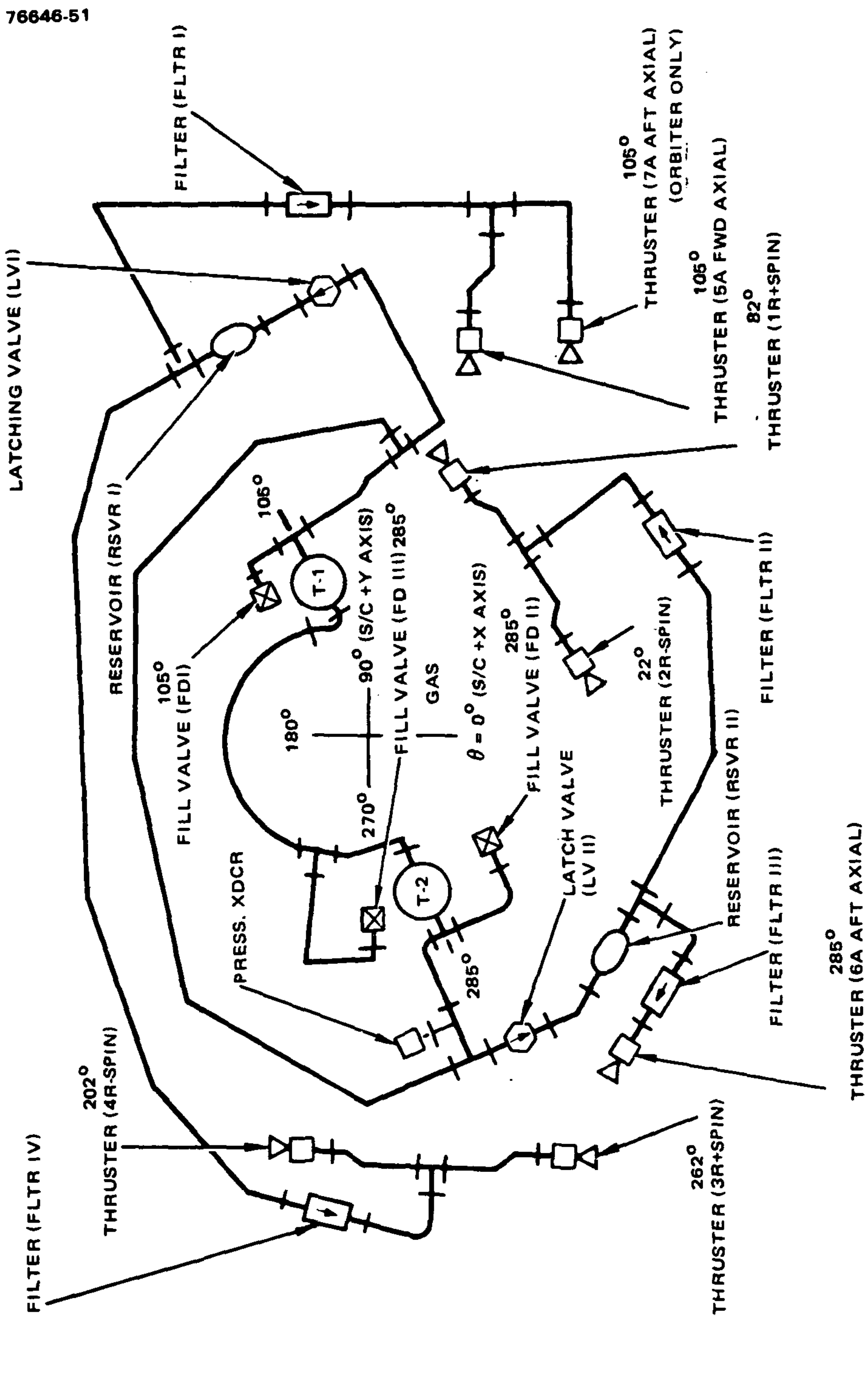


FIGURE 3.4.1-2. PROPULSION SUBSYSTEM LAYOUT

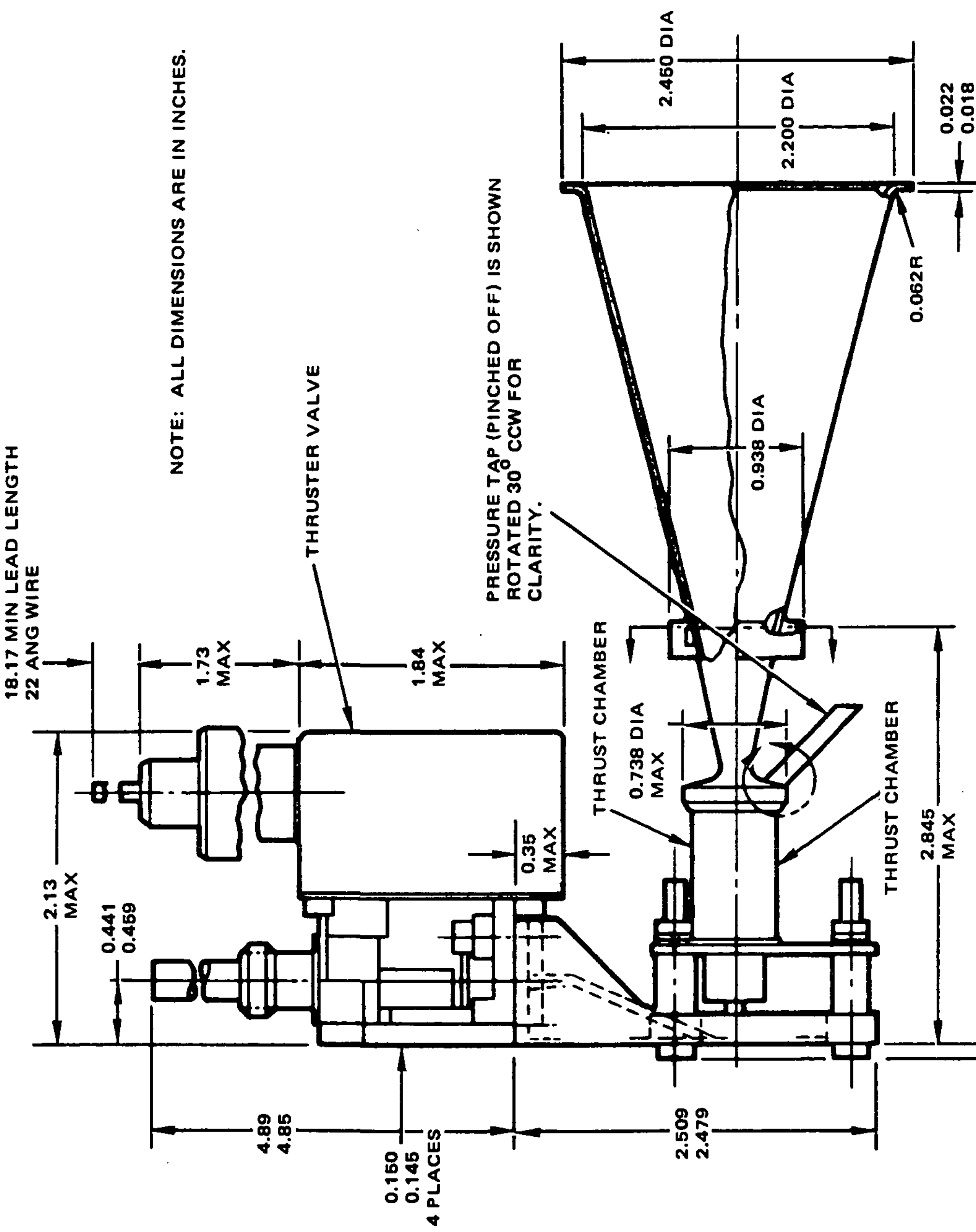


Figure 3.4.2.1-1. Thruster Assembly

