TRW SYSTEMS

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PHASE IA TASK B

FINAL TECHNICAL REPORT

VOYAGER SPACECRAFT

Volume 2
PREFERRED DESIGN: SUBSYSTEMS

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I. INTRODUCTION

Volume 2 presents the TRW preferred design for all subsystems of the Voyager spacecraft, corresponding to Section IV of the JPL description of the Final Technical Report. It also describes the planetary vehicle adapter and indicates interface requirements assumed for the spacecraft science payload. The volume has four major parts, covering 1) spacecraft bus subsystems, 2) propulsion subsystem, 3) science subsystem, and 4) planetary vehicle adapter.

In accord with JPL practice, each subsystem is considered under five basic headings: 1) general description, 2) requirements and constraints, 3) functional interfaces, 4) design description, and 5) parameters and performance summary. Whenever pertinent, design alternatives and growth potential for the subsystem are also discussed.

The design criteria for the subsystems discussed here are derived from the studies reviewed in Volume 1. In that volume, over-all spacecraft requirements and constraints are defined, and the mission sequence, telemetry list, reliability assessments, and power budget are summarized.

Many of the subsystems are essentially identical to those described in the Task A report. Changes that have been introduced are based on revised JPL requirements or tradeoff studies conducted during Task B. Of the changes made during Task B, the most significant are:

- Propulsion Subsystem. Use of the variable thrust LEM descent engine for midcourse corrections and orbit injection instead of the solid engine for orbit injection and the monopropellant for midcourse correction.

- Structure Subsystem. Use of the LEM descent stage structure as the basic flight spacecraft structure instead of a newly developed structure.

- Radio Subsystem. Use of larger antennas and greater power output (50W) to provide up to 15,000 bits per second transmission rate. In addition, the omni-directional antenna is now on a 14-foot boom which is deployed after separation from the launch vehicle. The capsule radio link was deleted as a portion of the radio subsystem.

- Data Storage. Deletion of a solid-state buffer storage unit and provision of three additional tape recorders for science and engineering data.

- Guidance and Control Subsystem. Changes to accommodate LEM engine requirements and the use of two cold gas thrust levels, 0.2 pound for cruise, and 3 pounds for maneuvers. The limb and terminator crossing sensors and an accelerometer are added to the subsystem.

- Power Supply. Enlarged solar array (280 sq. ft.) and a new module configuration. The power distribution function and the spacecraft time-reference oscillator are now included in the power subsystem.

Since portions of the description of this command subsystem are classified confidential, the complete subsection covering that subsystem, Section 7, has been separately bound. It is conceived, however, as a part of this volume.

II. SPACECRAFT BUS SUBSYSTEMS

1. POWER SUBSYSTEM

1.1 General Description

The power subsystem provides power in suitable forms for distribution to electrical equipment on board the flight spacecraft and to the flight capsule until its separation. Primary power is derived from the sun by means of silicon photovoltaic cells, mounted on a fixed solar array. Secondary silver-cadmium batteries are used to energize the spacecraft whenever the solar array is incapable of supporting the loads, as during launch, maneuvers, and eclipses. Appropriate controls are provided to maintain proper functioning of the subsystem.

A simplified block diagram of the power subsystem is shown in Figure 1. The elements of the power subsystem are:

- Solar array (8 panels)
- Shunt elements assemblies (2)
- Power control unit
- Batteries (3 in parallel)
- Battery regulators (1 per battery)
- 4 KHz inverters (1 main and 1 standby)
- 400 Hz inverters (1 main and 1 standby)
- Power Distribution Unit

1.2 Requirements and Design Constraints

1.2.1 Mission Constraints

The power subsystem must function continuously for at least 14 months, including 6 months in a Mars orbit. The distance between the sun and the spacecraft may vary from 1.0 AU to 1.67 AU during the mission. The distance at capsule separation may be as large as 1.47 AU, corresponding to a latest arrival date at Mars of 5 January 1972, plus ten days before capsule separation.

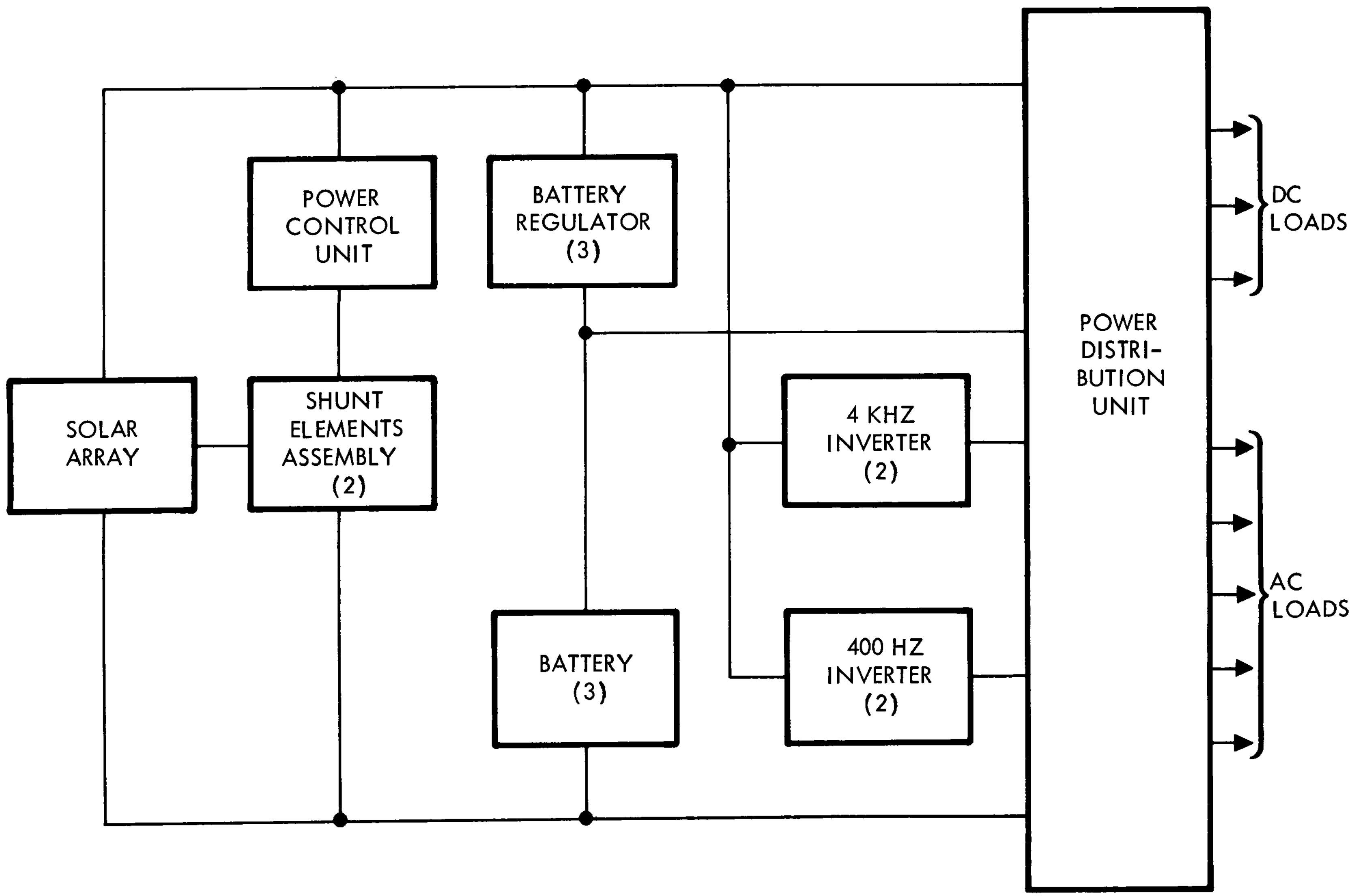


Figure 1. Simplified Power Subsystem Block Diagram

The batteries must be capable of supplying all applicable loads (except capsule load) during the following periods:

- Launch to acquisition: maximum duration of 230 minutes
- Manuevers: a maximum duration of 120 minutes for each trajectory correction, orbit insertion, and orbit trim maneuver
- Eclipses: a maximum duration of 138 minutes in orbit. Eclipse season begins 3 months after arrival.

1.2.2 Functional Requirements

The power subsystem will satisfy the following functional requirements

- Provide electrical power from a primary power source using solar energy for spacecraft operation
- Provide a secondary electrical power source in the form of storage battery power, and provide power from the primary source for recharge. Provide electrical power from the secondary power source when the primary source cannot handle the load.
- Condition power for spacecraft use: voltage, frequency, waveform, phase, noise level
- Provide power to the flight capsule from the solar array during periods of sun-line orientation
- Provide centralized switching and distribution of power
- Provide a precision oscillator for synchronization and for generation of telemetry, computer, and sequencer functions

1.2.3 Performance Requirements

Estimated average load power requirements are listed in Table 1 as a function of mission phases. Peak power requirements are listed in Table 2. For purposes of this power system analysis, the average values have been assumed to be representative steady state limits. The estimated load peaks constitute low energy requirements supplied by the batteries in those cases where the instantaneous power demand exceeds the solar array capability.

The power requirement of 128 watts for reaction control gas heaters is shown as a nonessential load and is not included in the total figures.

Table 1. Estimated Average Power Profile (Watts)

Subsystem Identification	Loads	Form	Prelaunch	Launch and Injection	Acquisition	Cruise (after battery charging)	Trajectory Corrections	Orbit Insertion	Planetary Vehicle Orbit Trim	Capsule Separation	Flight Spacecraft Orbit Trim	Flight Spacecraft Orbital Operations	Planetary Vehicle Orbital Operations (with battery charging) Sunlight	Planetary Vehicle Orbital Operations (with battery charging) Eclipse	NOTES
Radio	Electronics	4 kc	14	20	20	12	12	12	12	12	14	12	12	12	
Telemetry	"	"	5	5	5	5	5	5	5	5	5	5	5	5	
Command	"	"	13	13	13	13	13	13	13	13	13	13	13	13	
C and S	"	"	20	20	20	20	20	20	20	20	20	20	20	20	
Data storage	Tape recorders	"	17	0	0	7	4	4	17	4	17	4	17	17	33% duty cycle (maneuvers)
Science	Instr. and DAE	"	14	14	14	57	57	57	78	78	78	78	78	78	
GCS	Electronics	"	27	27	29	27	33	37	27	32	29	32	29	29	
Subtotal	4 kHz Inv. output	"	110	99	101	141	144	148	172	164	176	164	174	174	
	4 KHz Inv. input	50 VDC	128	118	120	165	168	173	201	192	206	192	203	203	85% max. eff.
Thermal Control	Heaters	"	0	5	15	44	54	54	24	34	34	34	34	34	
	RCS Heaters	"	0	36	36	(128)	36	36	(128)	36	36	36	(128)	36	128 W nonessential
Capsule	Unspecified	"	0	0	0	200	0	0	200	0	0	0	0	0	
Radio	TWTA	"	0	0	0	150	150	150	150	150	150	150	150	150	
GCS	Gyro Inv. input	"	0	0	8	0	8	8	0	8	8	8	8	8	
CCS	400 Hz Inv. input	"	1	1	2	2	10	56	3	3	3	3	3	3	
Propulsion	Gimbal actuators		0	0	0	0	([illegible]8.5 w-hr)	(70 w-hr)	0	(4.3 w-hr)	0	(3 w-hr)	0	0	Direct from batt. bus Energy in w-hr
	Valves	29-42 VDC	0	0	0	0	(0.2 w-hr)	(0.[illegible] w-hr)	0	(1.5 w-hr)	0	(1 w-hr)	0	0	
Total	Loads	50 VDC	129	160	181	561	426	177	578	423	437	423	398	434	Excluding gas heaters and propulsion spikes
Power	Batt. + Chg. Reg.	"	0	0	0	0	0	0	125	0	0	0	125	0	Cont. batt. charging
	Boost Reg	"	32	40	45	6	106	119	6	106	109	106	6	108	80% efficiency
	PCU	"	10	10	10	10	10	10	10	10	10	10	10	10	
Solar array	Output required	"	-	-	-	577	-	-	719	-	-	-	539	-	Excl. gas heaters
Battery	Output required	29-42 VDC	171	210	236	-	542	606	-	539	556	539	-	55-2	Excl. propulsion spikes
Battery Output	With science off							524						440	Excl. propulsion spikes

Table 2. Estimated Peak Power Profile (Watts)

Subsystem Identification	Loads	Form	Prelaunch	Launch and Injection	Acquisition	Cruise (with battery charging)	Trajectory Corrections	Orbit Insertion	Planetary Vehicle Orbital Operations (with battery charging)	Planetary Vehicle Orbit Trim	Capsule Separation	Flight Spacecraft Orbit Trim	Flight Spacecraft Orbital Operations		Notes
													Sunlight	Eclipse	
Radio	Electronics	4 kc	23	20	20	12	12	12	12	12	14	12	12	12	
Telemetry	"	"	5	5	5	5	5	5	5	5	5	5	5	5	
Command	"	"	13	13	13	13	13	13	13	13	13	13	13	13	
C and S	"	"	20	20	20	20	20	20	20	20	20	20	20	20	
Data storage	Tape recorders	"	60	0	0	12	12	12	60	12	60	12	60	60	
Science	Instr. and DAE	"	135	14	14	57	57	57	135	135	135	135	135	135	
GCS	Electronics	"	69	69	74	72	80	80	72	80	74	80	74	74	
Subtotal	4 kHz Inv. output	"	325	141	146	191	199	199	317	277	321	277	319	319	
	4 kHz Inv. input	50 VDC	382	167	173	224	233	233	372	325	376	325	374	374	
Thermal Control	Heaters	"	0	10	20	49	59	59	49	59	49	59	59	59	
	RCS Heaters	"	0	36	36	(128)	36	36	(128)	36	36	36	(128)	36	128 W nonessential
Capsule	Unspecified	"	0	0	0	200	0	0	200	0	0	0	0	0	
Radio	TWTA	"	0	0	0	150	150	150	150	150	150	150	150	150	
GCS	Gyro Inv. input	"	0	0	8	0	8	8	0	8	8	8	8	8	
GCS	400 Hz Inv. input	"	1	1	38	38	66	66	53	53	53	53	53	53	
GCS	Valves	"	0	0	48	48	118	118	48	118	48	118	48	48	
Total	Loads	50 VDC	383	214	323	709	670	670	872	749	720	749	692	728	Excluding propulsion spikes
Power	Batt. + Chg. Reg.	50 VDC	0	0	0	125	0	0	125	0	0	0	125	0	
	Boost Reg.		97	54	81	6	170	170	6	187	180	187	6	182	
	PCU	50 VDC	10	10	10	10	10	10	10	10	10	10	10	10	
Solar array	Output required	50 VDC	-	-	-	850	-	-	1013	-	-	-	833	-	Excluding gas heaters
Battery output	Required	29-42 VDC	490	278	414	-	850	850	-	946	910	946	-	920	Excluding propulsion spikes

These heaters increase reaction control gas specific impulse, reducing gas requirements, but are used only when the solar array capability exceeds the essential load requirements by a suitable margin.

The required solar array outputs during the various mission phases in which the solar array is illuminated by and oriented to the sun are summarized in Table 3.

Table 3. Solar Array Outputs

Mission Phase	Solar Array Output Required (Watts)
Cruise	577
Planetary vehicle orbital operation	719
Flight spacecraft orbital operation	539

The battery energy requirements given in Table 4 consist of average spacecraft loads plus short duration loads for propulsion valves and engine gimbal actuators.

Table 4. Battery Energy Requirements

Mission Phase		Battery Output Energy Required (w-hr)
Prelaunch, launch, injection and acquisition		944
Trajectory correction		1103
Orbit insertion	283	1283
Planetary vehicle orbital trim		1084
Capsule separation		1112
Flight spacecraft orbital trim		1082
Eclipses in Mars orbit (150 cycles)		1270

The power subsystem must be capable of accommodating transient loads up to 150 per cent of the normal steady state loads for any load switched in flight.

Synchronization will be provided, controlled from a precision oscillator in the power subsystem. The frequency accuracy for inverter synchronization will be ±0.01 per cent. Inverters will be capable of free running.

1.2.4 Design Requirements and Constraints

The solar array will be insulated on the dark side to control heat transfer between the array and the spacecraft, and to improve eclipse survival capability.

For the purpose of sizing the solar array at the design point, the α_{s}/ϵ of the filtered solar cells is assumed to be no less than 1.0.

The intensity of trapped radiation at Mars is assumed to be 10^{-5} times that of earth, with identical particle energy distribution.

Battery design conservatism is in accordance with Table 5.

Table 5. Conservative Battery Characteristics

Characteristic	Sealed Zinc Silver Oxide	Sealed Silver Cadmium
Maximum energy density (w-hr/lb)	50	30
Minimum energy specific volume (in^3/w-hr)	0.50	0.59
Maximum cycles at 30% depth of discharge	500	2,000

The power handling capability of the booster regulator will be 400 watts maximum per unit, its efficiency 82 per cent. Power handling capability of the shunt regulators will be 100 watts maximum per unit. Efficiency of the electrical inverters will be 90 per cent maximum at rated load. Voltage regulation will be no greater than ±2 per cent for load changes varying from half of rated load to full load.

1.3 Functional Interfaces

1.3.1 Flight Capsule

The power subsystem provides 200 watts of DC power from the solar array to the capsule during periods of sun line orientation. The distribution of power within the capsule is controlled by the capsule. No spacecraft switching condition or single failure mode will allow the capsule batteries to supply power to the spacecraft. The spacecraft power subsystem provides turn-on and turn-off capabilities of the power to the capsule during high usage periods and during any capsule status check.

1.3.2 Other Spacecraft Subsystems

The solar array consists of separate panels mounted directly to the spacecraft structure. The maximum diameter of the array is 240 inches. No deployment is required. The spacecraft orientation during cruise attitude will be such that the solar array receives solar radiation with incidence angles within ±1 degree of normal.

The power subsystem precision oscillator provides the frequency source for the generation of telemetry, computer, and sequencer functions.

Command requirements for power are given in Table 6.

The power subsystem provides sufficient monitoring points to document redundancy switching either automatically or by ground command. It also provides check points for diagnostic purposes and for evaluation of space environment on the subsystem as a whole and on individual critical components. A list of these telemetry monitoring points is given in Volume 1.

Power subsystem inputs are as follows:

- External power (50 VDC) to operate the spacecraft during prelaunch and checkout
- Control signals to operate switching circuits in the power distribution unit

Table 6. Power Subsystem Command Requirements

Unit	Number of	Number of Commands
Battery No. 1, charge control override	on/off	2
Battery No. 2, charge control override	on/off	2
Battery No. 3, charge control override	on/off	2
Battery No. 1, disconnect	open/close	2
Battery No. 2, disconnect	open/close	2
Battery No. 3, disconnect	open/close	2
Charge rate select	Charge rates 1 and 2 regulator No. 1 Charge rates 1 and 2 regulator No. 2 Charge rates 1 and 2 regulator No. 3	6
Reset synchronization switch logic	Main/standby	2

- Command signals to provide for inflight override of automatic switching functions and to select battery charging rates.

Power subsystem outputs are as follows:

- Main 50 VDC ±1 per cent bus for capsule, TWTA, heaters, and gyro inverter
- 50 V ±2 per cent rms, 4 KHz, single-phase, square wave bus for science, guidance and control, data storage, C and S, command, telemetry, and radio subsystems
- 50 V ±2 per cent rms, 400 Hz, two-phase, square wave bus for antenna and PSP drives
- 29-42 VDC battery bus for engine gimbal actuators and propulsion valves
- Conditioned analog telemetry signals representing power subsystem voltages and currents
- Timing frequencies to C and S and telemetry subsystems.

Each flight spacecraft subsystem is provided with its own power supply to convert the distributed 4 KHz AC power from the power subsystem to the required user internal supply voltages.

1.4 Design Description

1.4.1 Subsystem Description

A block diagram of the recommended power system for the Voyager mission is shown in Figure 2. The system consists of seven major functional elements: solar array, solar array shunt voltage limiter, power control unit, secondary battery, battery regulator, power conditioning inverters, and power distribution unit.

The solar array output is limited to 50 VDC ±1 per cent by shunt regulation of a portion of each series string of solar cell modules. Power dissipation in the shunt elements assembly is minimized by a unique sequential shunt configuration. The shunt elements are controlled from bus voltage sensing and error signal amplifier circuitry located in the power control unit (PCU).

The battery is charged from the 50-volt bus through a simple dissipative current regulator. Charging is terminated by a control signal from individual temperature-compensated cell voltage sensors mounted on the battery cells. When the highest cell voltage decreases below a preset level, constant current charging is again initiated. If the solar array does not have sufficient capability to charge the batteries, the PCU voltage sensing and error amplifier circuits supply an over-riding control signal to de-energize the charge regulator.

When the solar array is incapable of supporting the system load, the battery discharges through a boost regulator to maintain the bus at 50 volts. A switching-type boost regulator is used, which is controlled by the PCU voltage sensor and error amplifier circuits.

The three 30-cell, 26-ampere-hour, silver-cadmium batteries, each with a charge-discharge regulator, are operated in parallel under normal conditions. In the event of a battery or regulator malfunction, the associated battery and regulator are disconnected by the power switching and logic circuitry in the PCU. Two of the three batteries are capable of supporting essential spacecraft loads through maneuver and eclipse phases.