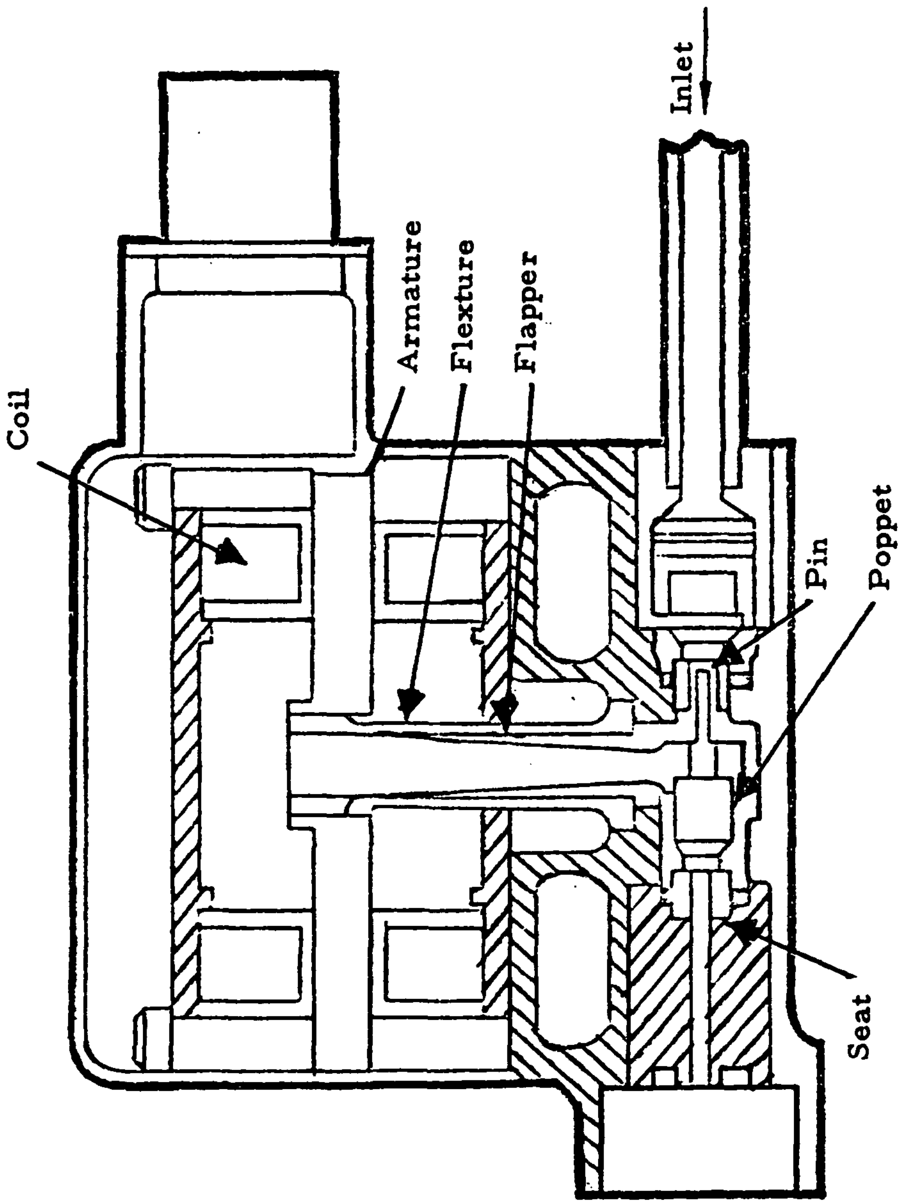


Figure 3.4.2.1-2. Thruster Inlet Valve

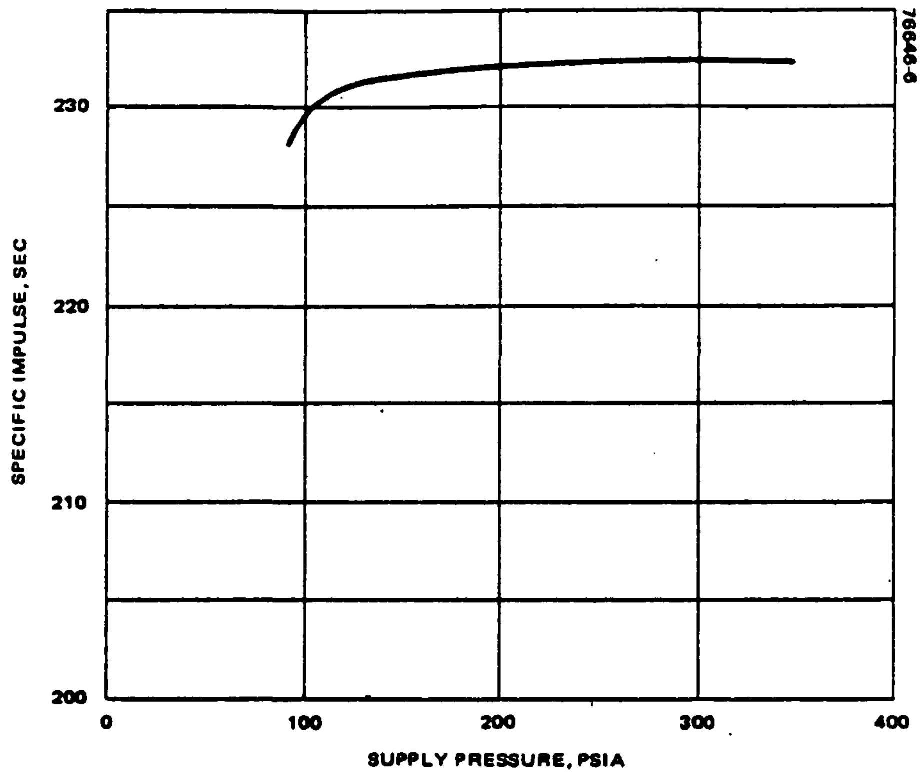


FIGURE 3.4.2.1-3. STEADY