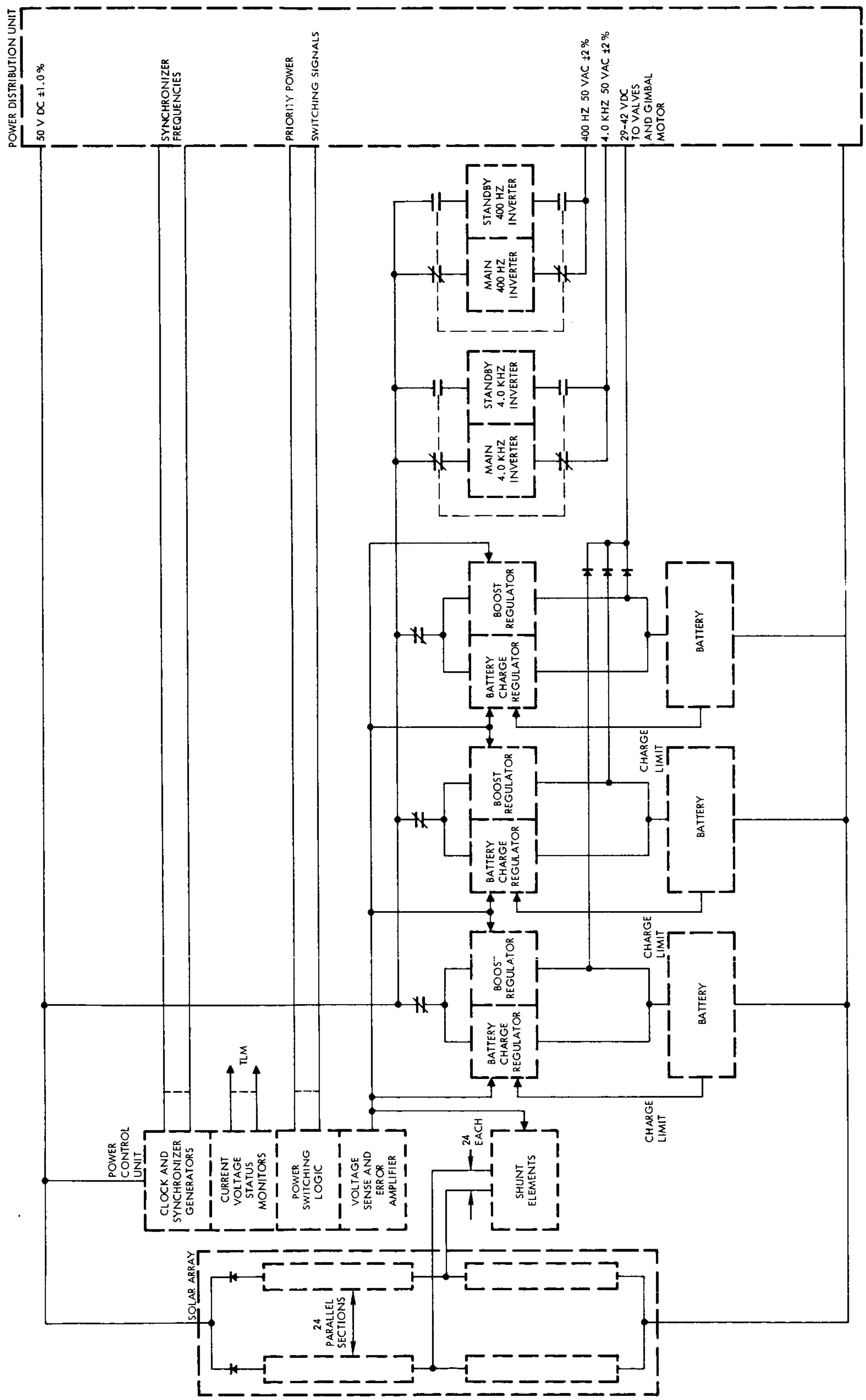


Figure 2. Voyager Electric Power Subsystem Block Diagram

The two main outputs from the system are the regulated 50-VDC ±1 per cent bus and a 50 VAC ±2 per cent, 4.0 KHz, single-phase, square wave bus.

A simple unregulated inverter is used to supply the 4-KHz AC output. Sequential inverter redundancy is provided by sensing AC bus undervoltage and switching to a standby inverter in the event of inverter failure. This sensing and switching function is performed by the power distribution unit. The majority of the loads are energized through transformer rectifier units from the AC bus. These rectifiers are considered part of the load equipment and may also include supplementary output regulators where required. Additional 400 Hz two-phase power is provided to supply AC power to the antenna and planetary scan package drive motors. Sequentially-redundant units are provided in the same manner as in the case of the 4.0-KHz inverter.

The 50-watt communications transmitters include regulated DC-DC converters to supply the several closely regulated (±0.5 per cent) DC voltages required by the tubes. As a result, these loads are energized directly from the 50-VDC bus. The use of rectifiers supplied from the 4.0-KHz bus to energize the TWT's is less advantageous in this instance because of the difficulty in providing close regulation of the high level DC voltages required by the tubes.

High-current, short-duration requirements for propulsion are supplied directly from the battery bus. These include solenoid valve and gimbal motor power.

A spacecraft synchronizing signal generator is provided in the PCU which generates sync frequencies. Some of these are used internally for synchronization of the boost regulators and inverters. The remainder of the synchronization frequencies are distributed throughout the spacecraft. The current and voltage monitors in the PCU provide conditioned analog signals to the spacecraft telemetry system for monitoring power system performance throughout the mission. All power, including regulated and unregulated DC, 4-KHz and 400 Hz AC, plus synchronization frequencies, are distributed throughout the spacecraft from the power distribution unit. In addition, all power switching for programmed loads is performed in the power distribution unit.

1.4.2 Subsystem Elements

a. Solar Array Design. A preliminary specification of the solar array is shown in Figure 3. The solar array consists of eight identical interchangeable panels, each of which contains six strings of eight parallel by 150 series-connected 2 x 2 cm silicon solar cells mounted on an aluminum honeycomb substrate 1 inch thick. The panels are mounted directly to the structure of the spacecraft by fasteners inset into the rear face in the appropriate pattern.

Solar Cells. The solar cell used is an n- on-p, one ohm-cm, 2 x 2 cm cell, 0.012 inch thick. It uses sintered titanium contacts on opposite sides of the silicon wafer, and has an installed, modularized efficiency of 9.5 per cent with coverglass and filter.

Cover Glass and Filters. The cover glass is 0.006 inch slide of Corning fused silica type 7940. A blue filter with a 50 per cent cutoff point at 400 microns is applied to the inner glass surface.

Cover Glass Adhesive. Sylgard 182 is used as the coverglass adhesive.

Solar Cell Module Buses. The module buses are silver plated molybdenum, with solder wetting inspection holes and thermal stress relief provisions. The top bus (N tab) is 0.006 inch thick and 0.1 inch wide. The bottom bus (P tab) is 0.002 inch thick and shaped as shown in Figure 4.

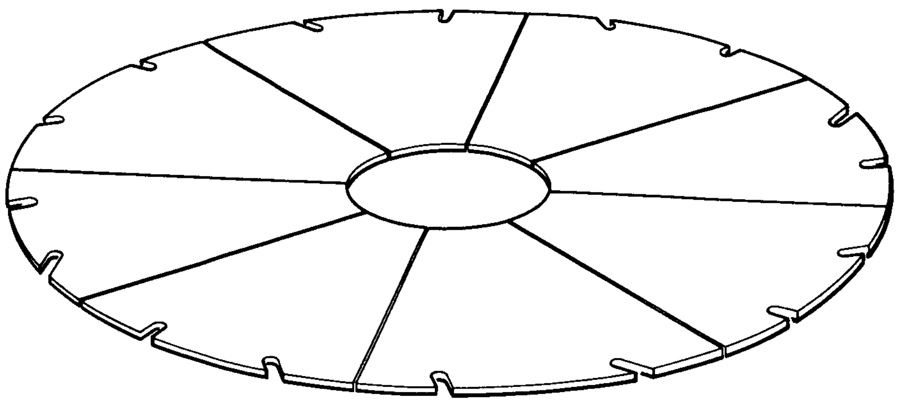
Solder. The solder selected for connecting N and P buses to the solar cells has the following composition: tin, 62 per cent; lead, 36 per cent; and silver, 2 per cent.

Module Assembly. Modules of eight cells in parallel are assembled in a 4 x 2 configuration, using a single central P collector strip with two groups of four cells arranged edge to edge, and N collector strips along the front contacts. See Figure 4.

Module Bonding Adhesives. The module to substrate bonding adhesive selected is a blend of RTV-560 and RTV-580. Each cell is to be mounted on an individual pedestal 0.006 to 0.016 inch thick.

PRELIMINARY SPECIFICATION

Solar Array



Array, Physical Characteristics

Outer diameter:	240 inches
Inner diameter:	62 inches
Number of panels	8
Total number of cells:	57,600
Area	284 square feet
Weight	290 pounds
Number of cells in parallel	384
Number of cells in series	150
Packing factor	0.842 (active Si)

Module Characteristics

Number of cells in parallel	8
Number of cells in series	1
Glass thickness	0.006 inches
Filter	400μ Blue
Efficiency	9.5% 27°C AMO
Cover material	Corning 7940

Performance Characteristics

Nominal power output at 50 VDC

1 AU	1010 watts
1.47 AU	808 watts
1.67 AU	600 watts

Cell Characteristics

Size:	2 x 2 cm
Base resistivity	1.0 ohm-cm
Type	N on P

Figure 3. Preliminary Specification—Solar Array

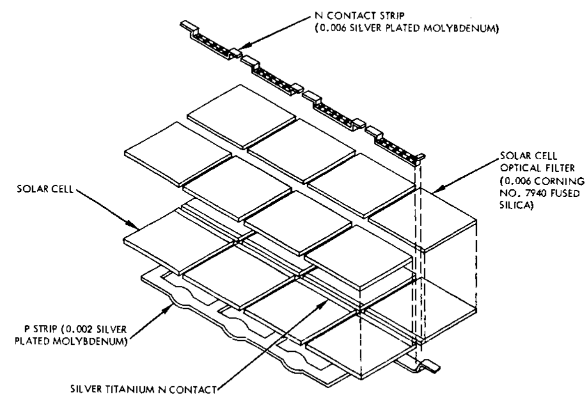


Figure 4. Eight-Cell Module Assembly Exploded View

Substrate Electrical Insulation. The substrate dielectric coating tentatively selected, pending final minimum upper allowable bacteriological decontamination bake temperatures, is glass cloth impregnated with SMP 62/63, 0.0035 inch thick. The SMP, when subjected to the temperature range required for bacteriological decontamination, will experience discoloration; however, initial tests indicate that required properties of the SMP will not be affected by this temperature exposure.

Substrate Materials. The substrate is a flat aluminum alloy honeycomb panel. Covers are 0.010 inch thick and core is 3/8 cell and 0.0015 inch thick. Core and covers are bonded together using FM 1000 film of thickness equivalent to 0.04 pound per square foot. All aluminum alloy will be 5052H39. The core depth is a nominal 1 inch.

Wiring. All wiring will be multistrand T.F.E. fluorocarbon insulated hook-up wire. Heat sinks will be employed to inhibit solder from wicking up the stranded wires, thus causing rigid wiring at the solder joints. Each string of 150 modules will have a separate positive and a separate negative lead connected from the array to the spacecraft bus. Wire routing will be such as to cancel small magnetic loops developed in the module string layout.

Solar Panel Layout. A layout of a typical solar panel is shown in Figure 5.

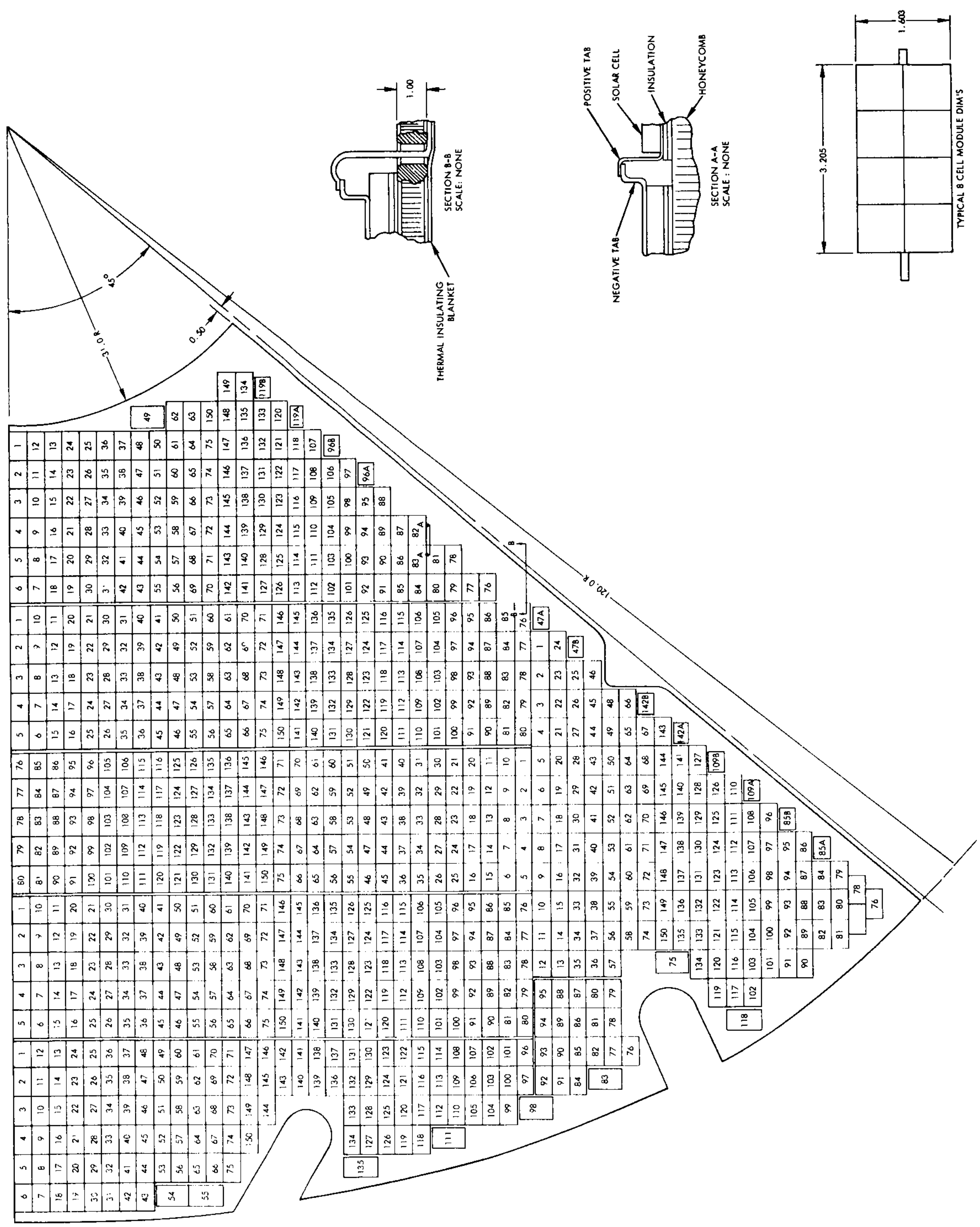


Figure 5. Proposed Voyager Solar Panel for Phase IA Task B Study

b. Battery Design

Since loss of power during maneuvers constitutes a mission failure, a redundant battery system is provided. A preliminary battery specification is shown in Figure 6.

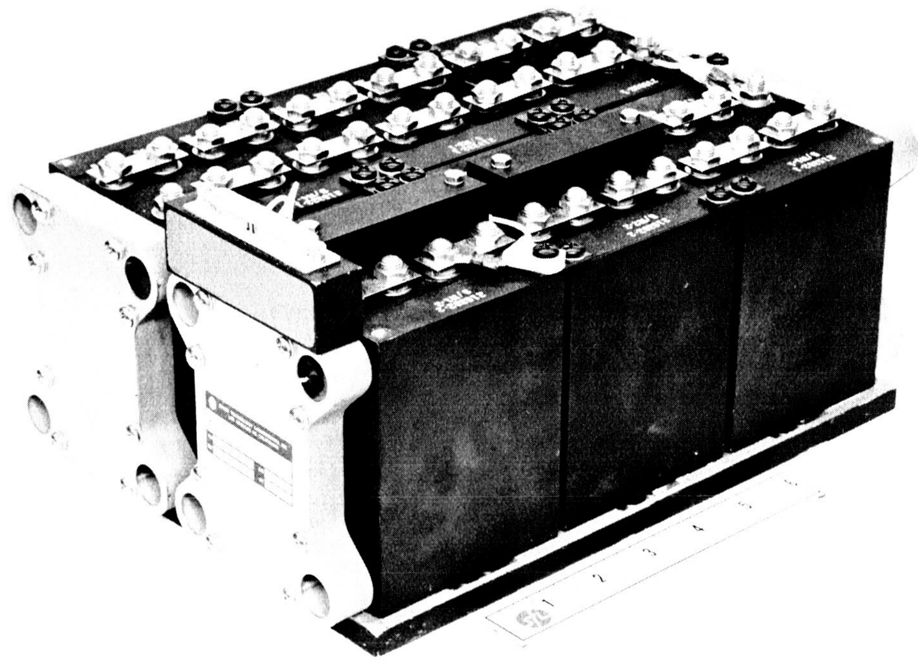
In normal operation, all loads are shared by three identical silver-cadmium batteries. With a 50-volt bus and a nominal peak charge voltage per cell of 1.6 volts, a 30-cell battery permits use of a simple dissipative current limiter with a cell level sensing feedback for charge control. The reasons for selecting silver-cadmium over silver-zinc cells, as well as the design of the battery, except for a small capacity change, and the relationship between battery and charge controls is identical with that desdribed in Phase 1A, Task A Report, Volume 5, Section IV-3.6.

The most critical design requirement is that any two batteries must be capable of supplying the essential loads if one battery fails. Table 1 indicates that the maximum power required is 606 watts (scientific experiments "on") and 524 watts with experiments "off" at orbit insertion. The maximum energy rate required at the end of the mission is 552 watts with experiments "on," and 440 watts with experiments "off."

The essential load is 524 watts of battery discharge power, with scientific loads turned off. The discharge energy required for a 2-hour maneuver is then 1048 watt-hours or 524 watt-hours per battery in a single battery failure mode. Allowing a maximum depth of discharge of 80 per cent (based on a 150-cycle life), the required capacity per battery is 655 watt-hours. The corresponding capacity at a nominal 30 volts is 22 ampere-hours. These data are based upon nominal performance at an temperature of 75°F. Empirical data indicates an approximate 20 per cent capacity loss from 75°F to 50°F and the required single battery capacity is therefore 26 ampere-hours and the total installed battery capacity is 78 ampere-hours, which is approximately equivalent to 63 ampere-hours (1890 watt-hours at 30 volts) at 50°F.

The end of mission capacity requirements (1012 watt-hours, science off) permits an 80 per cent depth of discharge under worst case conditions (50°F ambient) and in the single battery failure condition.

PRELIMINARY SPECIFICATION
Battery



Performance Characteristics
(SINGLE BATTERY)

NOMINAL CAPACITY:	
At 75°F	26 ampere hours
At 50°F	21 ampere hours
Voltage range	28.5 volts to 43.2 volts
Average discharge current	6.2 amperes
Maximum discharge current	50 amperes

Physical Characteristics

Battery type	Sealed silver cadmium rechargeable
Number of cells	30 series connected
Charge control sensing	Individual cell voltages
Size	7 x 19.5 x 8 inches
Weight	46 pounds

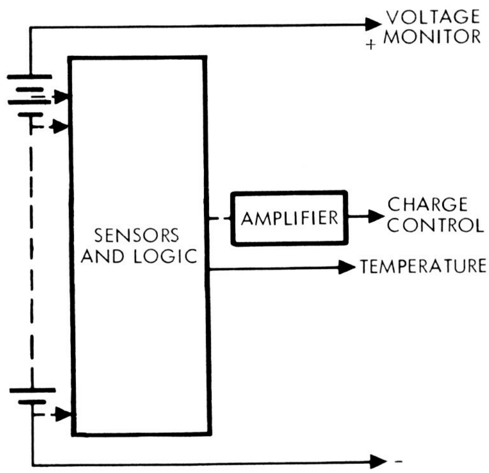


Figure 6. Preliminary Specification—Battery

c. Power Control Unit and Shunt Element Assemblies

The functions of the power control unit (PCU) are to:

- Limit main bus voltage when solar array power exceeds connected load power
- Provide proper management of available power by priority load control and power storage control
- Provide synchronization frequencies for the entire spacecraft
- Provide telemetry outputs to define the state and operation of the power subsystem.

Figure 7 summarizes the functions and characteristics of the PCU.

Voltage Limiting. Voltage limiting requirements are to:

- Limit main bus voltage to 50 ±0.5 VDC when array power capability exceeds connected load power
- Provide a main bus impedance of 0.25 ohm maximum, DC to 50 KHz
- Consume minimum power when connected load power exceeds array power.

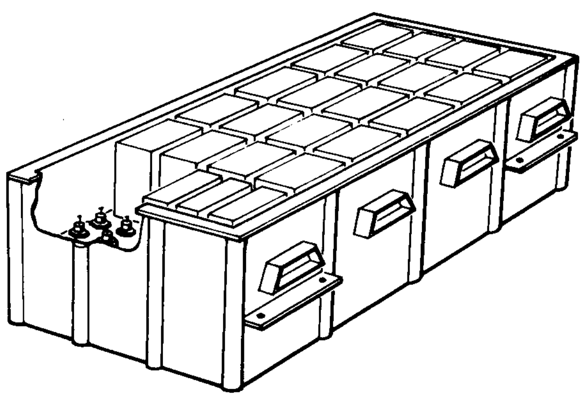
A functional block diagram of the voltage limiter is shown in Figure 8. Circuit level and part level redundancy techniques are used for maximum reliability and minimum power dissipation. Circuit redundancy, or majority voting, is used in early stages of amplifiction, where maximum gain and minimum signal loading are important and failure-mode dissipation in simple logic gates is not a significant power consideration. Part redundancy, or quad connection, is employed in final stages where failure-mode independent power dissipation is important for power conservation, and relatively low gain is acceptable.

The error amplifier is a majority voting analog circuit. The error outputs of three differential amplifiers, EA1-3, each with separate reference VRF1-3 and a common feedback, are connected in a quasi bridge amplifier. This amplifier combines voltage amplification by VA1-6, and logic gating by three AND gates. Its output voltage is independent of any

PRELIMINARY SPECIFICATION
Power Control Unit

Functions

Function 1 Limit solar array main bus voltage to 50.0 ± 0.5 vdc
Function 2A Provide boost regulator control ON signal: main bus voltage 49.6 vdc OFF signal: main bus voltage 49.8 vdc
Function 2B Provide battery charger control ON signal: main bus voltage 50.1 vdc OFF signal: main bus voltage 49.9 vdc
Function 2C Provide nonessential load control Connect capsule load when normal load is connected; batteries are not charging; and excess array capacity exceeds 200 watts. Connect heater load when capsule load is connected and excess array capacity exceeds 128 watts. Disconnect heater load when batteries discharge. Disconnect capsule load when heater load is disconnected and batteries discharge.
Function 3 Provide sync for spacecraft equipment 270 KHz 4 KHz 400 Hz
Function 4 Provide telemetry signals • Main bus voltage • Total shunt current



Subsystem Characteristics

Power dissipation	20 watts maximum
Weight	10 pounds
Size	360 cubic inches
Quantity required	1

Figure 7. Preliminary Specification — Power Control Unit

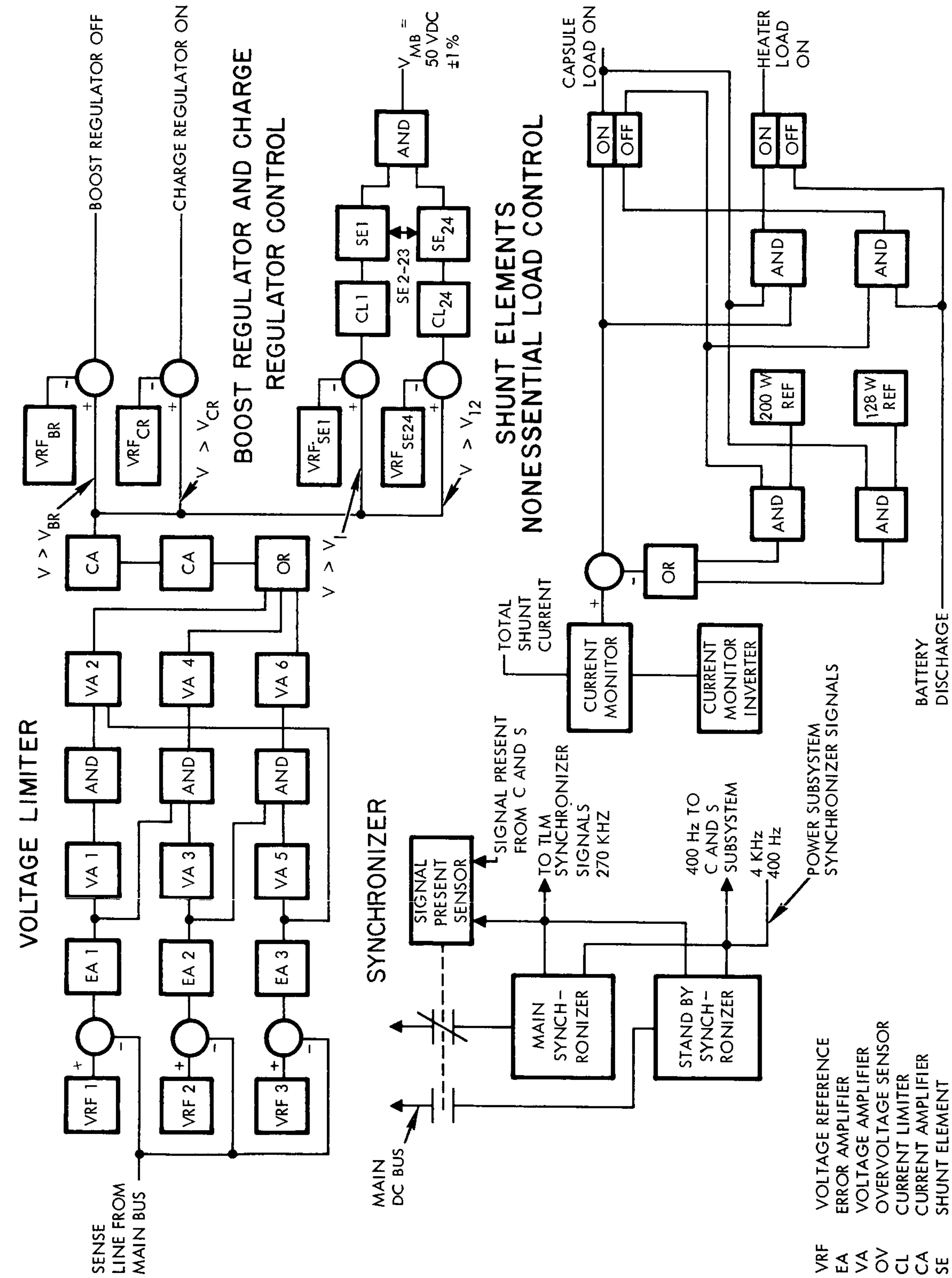


Figure 8. Power Control Unit Functional Block Diagram

single part failure, and many multiple part failures. The bridge output is then amplified by two series-connected quad current amplifiers to the drive level required by power shunt elements.