STATE SPECIFIC IMPULSE (T = 30 SEC) VERSUS SUPPLY PRESSURE

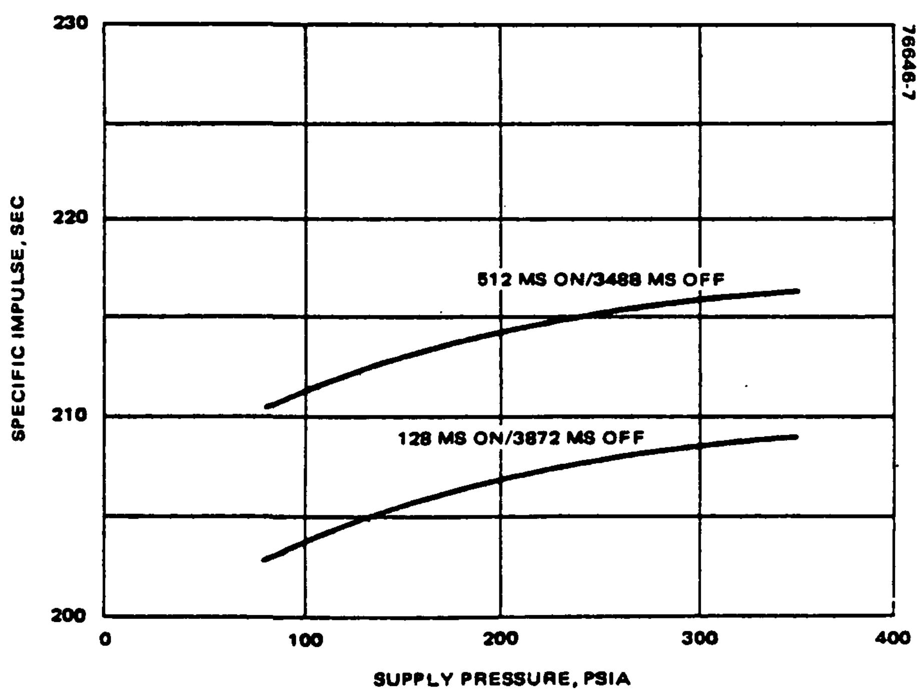


FIGURE 3.4.2.1-4. PULSE MODE SPECIFIC IMPULSE (NOMINAL)