A sequential shunt regulation approach is selected so that main bus voltage is always controlled by an amplifier operating in its linear transconductance range. Additional advantages are minimization of shunt element power, and continuing operability with a failed shunt assembly. Each section of the solar array is provided with a suitable tap point and a shunt element, or transistor quad, is connected between this point and the return bus. The shunt elements are controlled to draw current from the shunted solar cell string in proportion to main bus voltage error as amplified by the current amplifiers. A preliminary specification of a shunt element subassembly is shown in Figure 9.

Each shunt element is designed to begin conducting at a higher error signal than the preceding element, corresponding to voltage references VR 1-24. These references are designed such that each successive element begins to conduct when the preceding element saturates. Consequently, a single linear element always controls main bus regulation while other elements dissipate little or no power in saturated or open states.

Base drive current to saturated elements is limited by active current limiters, designed to insert high impedance above drive requirements. In actual design practice, the drive limiting and voltage referencing functions would be combined in a single amplifier.

Load Control. Priorities assigned for distribution of power are as follows:

a) Power to essential loads and all connected science experiment loads

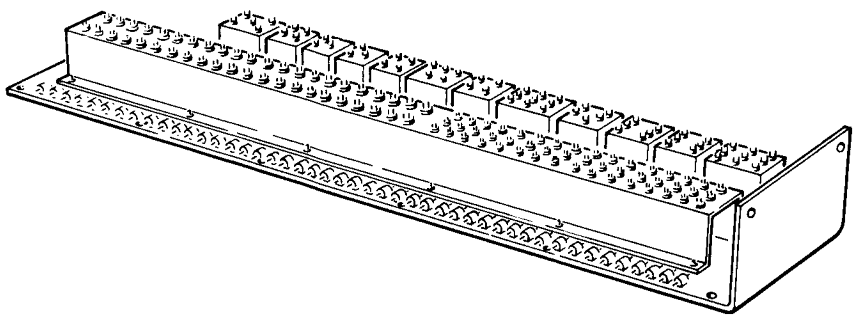
b) Charge batteries

c) Power to capsule when P >(a) + (b) + 200 watts

d) Power to reaction control system gas heaters, when P >(a) + (b) + (c) + 128 watts and RCS heaters have been commanded on.

PRELIMINARY SPECIFICATION

Shunt Element Assembly



Function

Solar array voltage limiter active control element.

Physical Characteristics

Size	6 x 3 x 25 inch
Weight	8 pounds
Volume	450 in^3

Performance Characteristics

Maximum heat dissipation.	100 watts

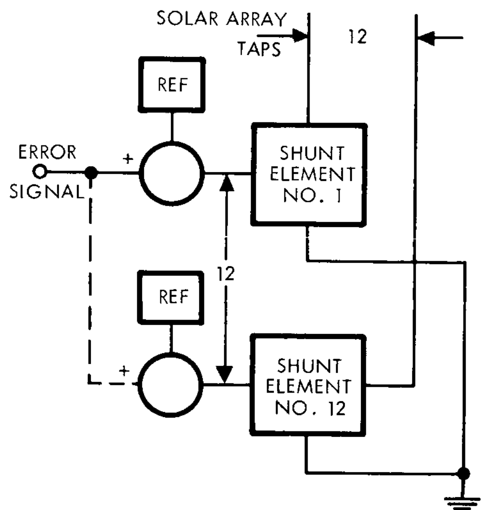


Figure 9. Preliminary Specification—Shunt Element Assembly

Detailed control requirements are to:

- Inhibit boost regulator operation when array power is equivalent to normal load power as indicated by main bus voltage of 49.8 VDC
- Enable battery charger operation when array power exceeds normal load power, as indicated by main bus voltage of 49.9 VDC
- Connect capsule loads when the battery is charged and 200 watts excess array power is available, as indicated by 4.4 ±0.2 amperes total shunt current
- Connect reaction control system heater loads when the capsule is connected and 128 watts excess array power is available, as indicated by 3.0 ±0.1 amperes total shunt current
- Disconnect gas heaters when battery discharge is initiated
- Disconnect capsule when battery discharge continues after heaters are off
- Disable charge operation when array capacity fails to exceed normal load power, as indicated by main bus voltage of 49.9 VDC
- Initiate boost regulator operation when normal load power exceeds array power, as indicated by main bus voltage equal to 49.8 VDC.

The sequence of these requirements provides continuously regulated power to the priority normal load at maximum efficiency. When array power becomes adequate to supply the normal load, as after eclipse, booster operation is inhibited to eliminate standby losses. A further increase in array power is next used to enable battery charge to store maximum energy for operation during a subsequent eclipse. After battery charge is completed, remaining excess array power is then provided to nonessential loads, first the capsule and then the gas heaters. Load control cycling is prevented by connection of nonessential loads only after adequate array power to supply them has first been measured.

Synchronization. Synchronization for the entire spacecraft is provided from a central precision oscillator (540 KHz) located in the electric power subsystem.

A 540-KHz crystal with a 0.01 per cent tolerance establishes the reference clock. An oscillator is driven by this crystal generating the 540-KHz master clock rate. This master clock rate is used directly by the C and S subsystem. Through a series of mod counters, this frequency is divided into the appropriate frequency required by the various subsystems. A mod 2 counter divides the master clock down to 270-KHz for the telemetry subsystem. These frequencies are distributed to the appropriate subsystems via the power distribution unit.

The 540-KHz master clock signal is also divided by a modulo 135 counter to drive the 4-KHz clock, and a mod 10 counter is driven by the 4-KHz clock to generate the 400-Hz clock. Both these frequencies are used by the power subsystem for synchronization of the inverters. The 400-Hz signal is also distributed to the C and S subsystem for further division to 1 Hz and 0.25 Hz. Two redundant synchronization assemblies are provided, only one of which is operative at any time. Switching from the main synchronization assembly to the backup is accomplished by detection of both the 135-KHz signal in the power subsystem and the 0.25-Hz signal in the C and S subsystem, and, in the event of disappearance of either signal, implementing the automatic switching function.

A block diagram of one complete synchronization assembly is shown in Figure 10. Switching logic is shown in Figure 11. Component redundancy is used in detectors and switching logic.

d. Battery Regulator

Charge Regulator. The battery charge regulator performs several functions. Basically, its primary function is to charge the batteries when the solar array is illuminated and has capability in excess of that required for the loads. When any three of the 30 cells in the battery reaches a voltage limit which is preadjusted to match an empirically determined voltage-temperature relationship, charging is temporarily terminated, recommencing when the controlling cell voltages drift downward through a narrow deadband. Data supporting choice of this charge control may be found in Phase IA Task A Report, Volume 5, Section IV.3.

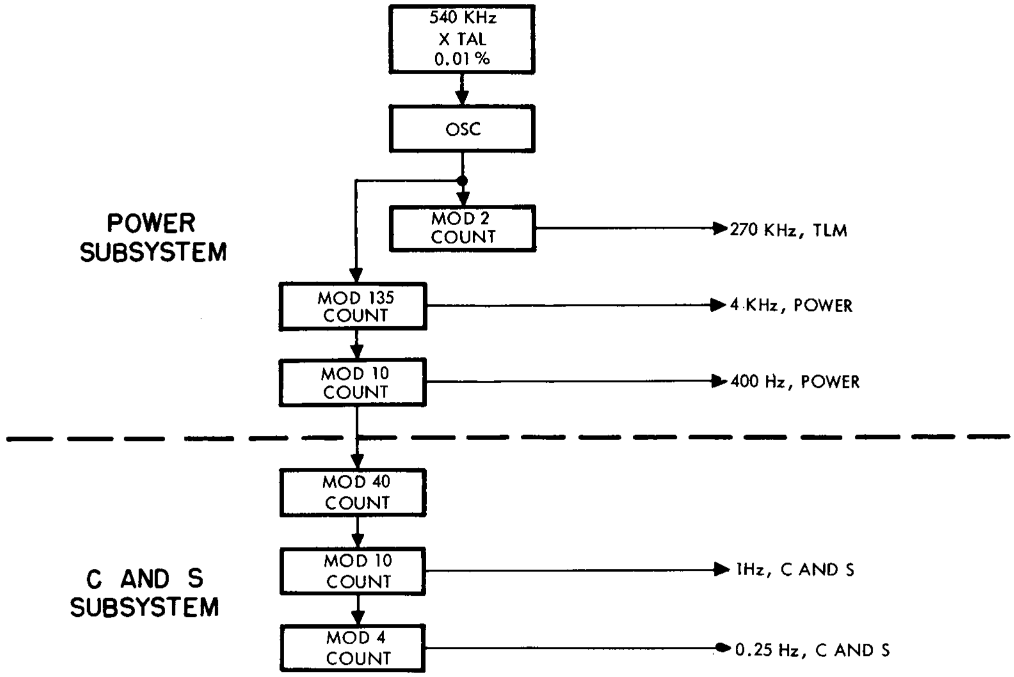


Figure 10. Synchronization Assembly Block Diagram

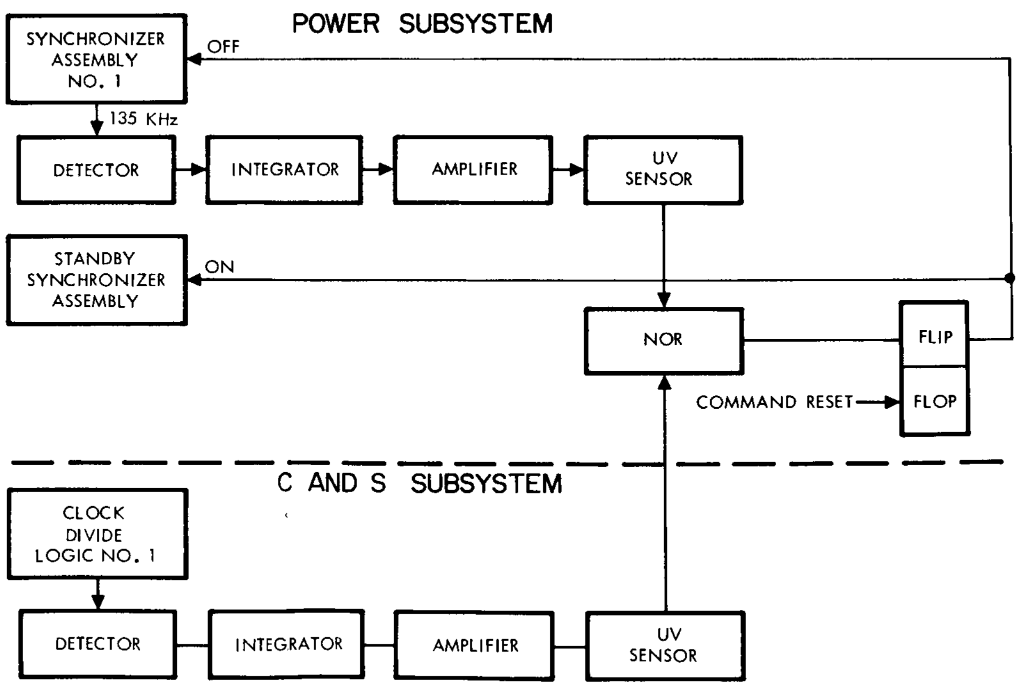


Figure 11. Switching Logic

In addition to the charge control function, the battery charge control also acts in conjunction with the solar array controls, varying the internal resistance of a series element of the charge controller in response to a signal from the array bus voltage limiter, reducing the battery charging current to that level necessary to maintain the main bus at 50 volts.

The basic regulator is limited by current feedback which limits the maximum charge rate at 2 amperes.

The basic block diagram shown in the specification sheet indicates the system composed of five basic blocks: reference, disable, amplifier, series regulator, and current limit.

The reference determines the set point against which the unit regulates. The disable circuit utilizes the basic reference signal and modifies it as required by the control signals from the temperature and voltage logic and the array bus voltage signal. The amplifier takes the basic error signal between the output and current limit and amplifies this signal to drive the series regulator.

The series regulator controls the flow of power to the battery. It essentially consists of a series dissipative regulator. Maximum dissipation occurs at minimum battery voltage and maximum charge current. Minimum charge rate is automatically increased after a battery failure such that the remaining battery(s) can be charged at a higher rate. The current limit control determines the maximum current at which the regulator can charge the batteries.

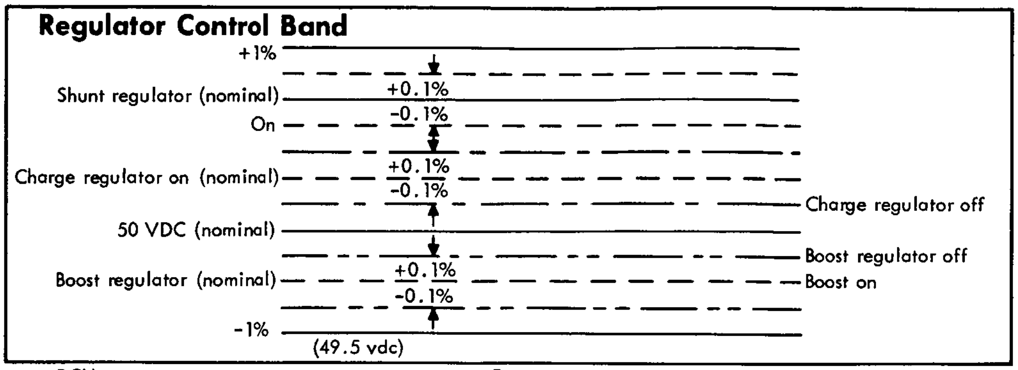
Boost Regulator. The primary function of the boost regulator is to provide power at the appropriate voltage to the load during eclipse periods or when the solar array is not providing sufficient power.

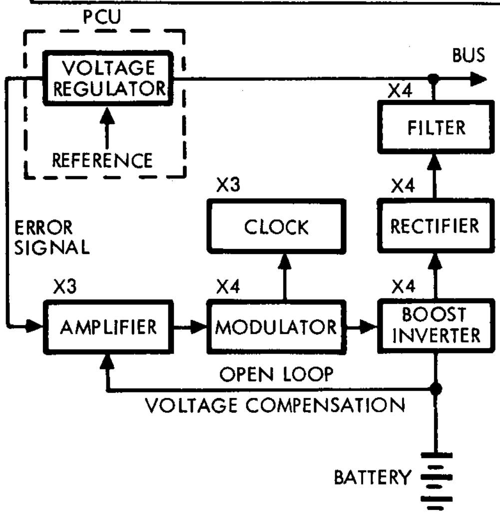
The regulator boosts the lower battery voltage to 50 volts and regulates the output bus. Boosting is performed until the array is capable of supporting the loads.

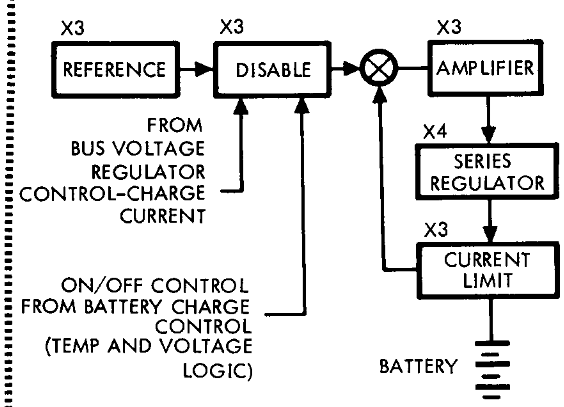
The block diagram shown in the battery controller specification indicates that the boost regulator consists basically of six blocks: amplifier, modulator, clock, boost inverter, rectifier, and filter.

PRELIMINARY SPECIFICATION

Battery Regulator







Boost Regulator

Function

Boost regulator permits discharge into main bus when solar array is in eclipse. Battery voltage boosted to 50 vdc.

Characteristics

1. Power output	400 watts
2. Power dissipation	87 watts
3. Voltage input	29-42 vdc
4. Voltage output	49.7 vdc ±.1%
5. Output impedance	(dc to 50 KHz) 0.25Ω
6. Operational frequency	4 KHz

Charge Regulator

Function

System permits full rated charge when solar array is illuminated and battery is not charged. It is limited by current limit setting.

Permits trickle when battery is fully charged. Disable circuit controls system from signals derived from bus voltage controls or battery temperature and voltage logic.

Characteristics

1. Input voltage	50 vdc + 1%
2. Output voltage	29-47 vdc
3. Input power	600 watts
4. Power dissipation maximum	42 watts
5. Standby power	0.6 watts

Figure 12. Preliminary Specification—Battery Regulator

The reference and error signal is derived from the voltage regulator contained in the PCU. The amplifier boosts the error signal to operate the modulator which determines the on/off period of the basic pulse-width-modulated boost inverter. It is controlled also by a signal which is derived from the conditions of the battery voltage.

The modulator is operated at a constant rate with a variable duty cycle and drives the boost inverter. The output of the boost inverters is rectified and filtered to produce the required output voltage and ripple content. A preliminary specification of the battery regulator is shown in Figure 12.

e. Power Inverters

Preliminary specifications for the 4.0-KHz and 400-Hz inverters are given in Figures 13 and 14.

The inverters will start and operate as specified under any combination of input voltage, load, and temperature. Control is accomplished by the application of input power to the inverter from a remote power switch in the power distribution unit. Load may be transferred in a similar manner.

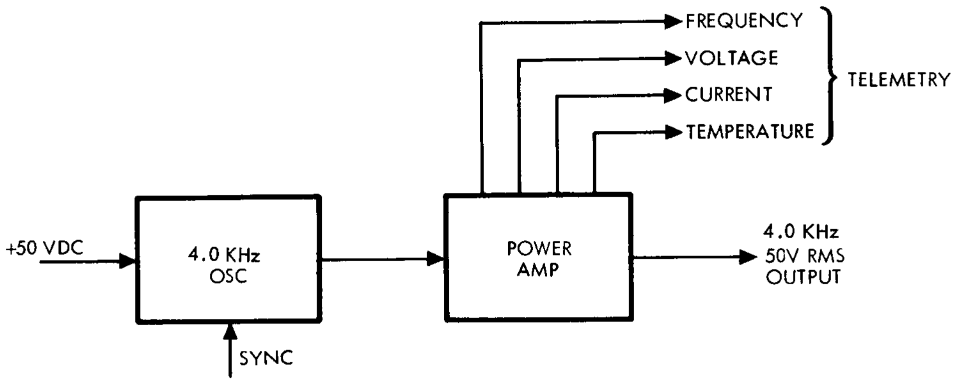
The inverters are synchronized from a low level power source, essentially square wave. Telemetry outputs monitor output voltage, output current, input current, and temperature.

The efficiency of the inverters is maintained over the load range by the use of internal feedback proportional to load current.

Functions within the inverter are packaged in individual modules. Electrical parts are arranged in "cordwood" fashion between insulated support planes. Dividing the modules into functions minimizes the number of interconnecting insulated conductors between modules, minimizing the cross-coupling of electromagnetic interference. The arrangement of modules in the inverter housing in sequence minimizes the length of these final wiring interconnections, and the concentration of parts within a module and the close proximity of wiring modes also contributes to the reduction of interference energy by cancellation of the fields external to the module. The modules are placed in an enclosed compartment for

PRELIMINARY SPECIFICATION

4.0 KHz Inverter



Physical Characteristics

Size	6 x 6 x 6 inches
Weight	7.3 pounds
Rated efficiency	90 percent at 50 vdc
Input voltage	+49.5 to +50.5 vdc

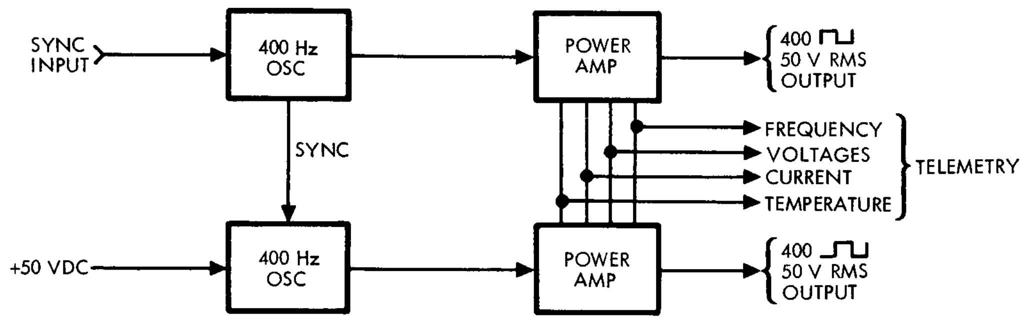
Performance Characteristics

Nominal voltage vac	50.0 rms (squarewave)
Phase	single
Percent regulation	Synchronous ±2.0 percent Nonsynchronous ±5.0 percent
Total rated power	325 watts

Figure 13. Preliminary Specification—4.0 KHz Inverter

PRELIMINARY SPECIFICATION

400 Hz Inverter



Physical Characteristics		**Performance Characteristics**	
Size:	6 x 5 x 2.5 inches	Nominal voltage vac	50.0 rms
Weight	3.0 pounds	Phase	Two
Rated efficiency	90 percent at 50 vdc	Percent regulation	Synchronous ± 2.0 percent Nonsynchronous ± 5.0 percent
Input voltage	+49.5 to +50.5 vdc	Total rated power	50 watts

Figure 14. Preliminary Specification—400 Hz Inverter

electromagnetic control. A smaller "clean" compartment containing the back of the connector and telemetry components receives the power conduction from and to the module area through feed terminals.

f. Power Distribution Unit

The function of the power distribution unit is the distribution of electrical power throughout the spacecraft. The inputs to this unit are the AC and DC power from the solar array, batteries, and inverters. The outputs include all the distributed loads from these buses, both continuous and switched. Power switching for the entire spacecraft is handled in this unit. A preliminary specification is shown in Figure 15.