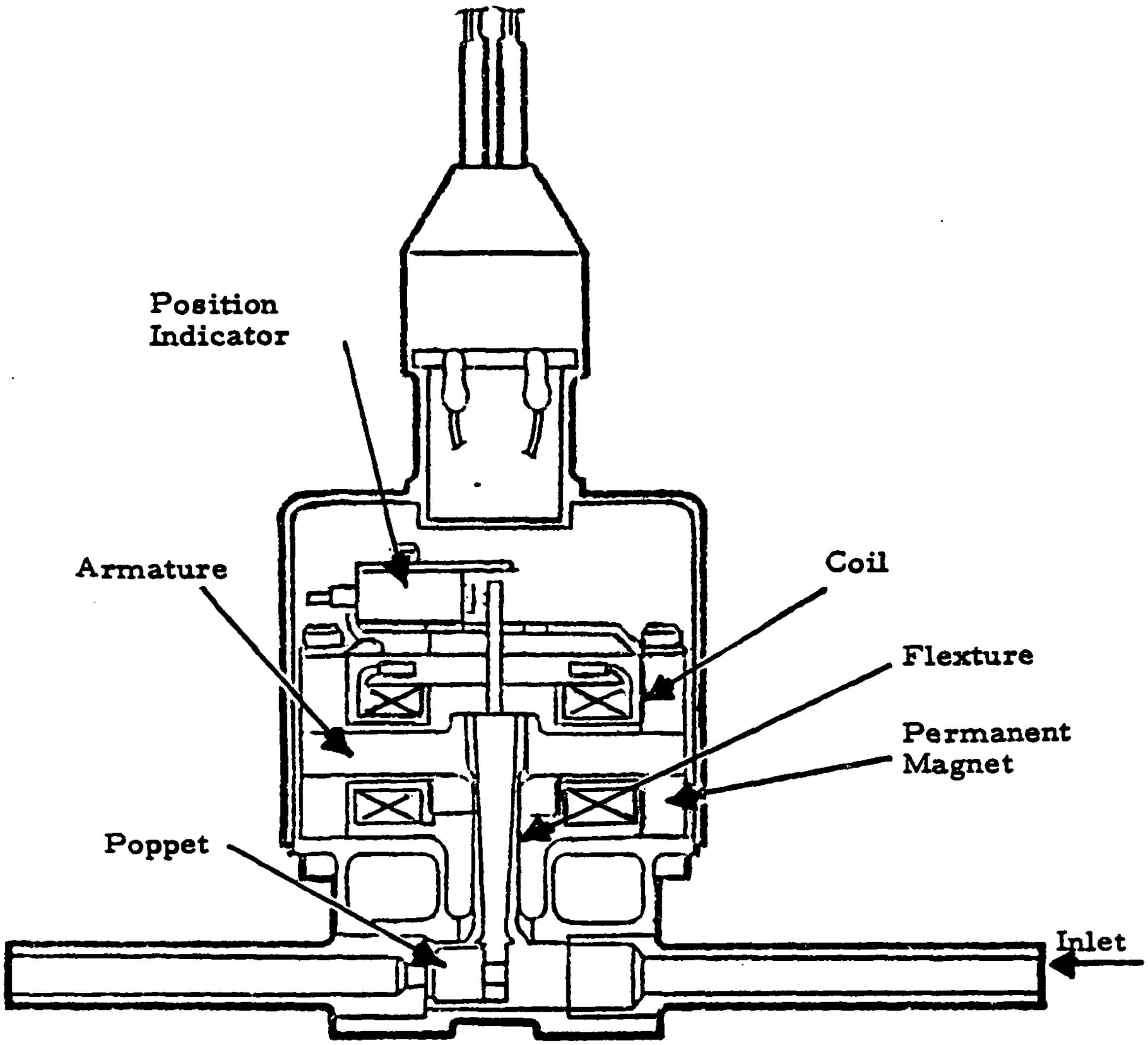


Figure 3.4.2.2-1. Latch Valve

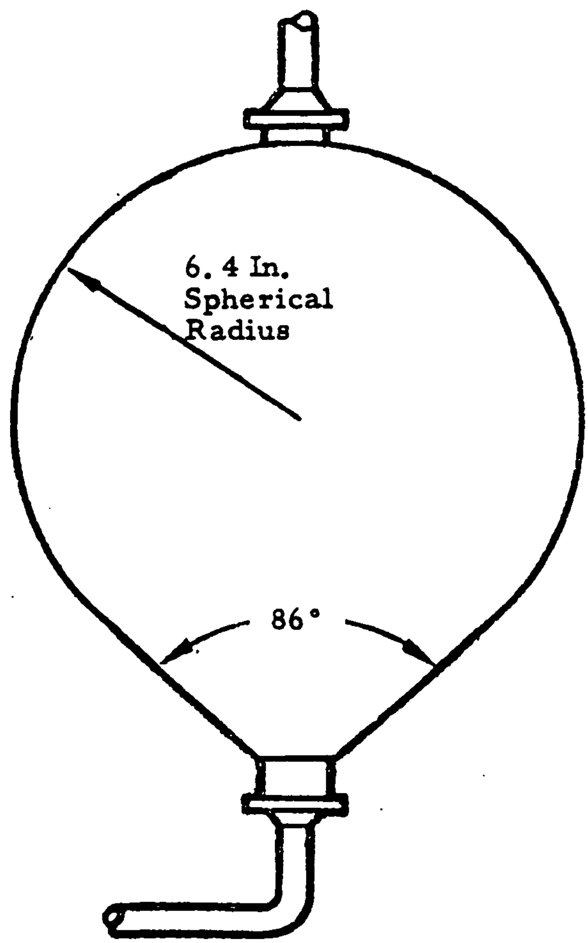