All subsystems receive their power through this assembly. Some, such as radio and pyrotechnics, use it as junction box only; a second group, such as thermal and capsule, provides sensors for power switching; and a last one containing most other subsystems is switched by command of the computer and sequencer, backed up by ground command. Some redundancy switching may be in either of these categories, i.e., self sensing or commanded.

The power distribution unit contains all connectors to connect the power subsystem to the main flight harness, which carries its output to the required destinations. A switching analysis will be performed to determine the use of solid state switches versus electromechanical relays. This analysis will contain, but not be limited to, magnetic effects, environmental behavior, leakage current constraints, current capacity, reliability, and power consumption. The power distribution unit will further contain protective devices to protect the power supply from catastrophic short circuits in the loads and to protect the loads from excessive overvoltages.

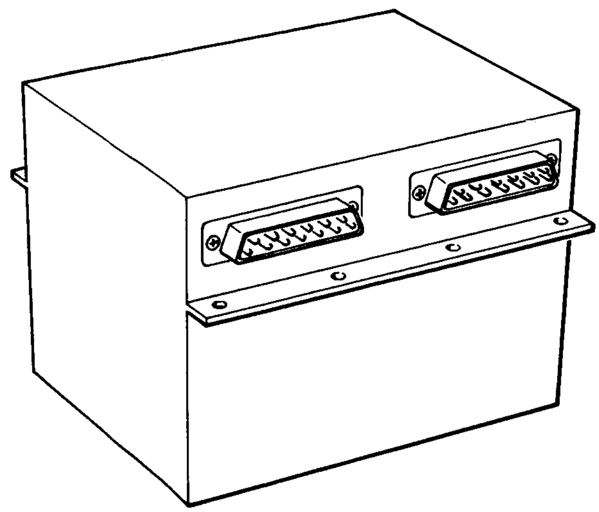
The unit, especially when electromechanical relays are used, is a potential contributor to radio frequency interference (RFI). Antiarcing contact protectors will be individually tailored to their relay and load circuits.

1.5 Subsystem Performance Parameters

The performance and physical parameters of the power equipment are summarized in Table 7.

PRELIMINARY SPECIFICATION

Power Distribution Unit



Function

Distribution and switching of electrical power in spacecraft

Operating Characteristics

Inputs

50 vdc ±1 percent from solar array and batteries
29-42 vdc unregulated from batteries
4 KHz 1ϕ 50 vac ±2 percent from inverter
400 Hz 2ϕ 50 vac ±2 percent from inverter
Commands from computing and sequencing and command subsystems

Outputs

Commanded 4 KHz to redundant PCM encoder
Commanded DC to flight capsule
Commanded DC to thermal heaters
Commanded DC to solenoid propulsion valves
Continuous AC and DC to all other S/C loads

Physical Characteristics

Size 6 x 6 x 8 inches
Weight 7.5 pounds

Figure 15. Preliminary Specification—Power Distribution Unit

Table 7. Electric Power Equipment Summary

Equipment	Efficiency*	Power Dissipation**	Weight (lb)	Volume (in^3)
4.0-KHz inverter	0.90	——	7.3	216
400-Hz inverter	0.90	——	3.0	75
Power control unit	——	20 watts max	8.0	330
Shunt element assembly	——	5 to 100 watts	8.0	450
Battery regulator			14.0	432
Charge	——	0.6 - 42 watts	——	——
Boost	0.82	——	——	——
Battery	0.60	——	46.0	1090

*Where power dissipation is proportional to output.

**Where power dissipation is not a function of power output.

1.5.1 Solar Array Performance

The solar array is sized to provide the required power throughout the mission life under "worst case" conditions, which are defined as the maximum predicted temperature ($\alpha/\epsilon = 1$) combined with the most adverse combination of design factors and mission trajectory. Because of the differences between former (Task A) design conditions and those which now apply, including the reduced estimates of radiation dose at Mars and increased temperatures due to the insulation which has been added to the back of the solar array, tradeoff studies were made for the determination of optimum base resistivity. The degradation factors used are discussed in Section 1.5.3. Predicted array temperatures are presented under Thermal Control, Section 11.4.1(b).

Single cell current-voltage (I-V) curves, which appear in Figure 16, were generated for individual solar cells of 1 and 10 ohm-centimeter base resistivity using 6-mil fused silica protective covers. Table 8 shows that 1 ohm-centimeter cells are preferred because of their higher maximum power available at 1.47 AU and greater maximum power voltage. The selection of 6-mil-thick coverglass is based upon achieving minimum array weight for a specific end-of-life power.

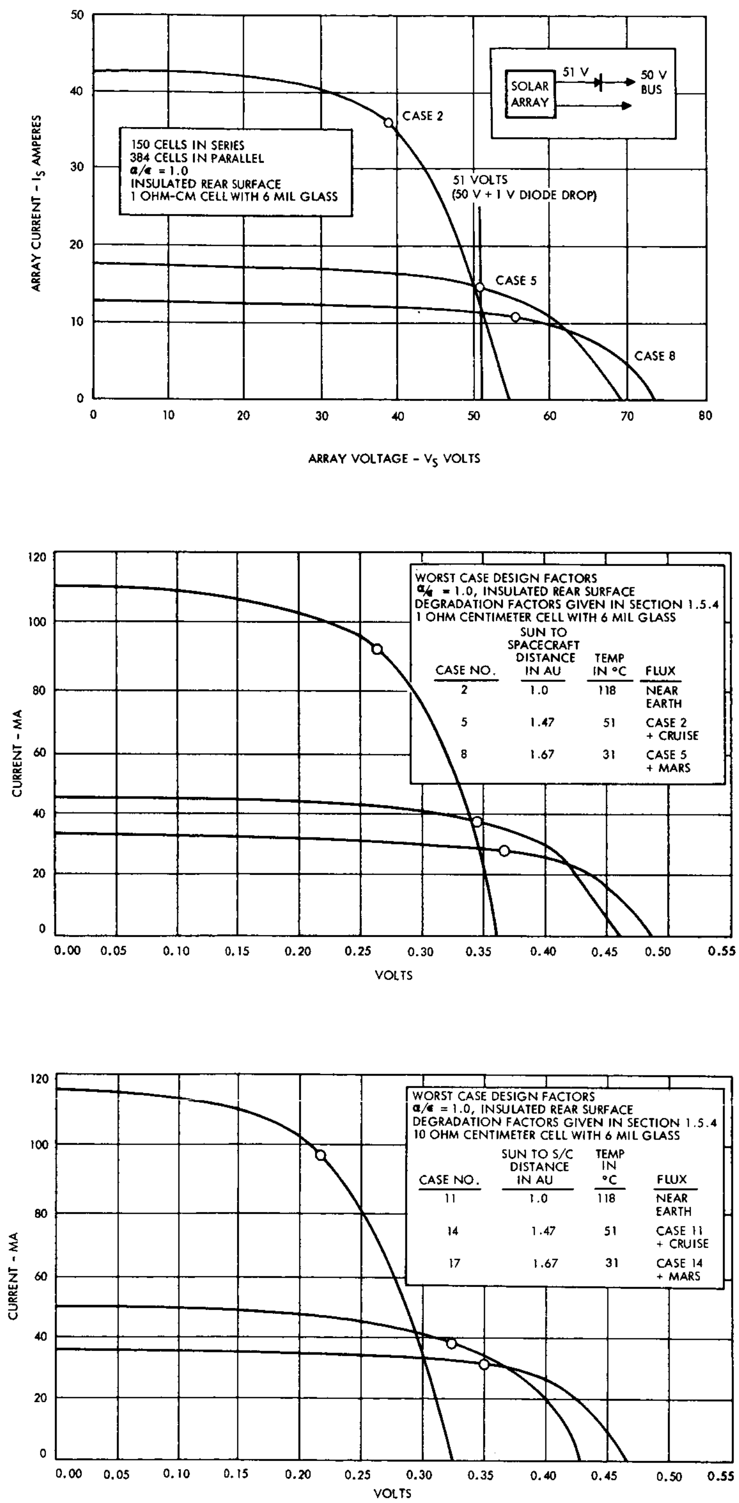


Figure 16. Worst Case Solar Array I-V Curves

Table 8. Single Cell Maximum Power and Voltage

Case		AU	1.0 ohm-cm		10 ohm-cm	
			P_{max} watts	V_{max} volts	P_{max} watts	V_{max} volts
2, 11	Worst case temperatures and design factors	1.00	0.0244	0.260	0.0207	0.220
5, 14		1.47	0.0130	0.340	0.0123	0.310
8, 17		1.67	0.0102	0.370	0.0101	0.350

The cases analyzed in the above table include the two key mission phases which are critical in determining the required array area: 1) the power required immediately prior to capsule separation at 1.47 AU, and 2) the power required at 1.67 AU at Mars. The power required is 719 watts and 539 watts, respectively, at the 50-volt bus.

The solar array will operate at a nominal bus voltage of 50 volts. Allowing 1 volt for the blocking diode voltage loss requires that N_S series solar cells deliver the required current and power at 51 volts. Thus $N_S = 51/v$ where v is the selected per-cell operating voltage.

The required number of parallel connected cells, N_P, is given by $N_P = I_R/i$ where I_R is the total required current, including power subsystem losses, at 50 volts, and i is the per-cell current at v volts. The values of v and i are determined from the worst case IV curves, cases 5 and 8, as indicated in Table 7.

A value of v must be selected so that the total current supplied (N_P x i) is equal to or greater than required (I_R) at 1.47 and 1.67 AU.

Using the given power requirements, I_R is easily determined and is given in Table 9.

Table 9. Array Current Requirements

AU	Power Required, (watts)	Voltage, V (volts)	Current Required, I_R (amperes)
1.0	577	50	11.5
1.47	719	50	14.4
1.67	539	50	10.8

A value of v = 0.340 volt is selected, this is the voltage at the maximum power point at 1.47 AU, which is the point at which power margin is lowest.

Table 10. Array Current Output at 0.34 volt per Cell

AU	(volts)	I (amperes)	p (watts)	N_S No. of cells	N_P No. of rows	I_S (amperes)	I_R (amperes)
1.0	0.340	0.0340	0.0116	150	378	12.8	11.5
1.47	0.340	0.0381	0.0130	150	378	14.4	14.4
1.67	0.340	0.0291	0.0989	150	378	11.0	10.8

where p is the power per solar cell at 0.340 volt.

Table 10 shows that the minimum number of cells in parallel to satisfy the required array output current is 378.

The results of the solar panel module layouts (Figure 5) provide a practical arrangement of 150 series cells and 384 parallel cells. Thus, six excess parallel cells are available to provide a power margin.

In terms of power, the results of the array analysis are summarized in Table 11. The worst case solar array I-V curves are shown in Figure 16.

Table 11. Array Power Margin Summary (Worst Case Design and Trajectory)

Case	AU	P_R	P_s at 50V	Percent Excess Power
2	1.0	577	652	13.0
5	1.47	719	732	1.8
8	1.67	539	559	3.7

It should be noted, however, that the power margins so calculated are for worst case design considerations (i.e., $\alpha/\epsilon = 1.0$, worst case degradation factors), combined with worst case trajectories and full load conditions.

Performance of the array at nominal rather than worst case design conditions increases available power margins, as shown in Table 12.

Table 12. Array Power Margins—Nominal Design, Worst Case Trajectory

AU	P_R	P_s at 50V	Percent Excess Power
1.0	577	1010	75.0
1.47	719	808	12.4
1.67	539	600	11.3

In addition, use of nominal rather than worst case trajectories and programming of experiment loads will engender further improvement in power margins.

1.5.2 Battery Performance

The data in Table 13 delineates expected battery performance during normal and peak load conditions in the normal mode and in a failed condition (single battery failure) and are based on actual test data.

Table 13. Predicted Battery Performance

Characteristics	Normal Operation	One of 3 Batteries Failed
1. Open circuit voltage	42-43.2	42-43.2
2. Average loads*	32.5	32.0
3. Peak load*	32.0	31.6
4. Transient load	29.8	28.5

*Science off during failure mode (single battery failure) operation.

1.5.3 Power Margins

Figure 17 illustrates solar array power margins during the various mission phases. Solar array power output is shown for nominal and worst case design conditions. Estimated average power demands from Table 1 for the various mission phases are also plotted, including periods of battery discharge. The basic power requirements include estimated system losses in power conditioning, regulation, and control equipment. In addition to the basic demand, increased power requirements for the reaction control system gas heaters (nonessential load) are also shown. A minimum power margin of 3.7 percent is available at the end of life (1.67 AU) with all essential loads connected.

Figure 18 illustrates the margins in stored energy during periods when the solar array is not illuminated. In the event of a single battery failure, the two remaining batteries would still be capable of supplying all essential loads (except science) during all maneuvers and maximum eclipse operation. If two batteries fail in orbit at Mars, the remaining battery will support the spacecraft for eclipses up to about 1.4 hours in duration.

1.5.4 Solar Array Degradation Factors

The design factors for the Voyager solar cell array are defined in Table 14.

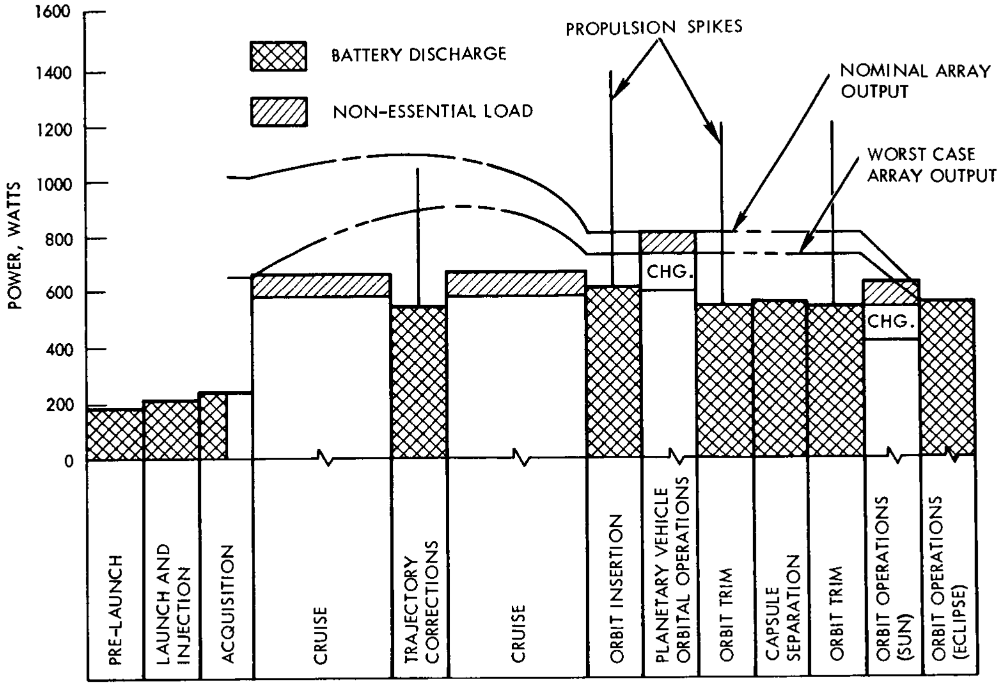


Figure 17. Power Margins as a Function of Mission Phase

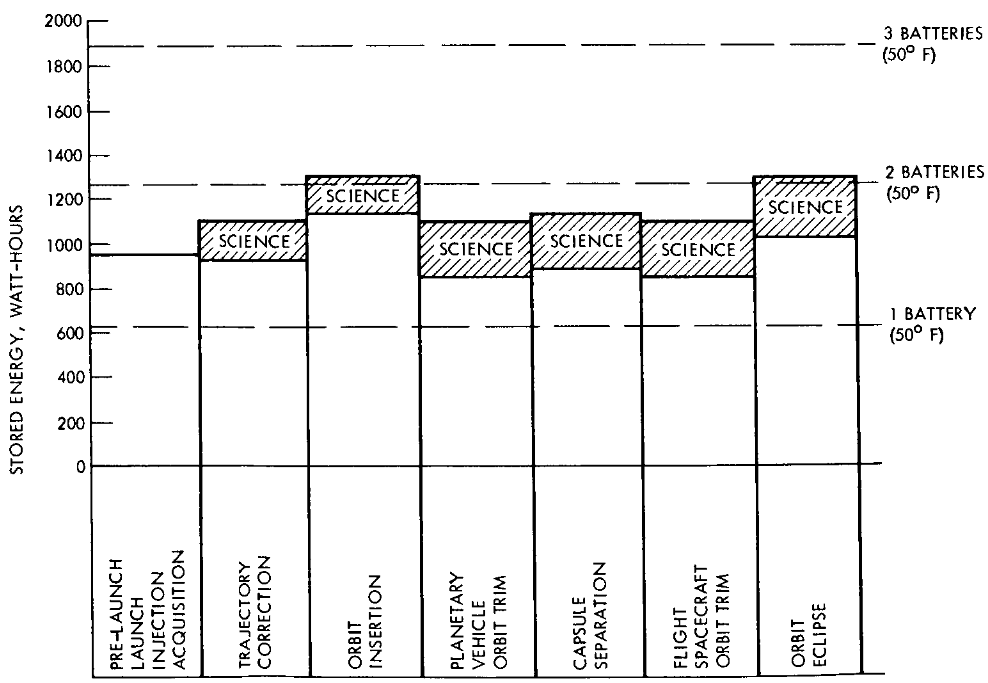


Figure 18. Stored Energy Margins as a Function of Mission Phase

Table 14. Definition of the Design Factors for the Voyager Solar Cell

Factors Current	Performance Degradation (per cent)
Angle of incidence - ($\pm 1^{o}$)	-
Ultraviolet and particle irradiation	-5
Current measurement tolerance	±2
Efficiency prediction uncertainty	±3
Solar constant variation	-
Micrometerorite and sputtering	-5
Particle radiation damage to solar cell (Table 15)	-
Voltage	
Thermal cycling degradation	-2
Wiring loss	-2
Voltage measurement tolerance	±2
Particle radiation damage to solar cell (Table 15)	

The charged particle radiation damage to the glass-covered solar cells has been calculated by techniques identical to those used in Task A. The most significant fact is that the Martian magnetic field is assumed as having an intensity no greater than 10^{-5} times the intensity of the geomagnetically trapped radiation. The resulting Martian electron and protron flux is insignificant with regard to solar cell degradation; hence solar flare particles are the only contributors to the decrease in maximum power in the 6-month Martian orbit. These fluxes have been converted into 1-Mev damage equivalent electrons per square centimeter. Table

15 shows the 1-Mev flux and degradation factors for 1.0 and 10 ohm-cm base resistivity cells at 1.0, 1.47, and 1.67 AU, with 6-mil fused silica covers.

Table 15. Radiation Flux and Solar Cell Damage

Nominal Base Resistivity ohm-cm	AU Value	Equivalent, 1-Mev Electron Flux Particles/cm^2	I_{sc}	I_{mp}	V_{mp}	V_{oc}	P_{max}
1 and 10	1.0	0.0	1.0	1.0	1.0	1.0	1.0
1	1.47	1.6×10^{14}	0.91	0.91	0.957	0.943	0.871
1	1.67	2.62×10^{14}	0.887	0.887	0.946	0.930	0.839
10	1.47	1.6×10^{14}	0.910	0.910	0.944	0.943	0.859
10	1.67	2.62×10^{14}	0.889	0.889	0.930	0.930	0.827

1.5.5 Reliability Estimate

A reliability block diagram of the power subsystem is shown in Figure 19, including the estimated unit and subsystem reliabilities in the time period from liftoff to the completion of one month in orbit at Mars.

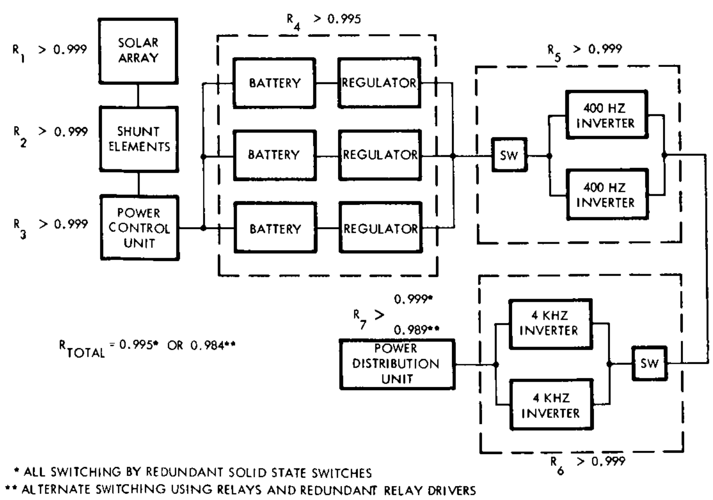


Figure 19. Reliability Block Diagram

1.6 Changes From Task A Design

As a result of revised requirements, the selected solar array design for Task B is different from that for Task A in the following major respects:

- Area: Increased from 190 to 284 ft^2 to meet increased power requirements.
- Insulation: Thermal insulation added to rear face of solar array to minimize heat transfer between array and spacecraft.
- Temperature: Higher array temperatures throughout the mission due to array insulation increases eclipse survival capability of array. Minimum temperature after 2.3 hour eclipse at Mars increased from -165°C to -133°C.
- Cell Base Resistivity: Changed from 10 ohm-cm to 1.0 ohm-cm as a result of new design conditions of temperature and radiation damage.
- Configuration: Changes in number, size, and shape of solar panels. Change from 10-cell modules to eight cell modules to facilitate panel layout.

In addition to the solar array design changes just listed, other changes in the power subsystem design have been incorporated as follows:

- Increase in the number of batteries and battery regulators from two to three. Two alternatives were possible. If two larger size batteries were used, reliability requirements call for a single battery to be capable of the total mission. By use of 2-of-3 redundancy in batteries and battery regulators, a considerable savings in weight is possible, since the installed capacity need be only 3/2 of that required rather than double. In addition, use of two battery-regulator pairs would require that the specified 400-watt limit of power handling capability be violated in the event of one battery or regulator failure.
- Increase in the number of shunt elements to 24, in two packages of 12 each. This is based upon the array layout of eight panels with six strings of cells on each panel. The possible number of shunt elements is therefore 8, 16, 24, or 48. Use of eight or 16 elements could lead to dissipation of more than 100 watts in a single control element under worst case conditions, whereas the dissipation in each of the two packages of 12 elements may be maintained below the specified limit of 100 watts.