Figure 3.4.2.3.1-1. Hydrazine Tank

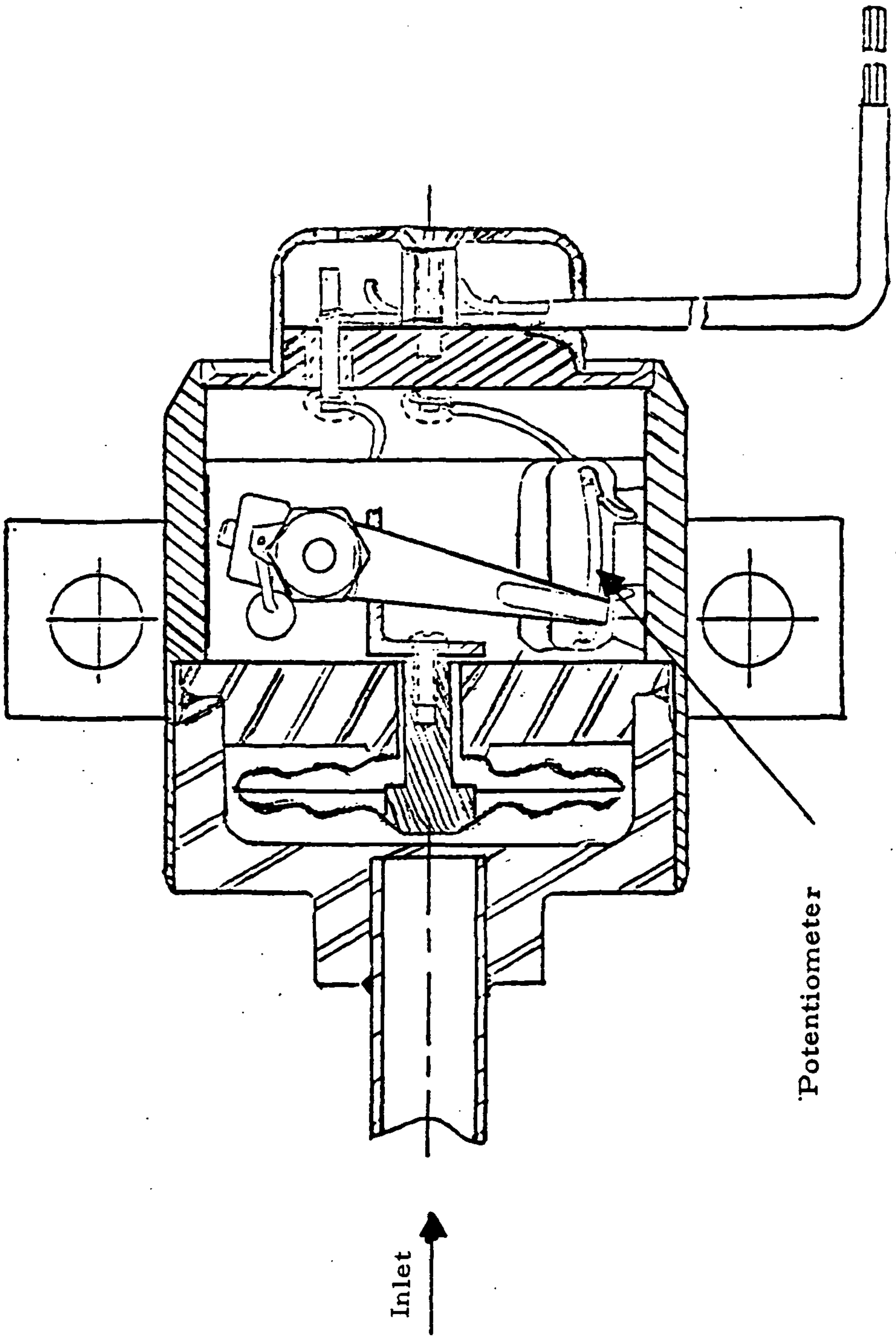


Figure 3.4.2.5.1-1. Pressure Transducer

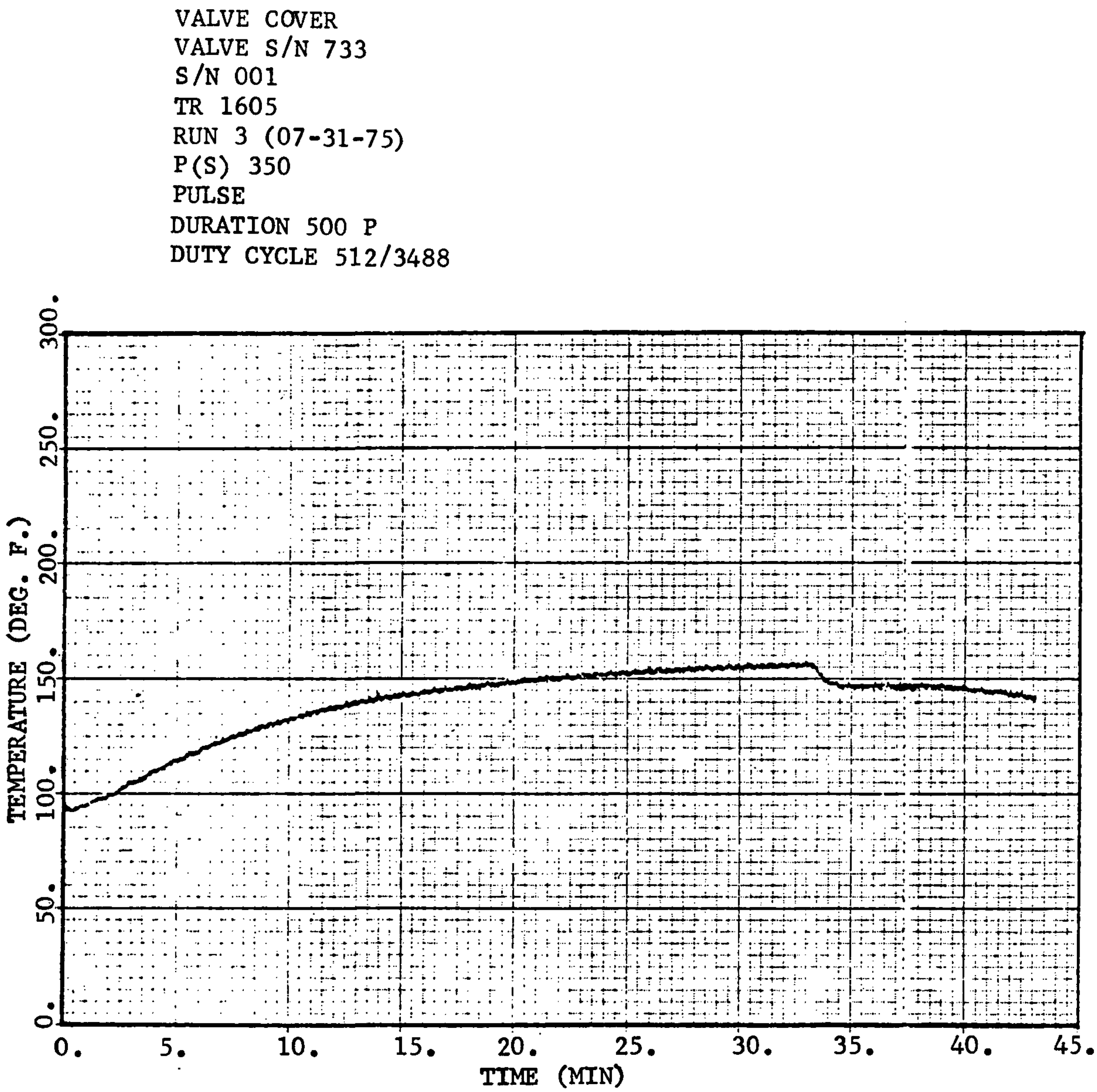


Figure 3.4.3.2.2-1. Typical Thruster Valve Temperature Profile-Pulsed Firing