- The precision oscillator and frequency standard, plus required frequency divider networks, have been added to the power subsystem, in conformance with JPL requirements.
- The power distribution unit has been added to the power subsystem, in conformance with JPL specifications.
- The 820-HZ inverters have been deleted from the power subsystem, and are now a part of the gyro reference assembly.

2. GUIDANCE AND CONTROL SUBSYSTEM

2.1 General Description

The guidance and control subsystem (G&CS) provides three-axis attitude control of the planetary vehicle and flight spacecraft at all times after separation from the launch vehicle. It also controls the orientation of the high-gain and medium-gain antennas, based on pointing commands from the spacecraft sequencer, and provides signals indicating the occurrence of limb and terminator crossings for use in sequencing of science instruments. It also measures vehicle acceleration during propulsion operations.

During interplanetary cruise the spacecraft pitch and yaw axes are stabilized with respect to the sun, thereby maintaining the solar array surface normal to sun incidence. Roll stabilization is provided using Canopus as a reference.

Upon ground command or commands stored in the computing and sequencing subsystem (C and S), the guidance and control subsystem will reorient the spacecraft to attitudes required for velocity adjustments or capsule separation. Attitude reference for these operations is provided by three single-degree-of-freedom gyros which are capable of being operated in a rate or a rate-integrating mode. After each maneuver the cruise orientation is re-established.

During orbital operations the guidance and control subsystem maintains three-axis stabilization relative to the sun-Canopus system and provides signals indicating Mars limb and terminator crossings. Should the sun or Canopus be occulted, the guidance and control subsystem will maintain the spacecraft's last attitude until the celestial references are again acquired.

Thrust vector control during engine firing is provided by gimballing the engine and controlling engine position about the pitch and yaw axes using electrical actuators. Control about the roll axis is provided by the high thrust pneumatics.

The high-gain antenna and medium-gain antenna are positioned relative to the spacecraft axes such that they point towards the earth at all times except during attitude maneuvers. The high-gain antenna has freedom about two axes, hinge and shaft, while the medium-gain antenna is controllable only about its hinge axis.

A simplified block diagram showing the major units and electrical interfaces of the subsystem is shown in Figure 20. Redundancies for reliability are not shown. Table 16 lists the units or assemblies in the guidance and control subsystem, and Figure 21 shows the general arrangement of the components on the spacecraft.

Table 16. Guidance and Control Equipment List

No. per Spacecraft	Unit
1	Gyro Reference Assembly
1	Accelerometer
1	Guidance and Control Electronics Assembly
2	Canopus Sensor (redundant units)
1	Sun Sensor (combines coarse and fine sensors)
1	Earth Detector
2	Limb and Terminator Crossing Detector
1	Reaction Control Assembly (a redundant system)
1	High-Gain Antenna Drive
1	Medium-Gain Antenna Drive
1	Antenna Drive Electronics
2	Thrust Vector Control Actuators

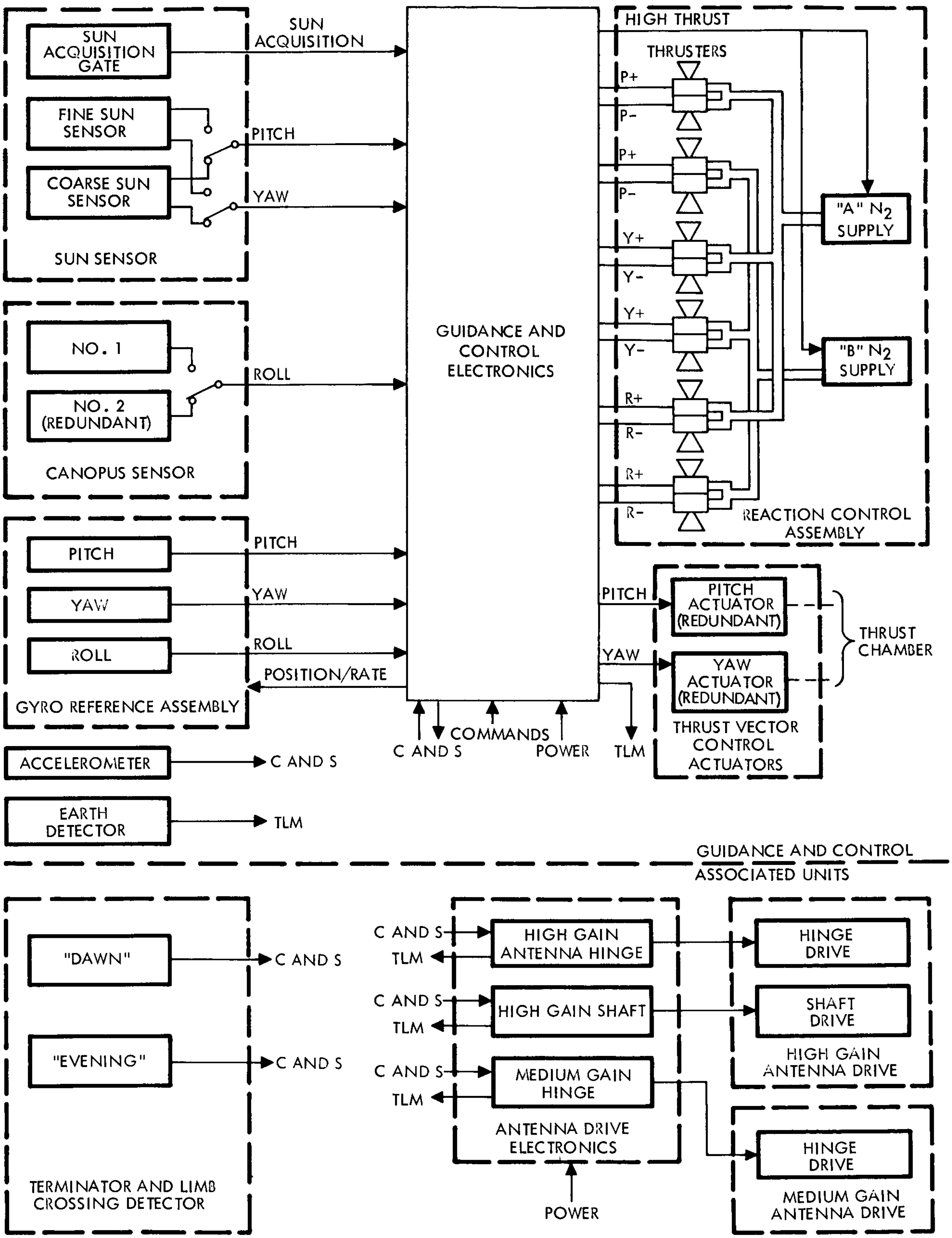


Figure 20. Block Diagram — Voyager Guidance & Control Subsystem

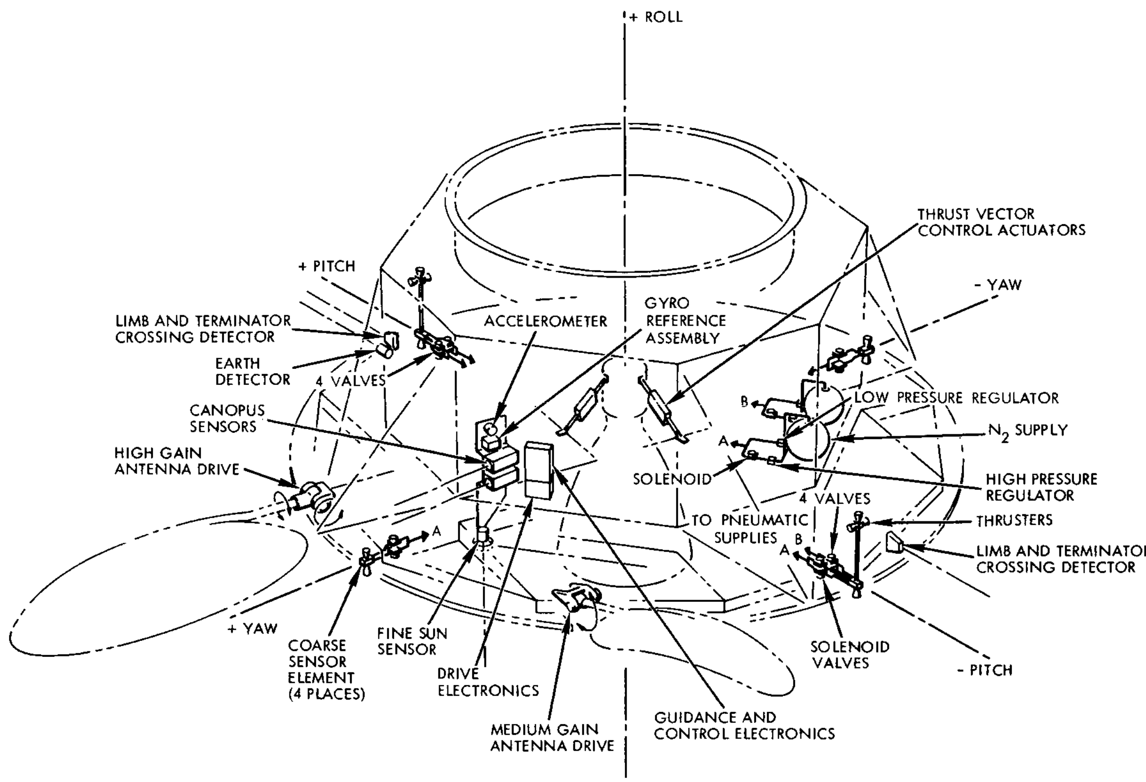


Figure 21. Guidance & Control Subsystem General Arrangement

The spacecraft is maintained in a steady orientation or is rotated at prescribed angular rates by controlled torques. During cruise, these torques are obtained by the expulsion of heated nitrogen gas through nozzles located so as to produce couples about the principal control axes. During maneuvers or attitude hold the gas is unheated. Two levels of thrust are employed, depending on the mode of operation of the spacecraft.

2.2 Requirements and Constraints

The requirements which define the pointing accuracy of the guidance and control subsystem are those of antenna pointing, pointing for velocity corrections, and attitude control during photographic operations. The requirements imposed by solar array illumination and thermal control are within the accuracies imposed by these elements.

2.2.1 Launch and Injection

To insure communications capability during the eclipse following launch, the guidance and control subsystem will maintain the attitude of the planetary vehicle within 10 degrees of its initial separation attitude in the presence of an initial rate as high as 1.5°/sec.

2.2.2 Celestial Reference Acquisition

The guidance and control subsystem will automatically acquire the sun and Canopus from any initial attitude. The subsystem will provide a controlled roll rate to permit calibration of the magnetometer and mapping of the star field prior to Canopus acquisition. The capability for inhibiting or overriding Canopus acquisition is included. Information indicating the roll rate and roll position are telemetered during the roll search. An earth sensor which senses sunlight reflected from earth provides an indication of proper Canopus acquisition.

2.2.3 Interplanetary Cruise and Orbital Operations

During interplanetary cruise and orbital operations, the guidance and control subsystem maintains orientation relative to the sun and Canopus such that the antenna pointing, Mars imaging, and velocity correction attitudes can be referenced to spacecraft axes.

The requirement for antenna pointing is based on no more than 1 db loss in gain, calling for an over-all pointing accuracy (including antenna servo errors) of ±0.7 degree, 3 σ, each axis.

2.2.4 Spacecraft Velocity Changes

The guidance and control subsystem is required to control attitude for velocity changes during trajectory corrections, orbit trim, and orbit insertion and to provide acceleration measurements. Accuracies related to these requirements are:

- For interplanetary trajectory corrections, the velocity increment must be directed to a desired attitude with respect to the sun-Canopus reference system to within ±0.76°, 3 σ, each axis for a period of up to one hour after maneuver initiation. For other velocity corrections the required attitude accuracy is 1.25 degrees, 3 σ, each axis.

- A signal for incremental velocity magnitude control purposes will be provided such that the proportional error is less than 0.1%, 3 σ, and the minimum detectable change in velocity is equal to or less than 0.15 ft/sec.

2.2.5 Flight Spacecraft-Flight Capsule Separation

For flight capsule separation, the guidance and control subsystem will point the spacecraft roll axis to any desired attitude with respect to the sun-Canopus reference system within ± 0.75°, 3 σ, each axis, for a period up to one hour after maneuver initiation.

2.2.6 Orbital Operations

During Mars photographic experiments, the guidance and control subsystem will maintain the spacecraft's orientation to within ±0.25°, 3σ, each axis. The angular rates about any axis will not exceed 10°/hour. Indications of limb and terminator crossings accurate to ±1.4 degrees will also be provided.

2.2.7 Attitude Hold

Upon command or loss of sun or Canopus, the guidance and control subsystem will maintain the existing attitude of the spacecraft to within ±2° (3σ) each axis for a period of up to three hours.

2.2.8 Other Requirements

The guidance and control subsystem will provide inertial control of the roll attitude when the spacecraft is sun-oriented with the capability of incremental adjustment of roll position in ±2° steps by ground command. Provisions will also be made to turn off and restore normal roll control by ground command. Geared actuators will be able to withstand stalled conditions at the output shaft without internal damage and will be sealed and pressurized with inert gas. A passive control stabilization, such as derived rate, will be used in the interplanetary cruise phase. Gyros will be used in a rate mode for rate reduction and control system compensation in the acquisition mode. Passive compensation may be used as a backup. The system will provide the capability of sun acquisition from any attitude even in the presence of a non-operative gyro.

2.3 Functional Interfaces

Figure 22 shows the relationships of the guidance and control subsystem and other subsystems of the Voyager spacecraft. Details of the interfaces are described in the ensuing paragraphs.

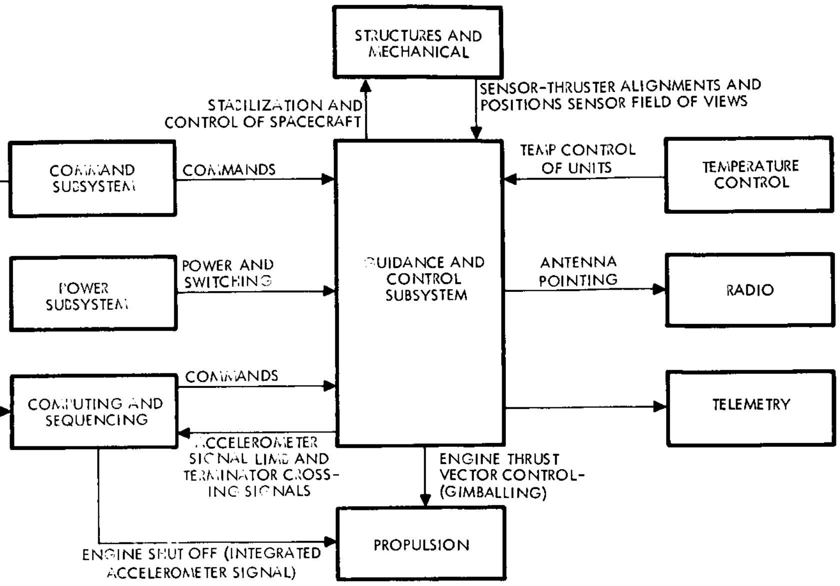


Figure 22. Guidance & Control Subsystem Interface Diagram

2.3.1 Electrical Interfaces

a. Power

Electrical power will be supplied in the following forms to the guidance and control subsystem:

50 VDC ±1%

50 VAC ±2% rms, square wave, 4 kc, 1 ϕ

50 VAC ±2% rms, square wave, 400 cps, 2 ϕ

29 - 42 VDC battery bus

In addition, switching of power for the heated thrusters of the reaction control subsystem (used as available), and for the DC motors of the thrust vector control actuators will be performed by the electrical power subsystem.

b. Radio

The guidance and control subsystem includes the drives and electronics required to position the high-gain antenna about two axes and the medium-gain antenna about its hinge axis. In addition, the drives house rotary coax joints which require accurate alignment and concentricity to prevent changes in insertion loss and VSWR as the antenna is rotated.

c. Engineering Measurements

Telemetered functions are listed in Table 17 along with details of the accuracy and sampling rates. A priority has been assigned to the signals based on the following criteria:

- Type 1 data concerning flight operations. These measurements will allow ground operators to make alternate operating choices if abnormal operation is noted.
- Type 2 data concerning system and subsystem performance. These measurements allow the ground operators to make alternate operation choice.
- Type 3 measurements indicating the effect of the space environment on spacecraft performance.
- Type 4 data to evaluate critical components, in particular those newly developed for Voyager.

A special telemetry mode will be provided for thrust vector control test prior to maneuvers. The guidance and control subsystem will receive a command to position the engine and the response and accuracy of the positioning will be verified to ascertain proper operation. If the performance is inadequate, redundant elements will be used for control.

d. Command Subsystem

Table 18 provides a list of commands to the guidance and control subsystem.

e. Computing and Sequencing Subsystem

The computing and sequencing subsystem will provide signals for control and timing and digital words for quantitative inputs to the G&CS. A list of these commands is presented in Table 19.

2.3.2 Mechanical Interfaces

a. Structural and Mechanical

Since the guidance and control subsystem is dispersed througho' . the spacecraft and is affected by the placement and stability of its

Table 17. Guidance and Co

Item	Measurement Type	Measurement Description and Location	Range		
			Max.	Min.	
1	Analog	Pitch gyro output - rate - position	±3°/sec ±15°	0 0	0. 0.
2	Analog	Roll gyro output - rate - position	±3°/sec ±15°	0 0	0. 0.
3	Analog	Yaw gyro output - rate - position	±3°/sec ±15°	0 0	0. 0.
4	Analog	Pitch control signal		0	0
5	Analog	Roll control signal		0	0
6	Analog	Yaw control signal		0	0
7	Discretes (2)	Pitch switching amplifier output	On	Off	-
8	Discretes (2)	Roll switching amplifier output	On	Off	-
9	Discretes (2)	Yaw switching amplifier output	On	Off	-
10	Analog	Roll integrator			
11	Discretes (4)	Mode control monitor	On	Off	-
12	Discrete	Gyro torquer current generator output	On	Off	-
13	Analog	Sun sensor pitch output - fine - coarse	±10° ±30°	0 0	0 0
14	Analog	Sun sensor yaw output - fine - coarse	±10° ±30°	0 0	0 0
15	Discrete	Sun sensor acquisition gate	On	Off	-
16	Analog	Canopus sensor roll error output	±2°	0	0
17	Analog	Canopus sensor star intensity output			
18	Discrete	Canopus sensor sun present signal	On	Off	-
19	Discrete	Canopus sensor star present signal	On	Off	-
20	Digital	Canopus sensor tracking position	±15°	0	-

ntrol Subsystem Telemetry List

	Frequency Resp. or Samp. Rate		Accuracy % Fu	
Nom.	Prefer'd	Min. Accept.	Pref	Min Accept.
2°/sec 25°	1.2 sec		2	5
2°/sec 25°	1.2 sec		2	5
2°/sec 25°	1.2 sec		2	5
	24 sec		2	5
	24 sec		2	5
	24 sec		2	5
	1.2 sec		-	-
	1.2 sec		-	-
	1.2 sec		-	-
	24 sec		2	5
	24 sec		-	-
	24 sec	4 min	-	-
	1.2 sec		2	5
	1.2 sec		2	5
	24 sec	4 min	-	-
	1.2 sec		2	5
	1.2 sec		2	5
	24 sec	4 min	-	-
	24 sec		-	-
	24 sec		-	-

2

...l Scale Priority	Reason for Priority
1	Switch modes to avoid excessive gas usage or loss of solar power
1	Same as Item 1
1	Same as Item 1
2	Detects subsystem malfunction
2	Same as Item 4
2	Same as Item 4
2	Same as Item 4
2	Same as Item 4
2	Same as Item 4
2	Same as Item 4
1	Same as Item 1
2	Verifies generator is on
1	Same as Item 1
1	Same as Item 1
2	Verifies sun in fine sensor
1	Same as Item 1
1	Same as Item 1
2	Verifies shutter is closed
2	Verifies star in sensor field-of-view
2	Verifies cone angle

3

Table 17. Guidance and Control

Item	Measurement Type	Measurement Description and Location	Range		
			Max.	Min.	
21	Discrete	Near earth sensor output	On	Off	-
22	Digital	Accelerometer output	7000 ft/sec	0	-
23	Analog	Pitch TVC actuator position	±6°	0	-
24	Analog	Yaw TVC actuator position	±6°	0	-
25	Analog	Pitch TVC actuator control signal	±6°	0	-
26	Analog	Yaw TVC actuator control signal	±6°	0	-
27	Analog	System A gas bottle pressure	3500psi	0	3
28	Analog	System B gas bottle pressure	3500psi	0	3
29	Analog	System A regulated gas pressure-high	650psia	0	4 p
30	Analog	System B regulated gas pressure-high	650psia	0	4 p
31	Discretes (12)	Control jet actuation	On	Off	-
32	Discrete	Pitch gyro spin motor rotation detector	On	Off	-
33	Discrete	Roll gyro spin motor rotation detector	On	Off	-
34	Discrete	Yaw gyro spin motor rotation detector	On	Off	-
35	Analog	820 cps inverter output	28 V	20 V	2
36	Analog	DC voltage 1			
37	Analog	DC voltage 2			
38	Analog	DC voltage 3			
39	Digital (12bits)	High-gain antenna hinge axis position	+140° - 90°	0	±
40	Digital (12bits)	High-gain antenna shaft axis position	±180°	0	±
41	Digital (12bits)	Medium-gain antenna hinge axis position	+140° - 90°	0	±

Subsystem Telemetry List (Continued)

)	Frequency Resp. or Samp. Rate		Accuracy % Full Scale			
Nom.	Prefer'd	Min. Accept.	Pref	Min. Accept.	Pri-ority	Reason for Priority
	24 sec	4 min	-	-	2	Verifies Canopus acquisition
	1.2 sec	24 sec	-	-	2	Verifies spacecraft velocity
	24 sec		2	5	2	Same as Item 4
	24 sec		2	5	2	Same as Item 4
	24 sec		-	-	2	Same as Item 4
	24 sec		-	-	2	Same as Item 4
)00psi	24 sec		2	5	1	Same as Item 1
)00psi	24 sec		2	5	1	Same as Item 1
)/630 sia	4 min	8 min	2	5	2	
)/630 sia	4 min	8 min	2	5	2	Same as Item 4
	1.2 sec	24 sec	-	-	2	Same as Item 4
	4 min	8 min	-	-	2	Verifies gyro is on
	4 min	8 min	-	-	2	Same as Item 35
	4 min	8 min	-	-	2	Same as Item 35
ŀ V	4 min	8 min	2	5	2	Same as Item 4
	4 min		.5	1	2	Same as Item 4
	4 min		.5	1	2	Same as Item 4
	4 min		.5	1	2	Same as Item 4
)0°	4 min		-	-	2	Verifies antenna pointing
)0°	4 min		-	-		Same as Item 43
)0°	4 min		-	-		Same as Item 43

Table 17. Guidance and Control S

Item	Measurement Type	Measurement Description and Location	Range		
			Max.	Min.	
42	Discrete	High-gain hinge actuator power	On	Off	-
43	Discrete	High-gain shaft actuator power	On	Off	-
44	Discrete	Medium-gain hinge actuator power	On	Off	-
45	Discrete	Limb and terminator crossing sensor outputs (4)	On	Off	-
46	Analog	Fine sun sensor temperature	150°F	0°F	1
47	Analog	Coarse sun sensor temperature	150°F	0°F	1
48	Analog	Canopus sensor temperature			
49	Analog	Pitch gyro temperature	150°F	50°F	1
50	Analog	Roll gyro temperature	150°F	50°F	1
51	Analog	Yaw gyro temperature	150°F	50°F	1
52	Analog	Accelerometer temperature	150°F	0°F	1
53	Analog	Pitch TVC actuator temperature	275°F	-65°F	-
54	Analog	Yaw TVC actuator temperature	274°F	-65°F	-
55	Analog	System A gas bottle temperature	100°F	0°F	5
56	Analog	System B gas bottle temperature	100°F	0°F	5
57	Analog	High-gain antenna actuator temperature	100°F	20°F	5
58	Analog	Medium gain antenna actuator temperature	100°F	20°F	5

bsystem Telemetry List (Continued)

	Frequency Resp. or Samp. Rate		Accuracy % F	
Nom.	Prefer'd	Min. Accept.	Pref	Min. Accept.
	4 min		-	-
	4 min		-	-
	4 min		-	-
	24 sec	4 min	-	-
00 °F	8 min		2	5
00 °F	8 min		2	5
	8 min		2	5
40 °F	4 min		2	5
40 °F	4 min		2	5
40 °F	4 min		2	5
00 °F	4 min	8 min		
20 °F	4 min		2	5
20 °F	4 min		2	5
0 °F	4 min	8 min	2	5
0 °F	4 min	8 min	2	5
0 °F	8 min		2	5
0 °F	8 min		2	5

2

...ll Scale	
Pri-ority	Reason for Priority
2	Same as Item 4
2	Same as Item 4
2	Same as Item 4
2	Verifies Mars limb & terminator crossings
4	Measures effect of space environment
4	Same as Item 51
4	Same as Item 51
4	Same as Item 51
4	Same as Item 51
4	Same as Item 51
	Same as Item 51
4	Same as Item 51
4	Same as Item 51
4	Same as Item 51
4	Same as Item 51
4	Same as Item 51
4	Same as Item 51

3

Table 18. Guidance and Control Subsystem Command List

	No. of Discrete Commands	No. of Direct Serial Commands	Remarks
Inertial hold mode (all axes inertial)	2		
Enable sun acquisition	1		
Canopus sensor on/off	2		
Canopus sensor select	2		Redundant unit selection
TVC actuator select	2		Redundant unit selection
Gyro power	2		
Enable accelerometer	2		
TVC gain select	2		Low gain with high propulsion thrust, high gain with low propulsion thrust
TVC on - off	2		
TVC test	1		
Fine and coarse control select (deadband)	2		0.5° and 0.25° deadband
High and low thrust reaction control select	2		
Roll spin - Canopus search	2		
Star acquisition gate override	2		
Roll incremental maneuver	2		2° steps
Roll axis inhibit	2		
High-gain antenna - hinge axis		1	12 bits
High-gain antenna - shaft axis		1	12 bits
Medium-gain antenna - hinge axis		1	12 bits
Enable terminator - limb crossing detectors	2		
Roll axis override	2		
Enable maneuver	2		
Earth Detector on - off	2		
Canopus sensor up-date	6		
Acquisition	2		

Table 19. Guidance and Control Sequencing Requirements

Function	No. of Separate Commands Required	Quantitative or Discrete Output	Comments
Pitch start and stop time	2	Discrete	Start and stop commands required one time per maneuver
Roll start and stop times	4	Discrete	Start and stop commands required up to two times per maneuver
Turn polarity	2	Discrete	Polarity selected two times
Canopus sensor cone angle update	6	Discrete	6 different angles
TVC test signal	2	Discrete	On and off
TVC power	2	Discrete	On and off (switch by power distributor)
TVC electronics on/off	2	Discrete	On and off commands required for trajectory corrections
TVC high/low gain	2	Discrete	For propulsion low and high thrust
Gyros on/off	2	Discrete	On and off commands required for maneuvers

Table 19. Guidance and Control Sequencing Requirements (Continued)

Function	No. of Separate Commands Required	Quantitative or Discrete Output	Comments
Dead-zone select	2	Discrete	On and off commands required for reorientation maneuvers and PSP imaging
Release high- and medium-gain antennas	2	Discrete	
Deploy low-gain antenna	1	Discrete	
Acquisition mode	1	Discrete	Initiates automatic search process
GCS on	1	Discrete	Backup to separation switch
Canopus sensor on	1	Discrete	Starts roll search mode
Gyros rate/position	2	Discrete	Function of modes
Sun sensor inhibit	2	Discrete	Initial attitude hold mode
Reaction control thrusters, high/low	2	Discrete	Function of mode
High-gain antenna hinge position		Quantitative	
High-gain antenna shaft position		Quantitative	
Medium-gain antenna hinge position		Quantitative	

Table 19. Guidance and Control Sequencing Requirements (Continued)

Function	No. of Separate Commands Required	Quantitative or Discrete Output	Comments
Earth detector on/off	2	Discrete	
Accelerometer on/off	2	Discrete	Used during propulsion operation
Roll rate - magnetometer calibration	2	Discrete	Magnetometer calibration Canopus search
Limb-terminator crossing sensor on/off	2	Discrete	
<u>Inputs to C&S</u>			
Accelerometer	1	Pulse train	
Limb and terminator crossings	4	Discrete	Light to dark and dark to light crossings from each sensor
Canopus Acquisition	1	Discrete	

sensors and control devices, its interfaces with the spacecraft structure are critical. Key aspects of this interface are:

- The alignment errors enter into the control system error budget. Table 20 shows the alignments required.
- The celestial sensors need unobstructed fields of view. Table 20 gives the fields of view for these units. The placement of the coarse sun sensor elements require special attention to prevent blind spots especially on the flight capsule side.
- The thrusters should be placed as far as possible from the spacecraft's center of mass to obtain maximum moment arms. Plume impingement should be minimized.
- The location of the center of mass of the planetary vehicle (and spacecraft) must be accurately controlled, since its uncertainty is the largest source of error in the pointing of the thrust vector during velocity corrections.

b. Temperature Control

Thermal control is a major design consideration since all of the units are affected by the thermal environment and many are located outside of the automatic control provided by the equipment compartment. The table on component design parameters in Section IV of Volume 1 gives the allowable operating and nonoperating temperatures of the units of the guidance and control subsystem. Some of the important temperature control problems are as follows:

- The limb and terminator crossing detectors, earth detector, thrusters, antenna drives, and coarse sun sensor elements are located outside of the equipment compartment; thus heaters may be required.
- The solenoid valves must be close to the thrusters in order to make the heated gas technique feasible; thus the valves must be outside of the equipment compartment and may require heaters.

c. Propulsion

The guidance and control subsystem controls the direction and magnitude of the velocity corrections provided by propulsion subsystem. Some aspects of this interface are:

- A dynamic interface exists in that the guidance and control subsystem controls the position of the thrust vector and the engine firing produces disturbance torques.

Table 20. Guidance and Control Subsystem Positioning Requirements

Item	Field of View	Centerline Location		Accuracy of Alignment*	Primary Reference	Comments
		Cone Angle	Clock Angle			
Fine sun sensor	±10°	0°	-	(± .03°) ± .1°	Spacecraft structure	Cone angle is 0° by definition, sensor is a basic reference
Coarse sun sensor,	a 2 π steradian	90°	α	(±0.1) ± 0.5 - 1.0°	To item 1	Bench alignment
	b 2 π steradian	90°	α + 90	(±0.1) ± 0.5 - 1.0°	To item 1	α = configuration dictated
	c 2 π steradian	90°	α + 180	(±0.1) ± 0.5 - 1.0°	To item 1	
	d 2 π steradian	90°	α + 270	(±0.1) ± 0.5 - 1.0°	To item 1	
Canopus sensor	30° x 4° Cone clock	Nominally 90°	0	(±0.03) ± 0.1°	Fine sun sensor and S/C structure	Clock angle is 0° by definition sensor is a basic reference
Near earth sensor	50° x 30° Cone clock	Variable	90°	± 0.5°	To item 3	Bench alignment, centerline adjusted to fit mission
Control gyros	-	-	-	(±0.1°) ± 0.14°	To item 3	Bench alignment
Accelerometer	-	0°	0	± 0.5°	Parallel to roll axis	
ACS jets 8 pitch and yaw 4 roll	-	0, 180° 90°	α, α + 180° } P&Y α ± 90° roll as req'd	2.50° 2.50°	To basic reference	Offsets to spacecraft ± axis
Limb and terminator sensor	90° x 2° Cone clock	90°	variable	± 0.5°	To item 3	Bench alignment centerline adjusted to fit mission

*Alignment accuracies in parenthesis are unit mounting surface to primary reference. Other numbers are accuracy of sensor measurement axis with respect to primary reference, e.g., the fine sun sensor's optical axis with respect to the primary reference. Where no numbers are given in parenthesis, alignments are not critical.

- A mechanical interface exists in that the thrust vector control is provided by guidance and control subsystems actuators which are attached to the combustion chamber.
- An electrical interface exists through the computing and sequencing subsystem for the control of velocity magnitude by the accelerometer.

2.4 Design Description

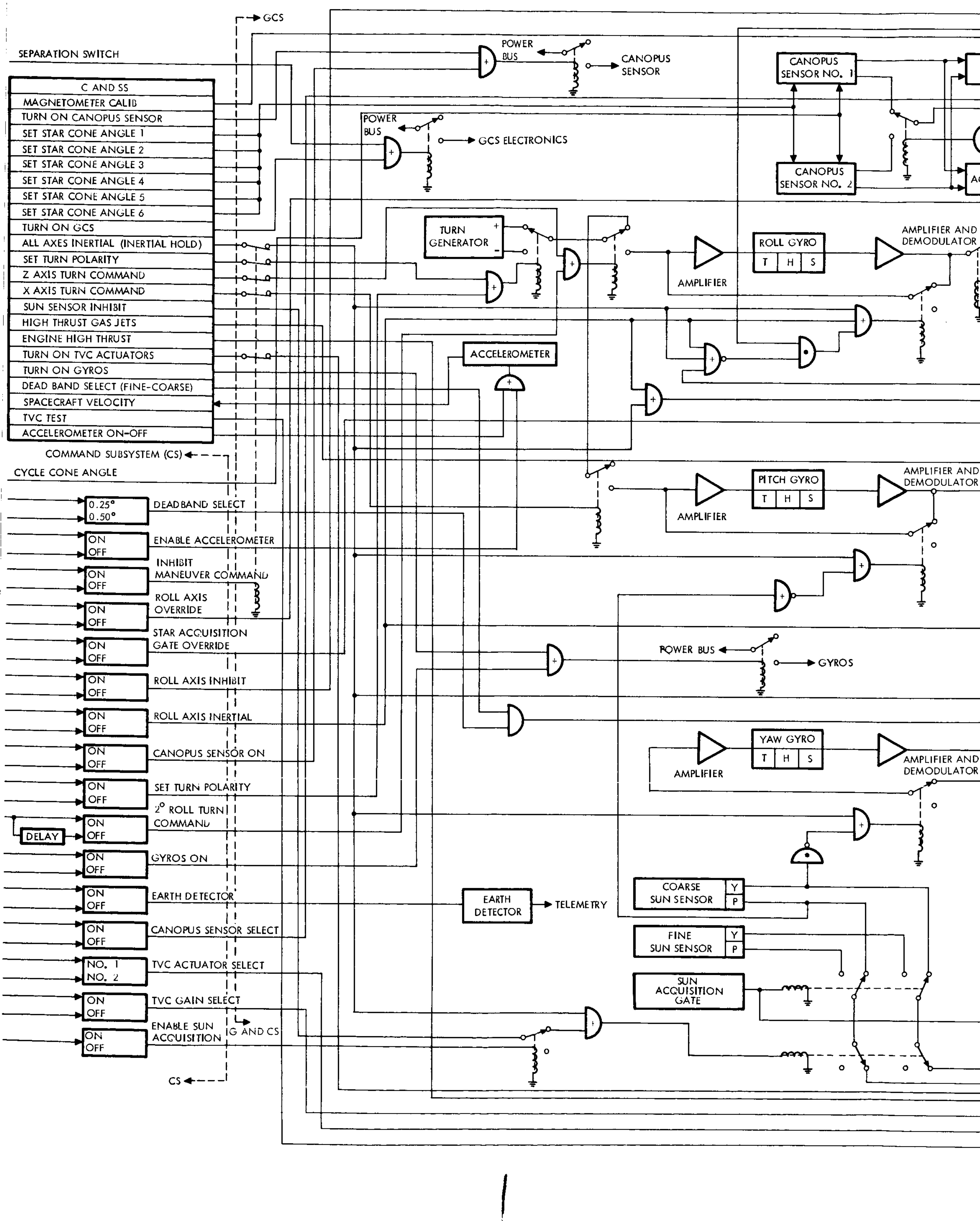
2.4.1 Subsystem Organization

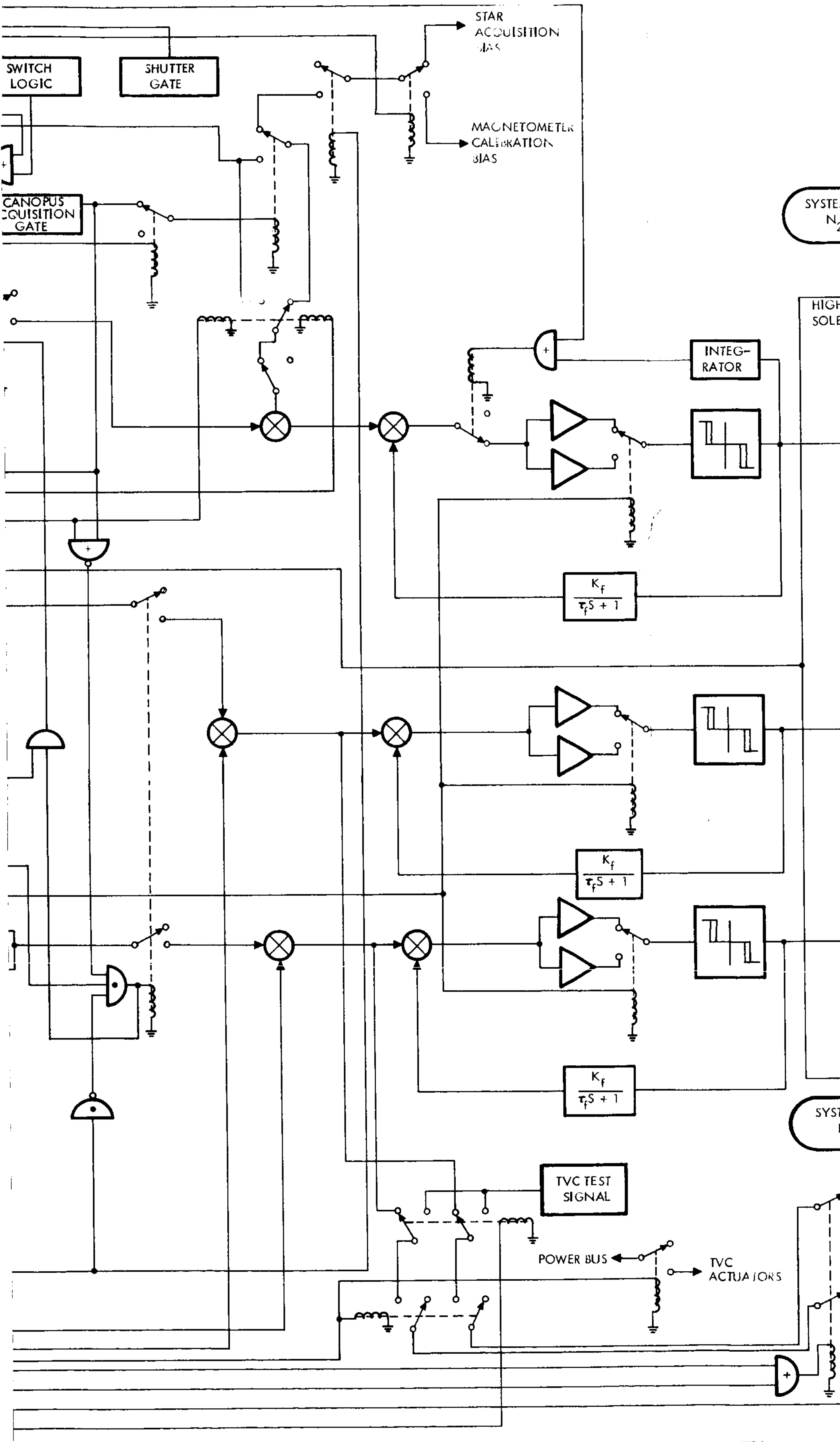
A functional schematic of the guidance and control subsystem is given in Figure 23. The implementation is straightforward and conservative, based on proven components and techniques to achieve a simple, reliable system. The only redundancy shown is that on the unit level. Major features of the system are as follows:

- Redundancy is obtained in the reaction control assembly by virtue of two separate pneumatic supplies with one of the thrusters for a given torque on each of the systems.
- The same thrusters are used for both high (3 lb) and low (0.2 lb) thrust by switched high and low pressure regulators.
- Fine and coarse pointing control is obtained by gain changes in the switching amplifiers to produce small (± 0.25°) and large (±0.50°) deadbands.
- The thrust vector control actuator loop has different gains for high and low thrust propulsion operation.
- Canopus sensor redundancy.
- A test capability is provided to determine, prior to engine firing, that the TVC actuators and electronics are working properly, and to select normal or standby components.

2.4.2 Normal Mission Operation

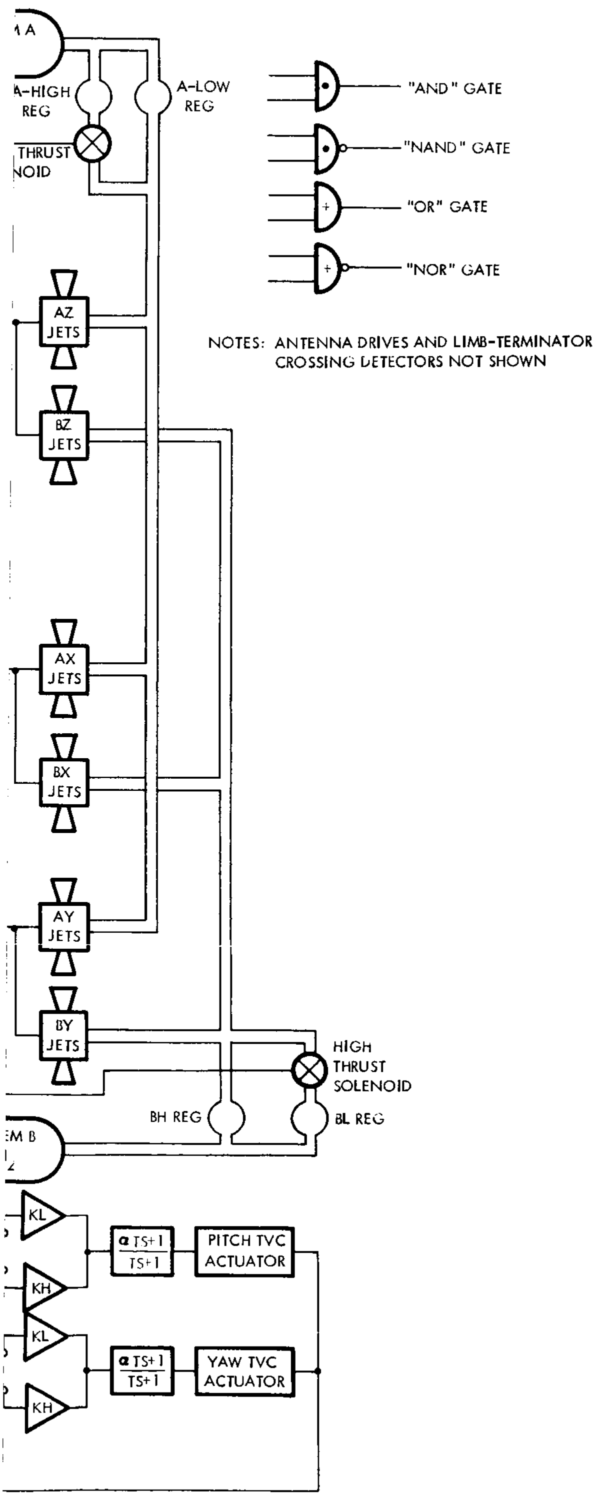
The guidance and control subsystem provides several modes of operation, incorporating various combinations of sensors and torque sources. The guidance and control subsystem modes and the associated equipment elements are summarized in Table 21 and described in the following sections.