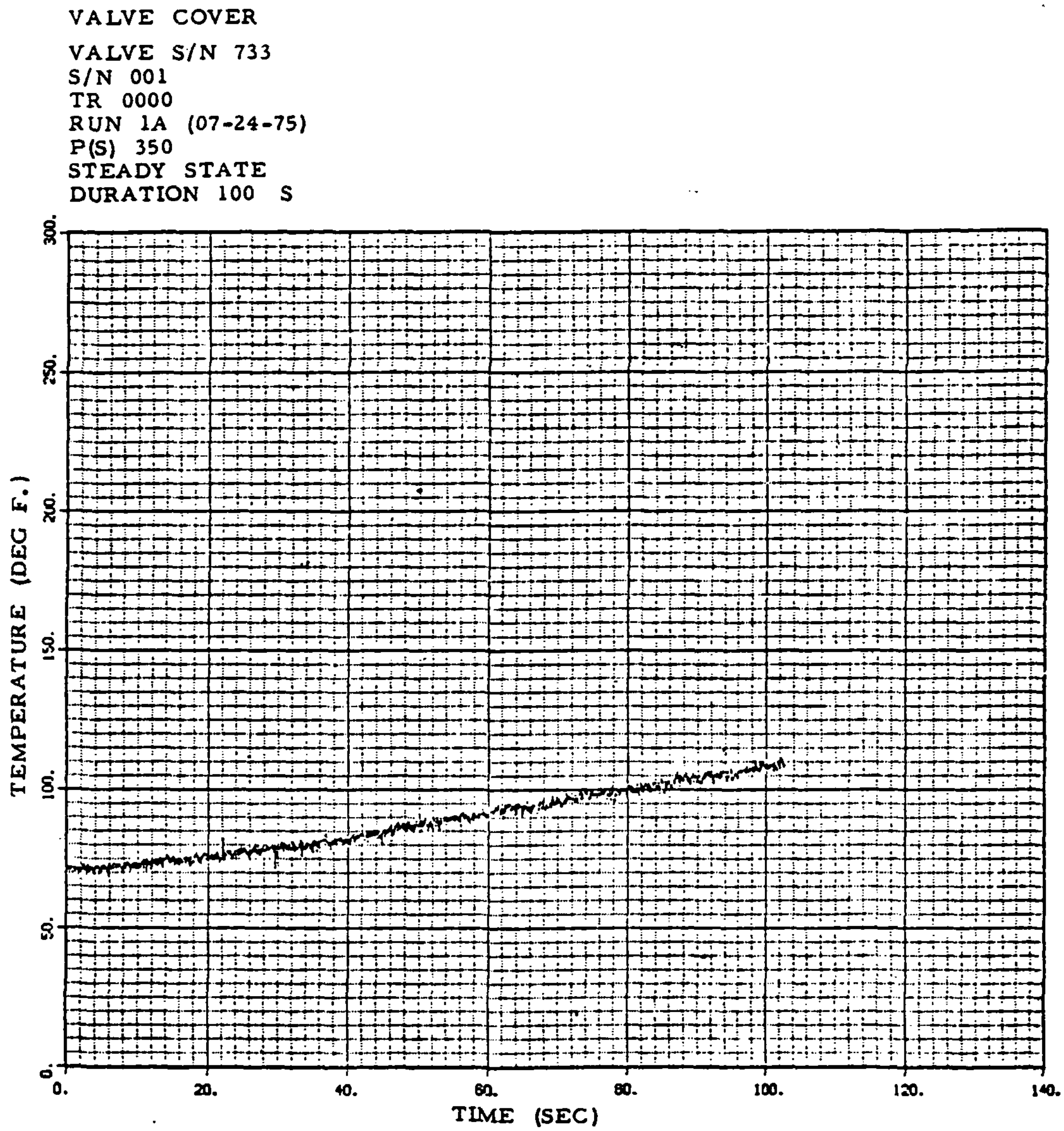


Figure 3.4.3.2.2-2. Typical Thruster Valve Temperature Profile - Continuous Firing