GCS
SEPARATION SWITCH
POWER BUS
CANOPUS SENSOR
CANOPUS SENSOR NO. 1
CANOPUS SENSOR NO. 2
C AND SS
MAGNETOMETER CALIB
TURN ON CANOPUS SENSOR
SET STAR CONE ANGLE 1
SET STAR CONE ANGLE 2
SET STAR CONE ANGLE 3
SET STAR CONE ANGLE 4
SET STAR CONE ANGLE 5
SET STAR CONE ANGLE 6
TURN ON GCS
ALL AXES INERTIAL (INERTIAL HOLD)
SET TURN POLARITY
Z AXIS TURN COMMAND
X AXIS TURN COMMAND
SUN SENSOR INHIBIT
HIGH THRUST GAS JETS
ENGINE HIGH THRUST
TURN ON TVC ACTUATORS
TURN ON GYROS
DEAD BAND SELECT (FINE-COARSE)
SPACECRAFT VELOCITY
TVC TEST
ACCELEROMETER ON-OFF
POWER BUS
GCS ELECTRONICS
TURN GENERATOR
AMPLIFIER
ROLL GYRO
T H S
AMPLIFIER AND DEMODULATOR
ACCELEROMETER
COMMAND SUBSYSTEM (CS)
CYCLE CONE ANGLE
0.25°
0.50°
DEADBAND SELECT
ON
OFF
ENABLE ACCELEROMETER
INHIBIT MANEUVER COMMAND
ROLL AXIS OVERRIDE
STAR ACQUISITION GATE OVERRIDE
ROLL AXIS INHIBIT
ROLL AXIS INERTIAL
CANOPUS SENSOR ON
SET TURN POLARITY
2° ROLL TURN COMMAND
DELAY
GYROS ON
EARTH DETECTOR
CANOPUS SENSOR SELECT
NO. 1
NO. 2
TVC ACTUATOR SELECT
TVC GAIN SELECT
ENABLE SUN ACQUISITION
G AND CS
CS
PITCH GYRO
POWER BUS
GYROS
YAW GYRO
EARTH DETECTOR
TELEMETRY
COARSE SUN SENSOR
Y
P
FINE SUN SENSOR
SUN ACQUISITION GATE


STAR ACQUISITION BIAS
SWITCH LOGIC
SHUTTER GATE
MAGNETOMETER CALIBRATION BIAS
CANOPUS ACQUISITION GATE
INTEG-RATOR
$\frac{K_f}{\tau_f S + 1}$
TVC TEST SIGNAL
POWER BUS
TVC ACTUATORS

Figure

2



3. Guidance and Control Subsystem Functional Schematic

3

Guidance and Control Subsystem Modes	Coarse Sun Sensor	Fine Sun Sensor	Star Sensor	Gyros		
				Rate	Pos	
					Not Torqued	P&R Torqu
1. Initial Attitude Hold				●	●	
2. Sun Acquisition	●	●		●		
3. Roll Spin-Search		●		●		
4. Canopus Acquisition		●	●	●		
5. Cruise and Orbital Operations		●	●			
6. Maneuver (Reorientation)					● (Yaw)	●
7. Inertial Hold					●	
8. Midcourse (Propulsion Low)					●	
9. Orbit Injection (Propulsion High)					●	

Table 21. G and CS Mode Operations

ed	Accelerometer	Earth Sensor	Jets High	Jets Low	TVC Hi Thrust	TVC Low Thrust	Ant Drives	Mode Initiation Main	Back
			•				•		
			•					C&S	Comm
			•					Sense Sun	Comm
		•	•					Acq Sun-Fine Sens	Comm
		•					•	Initial C&S others fine Sun Acq	C&S
				•			•	Canopus Acq	C&S
			•					C&S	C&S
			•				•	C&S	Comm
	•		•			•	•(1)	C&S	Comm
	•		•		•		•(1)	C&S	Comm

2

up	Mode Termination		Comments
	Main	Backup	
	C&S	Command	
and			
and	Acquisition Sun-Fine Sens		1st acquisition: enable by C&S. Backup by command
and	C&S	Command	Calibrate magnetometer; roll bias inserted 0. 22° /sec
	Canopus Acquisition		Roll bias inserted 0. 1°/ sec canopus acquisition verified by earth sensor and high gain antenna position
	Command C&S		Cruise mode. While imaging of PSP limit cycle reduced
	C&S	C&S	Switch to small deadband before maneuver.
and	C&S	Command	Loss of sun or Canopus will also initiate
and	Accel	C&S	(1)Desired
nand	Accel	C&S	

a. Launch

Power is turned on prior to launch for the gyros which are in the caged or rate mode. The remainder of the control system is disabled to prevent gas expulsion.

b. Initial Attitude Hold Mode

To provide communication with the earth after booster separation and during the initial eclipse, it is necessary to maintain the planetary vehicle attitude within ±10 degrees of the separation attitude. This function is performed by the initial attitude hold mode which operates in two discrete submodes. Initially, the gyros are in the caged (rate) mode and the switching amplifiers operate the high thrust (3 lb) reaction jets. As in all modes, the derived rate filters are maintained around the switching amplifiers. This configuration operates as a rate-nulling mode which will reduce the initial tip-off rate (1.5 degrees per second maximum) to approximately 0.01 degree per second in less than 10 seconds. Following the initial rate nulling period, the C&S issues a command to switch to the inertial attitude hold mode, which places the gyros in the rate-integrating mode. While the vehicle is in eclipse, the C&S commands sun acquisition, but since the sun is not present the guidance and control remains in the attitude hold mode. A block diagram of a single axis (pitch, roll or yaw) initial attitude hold mode is shown in Figure 24. (For clarity, redundancy is not shown on the block diagrams.)

Based on an initial rate of 1.5 deg/sec and a gas jet thrust level of 3 pounds, gas consumption during the initial attitude hold mode is estimated at approximately 1.5 pounds.

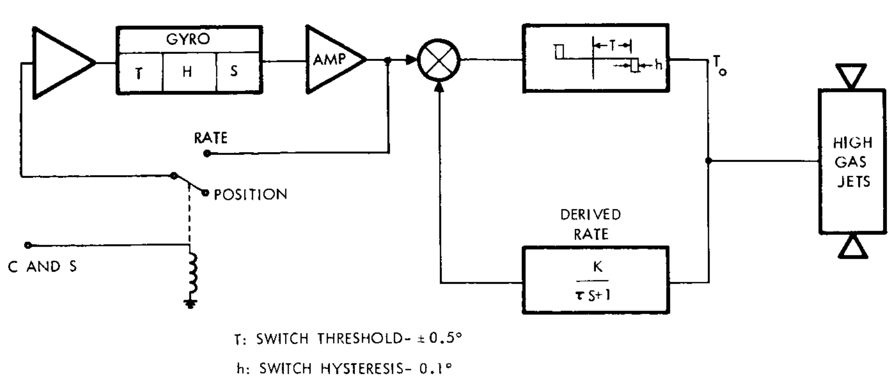


Figure 24. Block Diagram for Initial Hold (Single Axis)

c. Acquisition Mode

Yaw and Pitch Sun Acquisition Mode. After nulling the initial separation rates and maintaining inertial hold, the vehicle emerges from eclipse. When the sun is sensed by the coarse sun sensors (and not by the sun acquisition gate), control reverts to the acquisition mode and alignment of the vehicle roll axis with the spacecraft sun line begins. The control logic switches the gyros from the position mode to the rate mode and enables the coarse sun sensor inputs. The gyro outputs are subtracted from the saturated sensor outputs to establish a rate command to the vehicle. The electronic switching amplifiers and derived rate filters are the same devices used in the initial attitude hold mode. Although the derived rate filters are not required for this mode, they are used in the circuits to provide the capability of acquiring if a gyro fails. As indicated in Figure 25, the output signal of the coarse sun sensors saturate at 10 degrees error. Since the rate gyro gain has been selected as 50 seconds, the saturated region from 10 to 170 degrees commands a rate of 0.2 degree per second. Illumination of the fine sun sensor occurs when the planetary vehicle is within 10 degrees of the sun line and results in the sun acquisition gate switching the input from the coarse to the fine sun sensor. Since the output of the fine sun sensor decreases linearly from 10 degrees, the angular rate is commanded to decrease from 0.2 degree per second to zero as the pointing error is nulled.

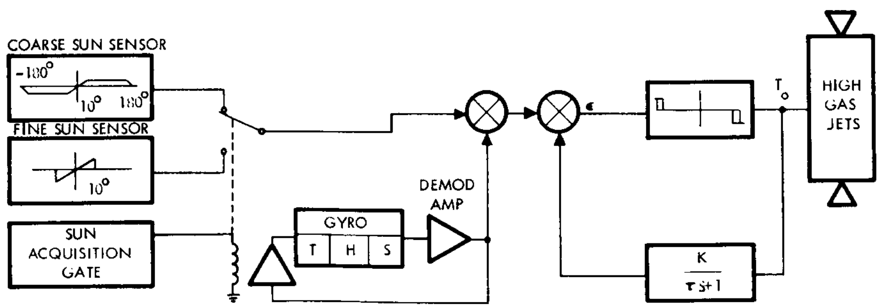


Figure 25. Block Diagram for Sun Acquisition (Pitch or Yaw Axis)

The possibility of shaping the coarse sun sensor characteristics near null such that the switching of sensors can be eliminated will be studied in Phase IB.

Expected performance characteristics are:

- Sun acquisition may take as long as 30 minutes.
- Total nitrogen consumed for an initial acquisition, 4 mid-course correction acquisitions, acquisition after deboost and capsule separation and 3 orbit trim acquisitions is approximately 5 pounds.

Roll-Search Mode. When the fine sun sensor is illuminated, a controlled spin rate about the roll axis is commanded (see Figure 26). Normal operation is to initiate the Canopus search mode. The Canopus search mode can be delayed, nowever, by a C&S magnetometer calibration override command. Assuming a magnetometer device similar to that employed on the Mariner C, a fixed roll rate will be commanded. If another type of magnetometer is selected for Voyager, an altogether different sequence of maneuvers may be required such as sequential rotation about all axes. When the spacecraft roll rate reaches the commanded value, the roll gyro output cancels the spin command signal, the jets are turned off and the vehicle continues to rotate about the roll axis.

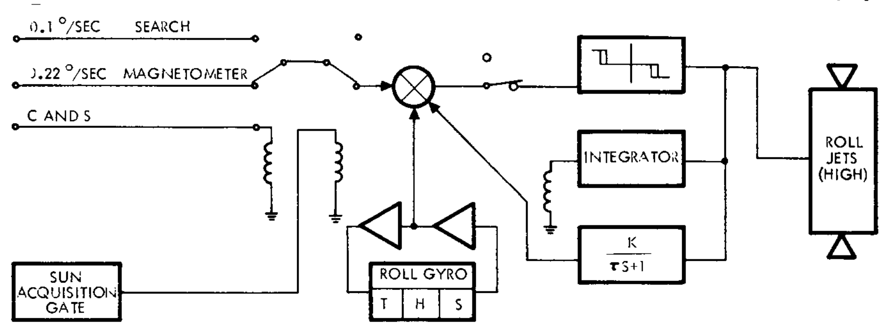


Figure 26. Block Diagram for Roll Search Mode and Magnetometer Calibration

During the magnetometer calibration, a map of the star magnitudes as sensed by the Canopus sensor is transmitted to earth to aid in Canopus acquisition. When the Canopus search mode is initiated, the C&S opens the Canopus sensor shutter and commands a roll search rate of 0.1 deg/sec. When a star of sufficient brightness appears in the field of view of the Canopus sensor, control is switched from the gyro to the Canopus sensor and roll acquisition is accomplished.

For the initial acquisition, an earth detector, which is illuminated by earth-shine if the vehicle is in the nominal cruise attitude, verifies proper Canopus acquisition. In addition, Canopus acquisition can be verified by pre-positioning the high gain antenna so that the beam intersects earth with the spacecraft in the proper cruise attitude. The intensity of the Canopus sensor output, telemetered to earth, also provides verification of acquisition. If the wrong star is acquired, a command is sent from earth to unlock and continue roll search.

e. Cruise Mode

When a star of sufficient brightness appears in the field of view of the Canopus sensor, the star acquisition gate initiates a 30 second timer which turns the gyros off, lowers the reaction control assembly jet thrust from 3 pounds to 0.2 pound and switches the guidance and control subsystem to the cruise mode. The function of the cruise mode is to maintain three-axis stabilization during transit to Mars (except during midcourse maneuvering and retropropulsion firing) and in orbit about Mars (except during orbit trim operations, eclipse periods, and capsule separation). The pitch and yaw axes employ the two-axis fine sun sensor for attitude reference with roll stabilization about the sun line accomplished by using the Canopus sensor as a reference. Block diagrams of the pitch or yaw axis and the roll axis cruise modes are given in Figures 27 and 28.

The design criteria employed to define the system parameters are:

- A limit cycle rate less than 10 degrees per hour during PSP imaging.
- A nominal limit cycle amplitude of ±0.5 degree.
- A minimum impulse firing time of 0.025 second.

The limit cycle rate requirement is specified to limit image smear during PSP photographic operations. The nominal limit cycle amplitude of 0.5 degree is required to meet antenna pointing accuracy requirements and to provide an acceptable nitrogen expenditure rate. Prior to maneuvers and during periods of photographic operation, the C&S commands the guidance and control subsystem to switch to a ±0.25 degree deadband in all channels. The nominal limit cycle firing time of 25 milliseconds was considered to be the minimum which could be reliably attained. The gas jet thrust level was selected at 0.2 pound, which resulted in an acceleration constant that varied from 0.14 to 0.94 milliradian per second squared. This value was selected on the basis of gas weight and limit cycle firing margin.

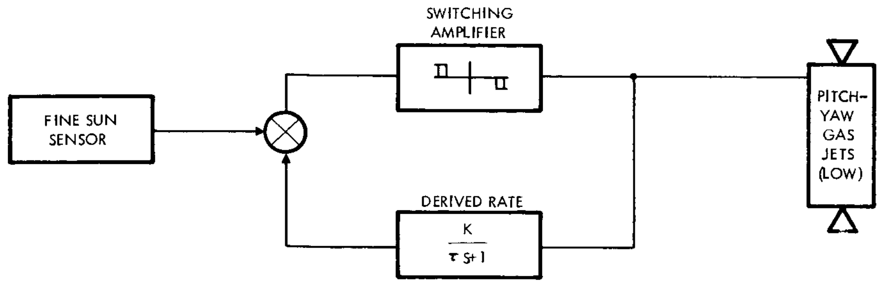


Figure 27. Block Diagram for Cruise Mode (Pitch or Yaw Axis)

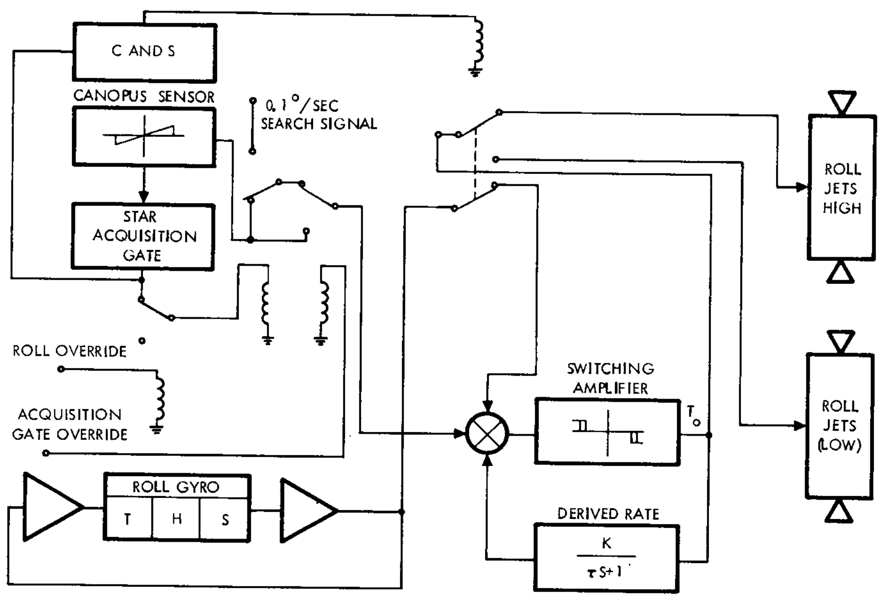


Figure 28. Block Diagram for Canopus Acquisition and Roll Cruise Mode

The nominal cruise mode limit cycle characteristics are shown in the phase plane plots of Figure 29. For an initial rate of 36 deg/hr, the cruise mode acquisition and limit cycle operation are illustrated for both the high inertia spacecraft prior to capsule separation and the low inertia spacecraft after capsule separation.

f. Inertial and Maneuver Mode

In the inertial mode, planetary vehicle (spacecraft) attitude error signals for all channels are provided by the gyros operating in the rate-integrating configuration. As in the cruise mode, rate damping is obtained from the derived rate filters. To provide satisfactory operation during all phases of the Voyager mission, the derived rate saturation level was set at 20 degrees.

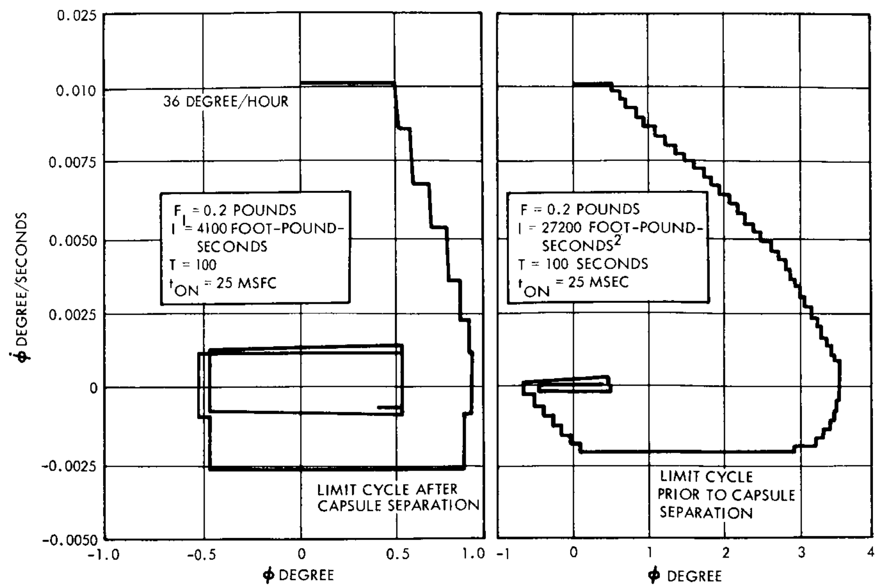


Figure 29. Cruise Mode Limit Cycle Characteristics

About an hour before a maneuver, commands are sent to turn on the gyros and to load maneuver data in the C&S. The telemetry subsystem is switched to the pre-maneuver mode to enable checkout of the thrust vector control components and selection of redundant components, if required. The checkout consists of applying test signals (by C&S) and observing the response via telemetry. When the maneuver enable and start time arrive, the control dead zone is set to ±0.25 deg, the gyros are uncaged and switched to replace the sun sensors, and torquing commands are transmitted by the C&S. The turning rate commands for maneuvers are provided by precision current generators which torque the gyros at 0.2 deg/sec. The polarity of the current, hence the direction of the turn, is determined by the presence or absence of a C&S signal, as indicated in Figure 30. In the maneuver sequence, a roll turn is followed by a pitch turn. This sequence results in a gas saving since the roll inertia is lowest. In some cases, a second roll turn may be required to point the antenna to earth. Following completion of a reorientation maneuver, the C&S removes torquing commands and the new orientation is maintained by inertial control. The guidance and control subsystem returns to the acquisition mode when the C&S removes the inertial and maneuver mode signal.

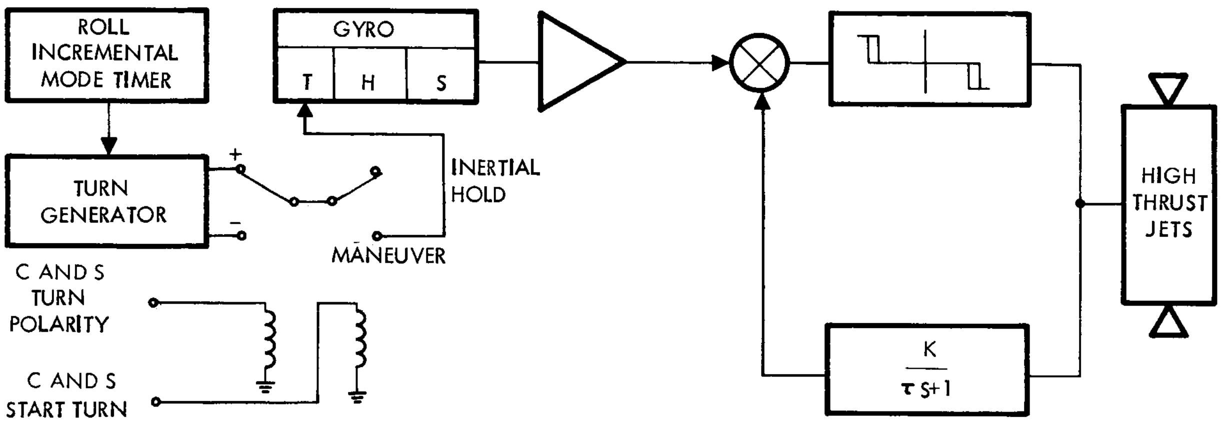


Figure 30. Single-Axis Inertial & Maneuver Mode Block Diagram

The inertial and maneuver mode is also used for periods of sun or Canopus occulation which can occur in orbit about Mars. After the eclipse, the guidance and control subsystem is returned to either the acquisition mode or the cruise mode as determined by the acquisition logic.

Based on a 180-degree maneuver angle, the maximum time and gas consumption during a single axis reorientation were determined by analog computer simulation to be 20 minutes and 0.39 pound, respectively. The maximum angular error between the gyro and the spacecraft was 6 degrees, which is within linear range requirement for the gyro.

g. Thrust Vector Control Mode

Figure 31 shows a block diagram of the pitch or yaw axis thrust vector control mode. This mode is used during periods of main engine operation and is preceded by the inertial hold mode. Attitude reference signals are provided by the gyros. Pitch and yaw control torques are obtained by gimballing the main engine and these torques are augmented by the high thrust jets. The high thrust cold gas jets are used for stabilization about the roll axis, using the inertial hold mode. Stability in pitch and yaw is provided by passive lead networks operating on the gyro signals. The gimbal actuators are redundant, with constant speed DC motors which drive magnetic clutches in response to gimbal angle commands.

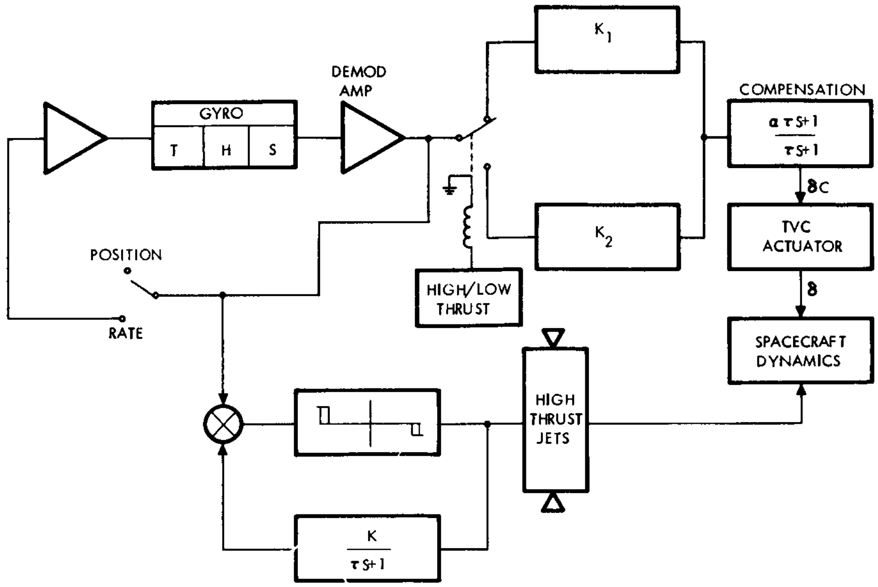


Figure 31. Block Diagram of Inertial and Maneuver Mode (Pitch or Yaw)

In Figure 31 a gain change from high to low thrust operation is indicated. The high thrust (7750 pounds) is used during the orbit injection mode; the low thrust (1150 pounds) is used for midcourse and orbit trim velocity corrections. The thrust variation imposes the necessity to change gains with thrust level commands. Based on pointing accuracy requirements and dynamic response considerations, the compensation parameter values shown in Figure 31 were selected as:

K_1 = 6 for low thrust operation

K_2 = 1 for high thrust operation

α = lead-lag ratio = 15

τ = 0.067 second

The velocity changes during engine thrusting are sensed by a linear accelerometer which has an accuracy of 0.1 percent. The quantization level of the velocity increments is 0.015 ft/sec/pulse. The velocity increment pulses are counted by the C&S and thrust is terminated when the desired velocity correction has been attained. After powered flight has terminated, the C&S switches to the sun acquisition mode.

2.4.3 Alternate and Backup Modes of Operation

a. Acquisition, Pitch and Yaw Gyro Failure

In case of failure of a gyro during sun acquisition, a design criteria was to provide automatic backup by the derived rate filters. Although the gas consumption is increased by a factor of approximately three with the backup configuration compared to the normal system, satisfactory sun acquisition can be achieved. Figure 32 shows a phase plane trajectory of the pointing error and rate for the backup acquisition mode from an initial pointing error of 180 degrees, without using gyros.