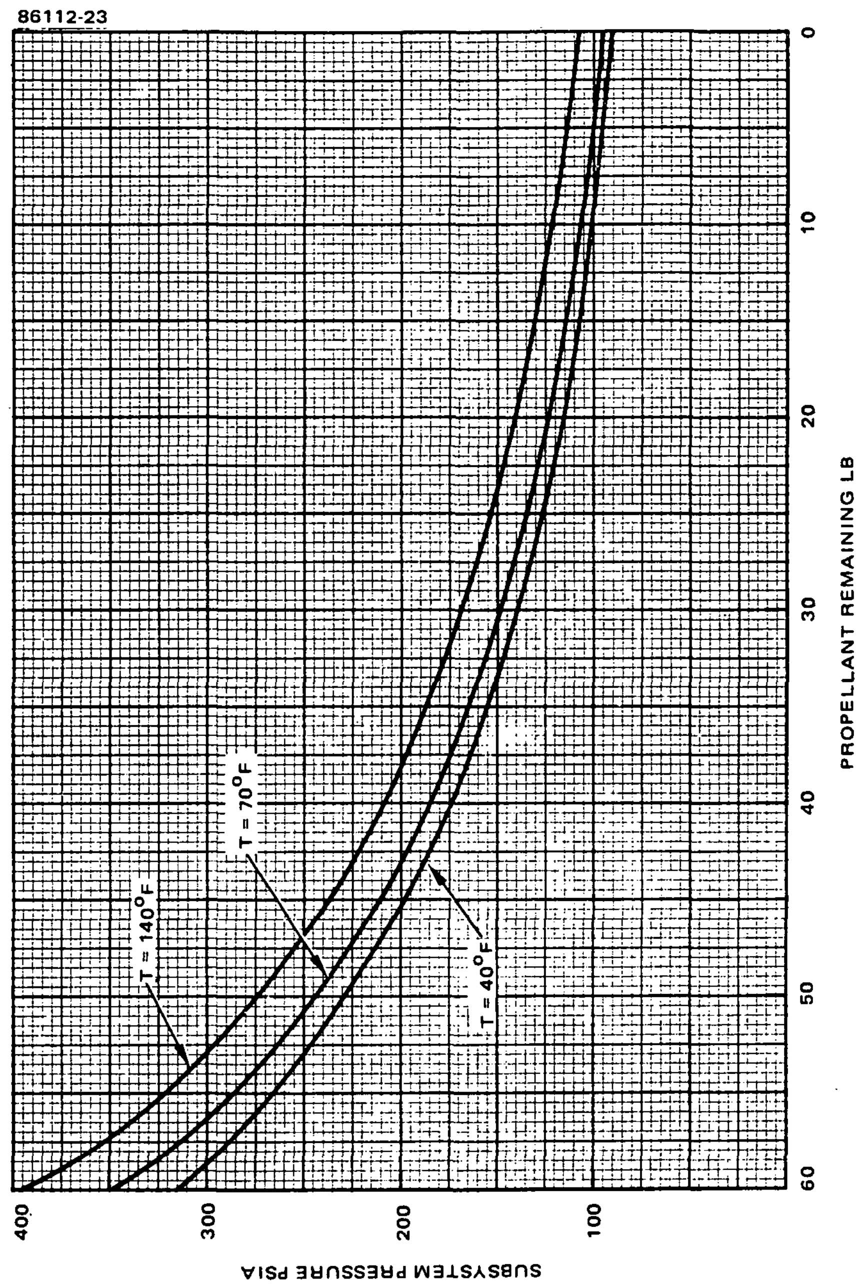


FIGURE 3.4.3.2.3-1. MULTIPROBE PROPELLANT REMAINING

This Page Intentionally Left Blank.