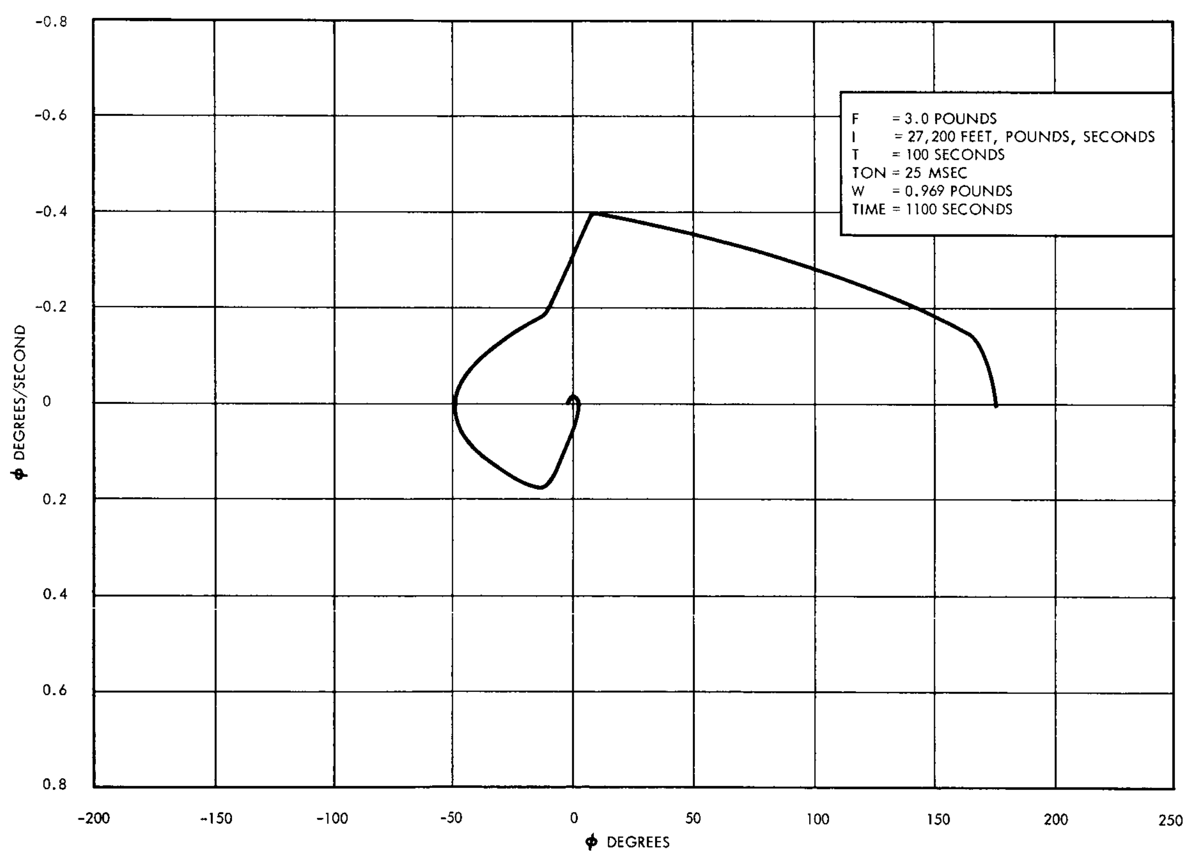


Figure 32. Pitch or Yaw Backup Sun Acquisition Trajectory

b. Acquisition, Roll Gyro Failure

Should the roll gyro fail to provide a rate signal to null the calibration or search rate command, the spacecraft would continue to accelerate until gas exhaustion or an override command was issued. To prevent this, the roll gas jet firing signals are electrically integrated and, after a firing time slightly in excess of the maximum expected (approximately 50 seconds), the input signal to the switching amplifier is interrupted, allowing backup modes of roll control to be exercised. This function requires that the integrator output be reset to zero following each roll maneuver mode. The reset will be accomplished by a command from the C&S.

c. Canopus Sensor Failure

In the event of a Canopus sensor failure, a roll incremental mode is provided. The flow of torquing current is controlled by fixed timers which increment the gyros ±2 degrees. The spacecraft can thus be incrementally positioned around the roll axis by ground radio command until the earth sensor or medium gain antenna indicates a proper roll attitude.

d. Pneumatics Failure

To improve system reliability, redundant tankage, gas feed systems and valves are used to supply coupled pairs of nozzles. If a valve fails to open, the acceleration constant will be halved and pure couples will no longer be applied to the vehicle in that axis. Although the linear velocity imparted to the spacecraft would normally be negligible, the velocity error can be corrected during midcourse maneuvers. If a valve fails to close, the disturbance torque applied to the spacecraft will be countered by the opposing coupled pair. The guidance and control subsystem will operate in a stable, one-sided limit cycle until the gas supply feeding the failed valve is exhausted. The gas supply system is sized to provide acceptable control throughout the Voyager mission in the event of a single open valve. In addition, the high and low thrust reaction jet systems can be used to provide mutual backup, although gas consumption would be excessive in cruise if high thrust were utilized and control during maneuvers would be sluggish and underdamped if the low thrust is used.

2.4.4 Subsystem Elements

Following are brief descriptions and preliminary specifications of the major units of the guidance and control subsystem. The Canopus sensor and the earth detector are not included since they are essentially the same as the units used on Mariner and have been described in detail in the Task A study report. A short discussion on the antenna drive electronics is included in the antenna drive section.

a. Sun Sensors

The sun sensors provide analog signals which locate the sun about two axes relative to the spacecraft over a field-of-view of 4π steradians and provides simultaneous electrical nulls in two axes which represent the preferred orientation of the spacecraft relative to the sun. The sensor consists of a coarse sensor assembly and a fine sensor assembly. The coarse sensor assembly consists of two pairs of solar cells back-to-back, with plano-convex lenses in optical contact with the cell surfaces, as shown in Figure 33. Solar cells are connected with opposing polarities across a low resistance producing a linear output in the region of ±20° about null. The coarse solar cells are mounted on the spacecraft periphery with maximally unobstructed viewing angles. The null stability of the coarse sun sensors is between 0.5 and 1 degree (as determined by the geometrical and thermal characteristics of the mounting structure) so that with suitable mounting surfaces and with proper initial alignment the coarse sun sensors can provide a good backup capability in the event of a failure of the fine sun sensor.

The fine sun sensor consists of a silicon photovoltaic quad cell mounted behind a mask which acts as a shadowing structure (see Figure 34). The sensor is designed to have nearly linear output in two orthogonal axes when the sun angle is within 10 degrees of the sun sensor axis.

The scale factors for the coarse and fine sun sensors are designed to be nearly equal between 8 and 10 degrees from null. The attitude control system is switched from the coarse sun sensor to the fine sun sensor within the 8 to 10-degree band so that the transition will not produce a discontinuity in the signal. The switching is initiated by a miniature silicon solar cell behind a plate having a pinhole aperture.

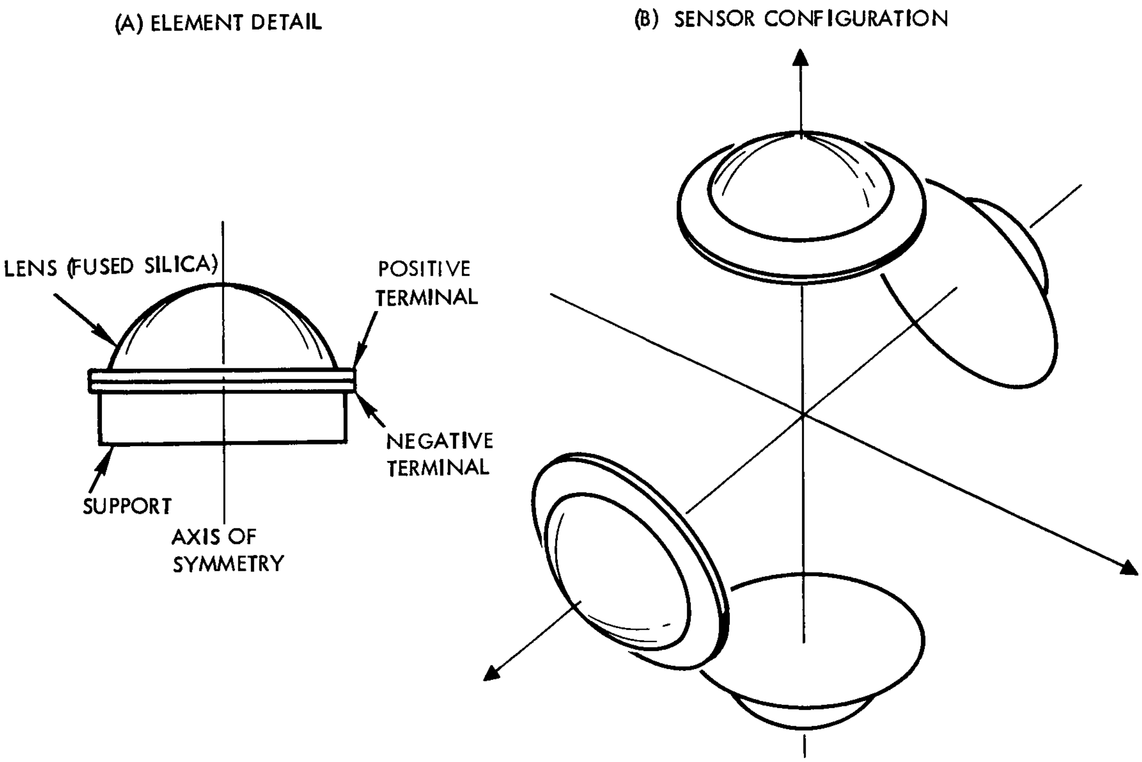


Figure 33. Coarse Sun Sensors

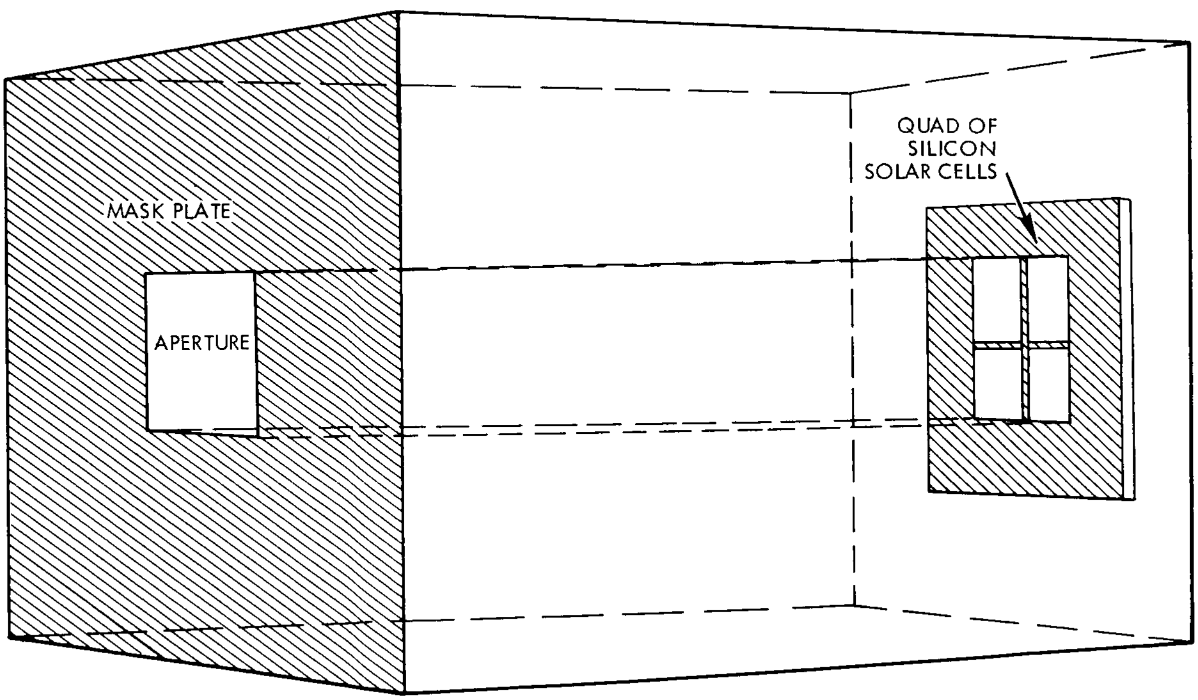


Figure 34. Fine Sun Sensors

Switching from coarse sun sensors to a fine sun sensor makes it possible to have a 4π steradian acquisition field of view without susceptibility to null errors caused by light reflected from the surface of Mars during the orbital phase of the mission. A preliminary specification for the sun sensor is given in Figure 35.

b. Gyro Reference Assembly

The gyro reference assembly consists of three single-degree-of-freedom gyros, a power converter, precision torquing current suppliers, and temperature control equipment. The equipment for each axis is essentially identical; a typical axis (roll) will be described by referring to the preliminary specification of Figure 36.

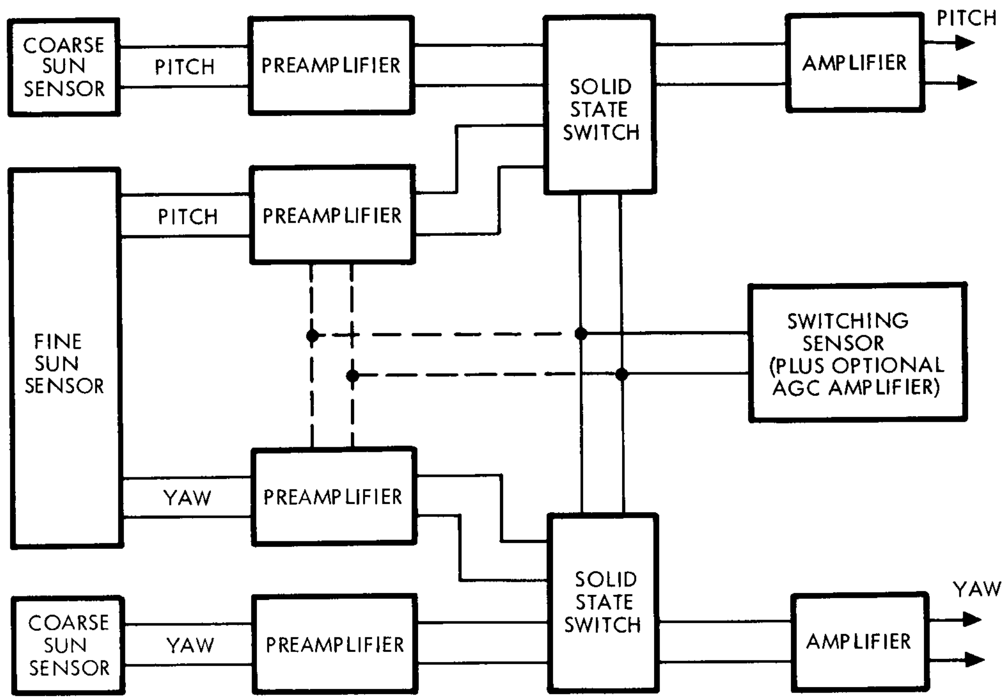
In the rate mode Switch No. 1 is as shown and the gyro is caged with an analog torquing signal. The electrical output is the voltage across the gyro torquer, and is proportional to the input roll rate from the spacecraft. Reorientation of the spacecraft is achieved by opening Switch No. 1 in the roll gyro loop and closing the roll gyro current Switch No. 2. The precise torquing current of the desired polarity is applied to the gyro torquer for a predetermined length of time. In the absence of torquing signals, the gyro maintains its inertial reference. The switching sequence is designed in such a manner that the gyros will always be caged by means of the rate loop (Switch No. 1 closed) prior to operating open loop (Switch No. 1 open). This insures that the gyro floats will be at pickoff null prior to their use as an inertial reference in the inertial hold and turn modes.

c. Guidance and Control Electronics

As diagrammed in Figure 23, the guidance and control subsystem electronics assembly, for which a preliminary specification is given in Figure 40, supplies power to the reaction jet solenoids and to the thrust vector control actuators in response to signals from the sun sensors, the Canopus sensor, and the gyro reference assembly. Mode control and switching signals are provided internally by sensors in the guidance and control subsystem and by the computing and sequencing and the command subsystems. The switching shown in Figure   is for functional illustration only. All switching will actually be performed using transistor switches to minimize electromagnetic interferences.

PRELIMINARY SPECIFICATION

Sun Sensor



Function

To provide angular position information about the pitch and yaw axes of the spacecraft relative to the sun.

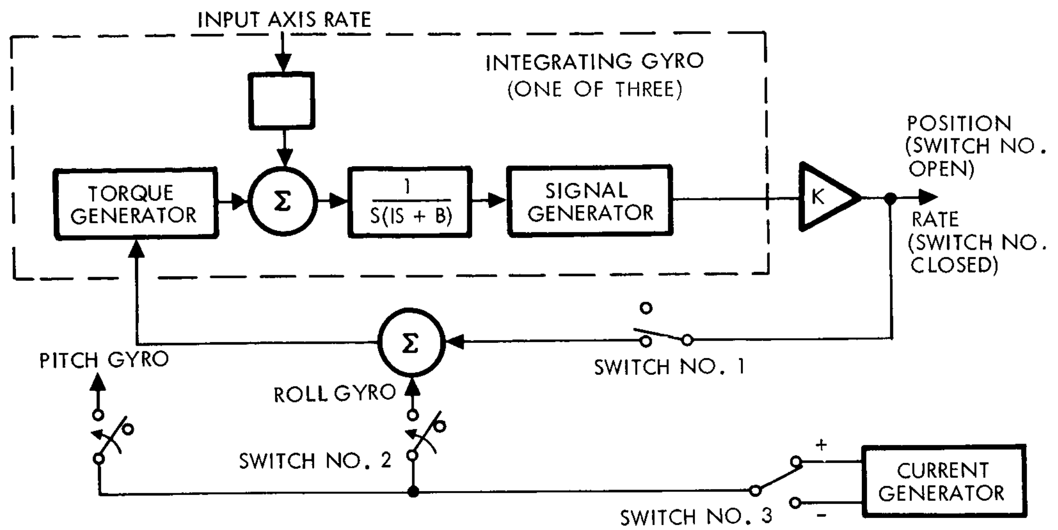
Performance Characteristics		
Fine Sensor Assembly	Field of view	±10° cone
	Linearity (each axis)	±10%
	Null accuracy (each axis)	±0.1°
Coarse Sensor Assembly	Field of view	4π ster
	Null accuracy (each axis)	±1°
	Linearity (over ±20° each axis)	±10%
Physical Characteristics (includes electronics)	Size	20 cubic inch
	Weight	1.0 pound
	Power	700 milliwatts

Sun Sensor

Figure 35. Preliminary Specification — Sun Sensor

PRELIMINARY SPECIFICATION

Gyro Reference Assembly (GRA)



Physical Characteristics (GRA)

Size	180 cubic inches (6 x 6 x 5 inches)
Weight	10 lb
Power required	17.5 watts average (38.5 watts peak)

Function

Provide spacecraft rate and position information to the GCS

Mode	Sw 1	Sw 2	Sw 3	Scale Factor
Inertial hold	OS	AS	EP	5.00 volts dc/ia degree
Rate	AS	AS	EP	30 volts dc/ia degree/second
+ Turn	OS	OS	AS	
- Turn	OS	OS	OS	

AS-as shown, OS-opposite shown, EP-either position

GYRO PRELIMINARY SPECIFICATION*

Type:	Floated, rate integrating, single degree of freedom
Weight:	1.0 pound (maximum)
Size:	2.0 inches diameter x 3 inches long
Input axis angular freedom:	±15 degrees (minimum)
Motor:	
Type	Synchronous hysteresis, 4 pole
Excitation	Two phase square wave 800 cps
Starting power	10.0 watts peak for less than 45 seconds
Running power	2.0 watts (maximum)
Signal generator power:	4000 cps, 0.5 watts (maximum)
Spin bearing:	Hydrodynamic with notches added to the journal portion
Long term drift:	
G-insensitive	Initial value +0.3 degree/hour (maximum) stability 0.4 degree/hour, 3σ, 1 year
G-sensitive	Initial value +1.0 degree/hour-g (maximum) stability 0.7 degree/hour-g, 3σ, 1 year

*Based on Nortronics GI-M1 Type Gyro

Gyro Reference Assembly (GRA)

Figure 36. Preliminary Specification — Gyro Reference Assembly (GRA)

The guidance and control subsystem electronics uses redundancy in several functions to increase reliability, such as in the switching and logic functions. The technique used is a straightforward series-parallel method which requires four times the number of components as a non-redundant circuit but permits operation in the presence of any part failure. To keep the diagram simple, not all the redundancy was shown in Figure 23. Several of the important circuits which were included to improve reliability are explained below.

The circuit for solenoid valve control is shown in Figure 37. As indicated, this method utilizes triple redundancy with voting logic incorporated in the valve drivers. Three identical channel inputs are connected to six valve drivers. Each input controls two of the six drivers and the output can be turned on or off by any two signals above the threshold; therefore, should one channel fail the voting logic will select the two similar outputs as correct.

As shown in Figure 38, thrust vector control is achieved by amplifying gyro angle signals to obtain engine deflection commands. Engine deflection is obtained by activating clutches to apply torque to the engine gimbal.

The gyro output signals are first amplified by the redundant preamp, which consists of three integrated operational amplifiers connected in a parallel summing configuration to provide the correct output in spite of a failure in any one of the amplifiers. Any failure which causes the output of the parallel combination to change will be treated as an error by the other two amplifiers, which will offset in the proper direction to cancel the error. The resulting error will then depend only on the gain of the parallel amplifiers and will be below the tolerances allowed. The output of the preamps is further amplified by the redundant power amplifiers which drive the two clutch coils in a bridge configuration, thereby providing positive or negative directional control from a single redundant amplifier.

The power amplifiers (Figure 39) will utilize a relatively new technique, presently employed on the throttle actuator of the descent engine of the LEM, which will not present a hardover condition to the clutch winding regardless of the type of failure within the amplifier. In

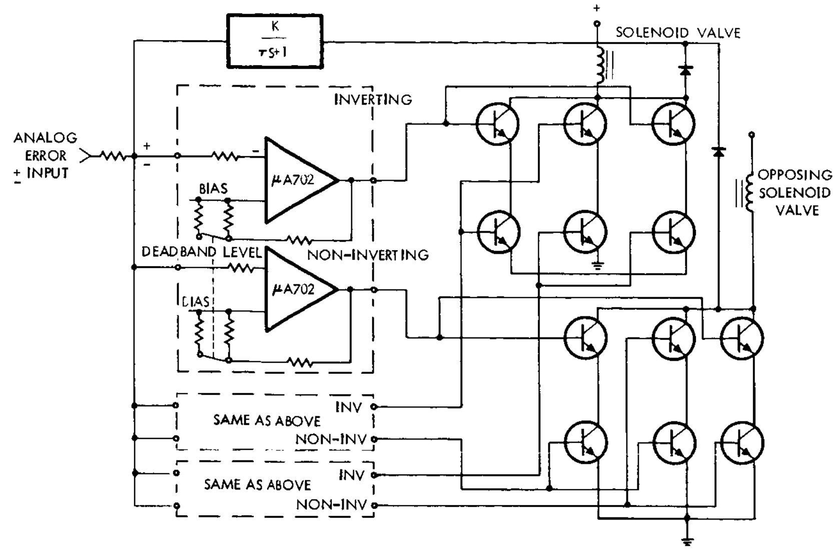


Figure 37. Single Channel Reaction Jet Control With Redundancy

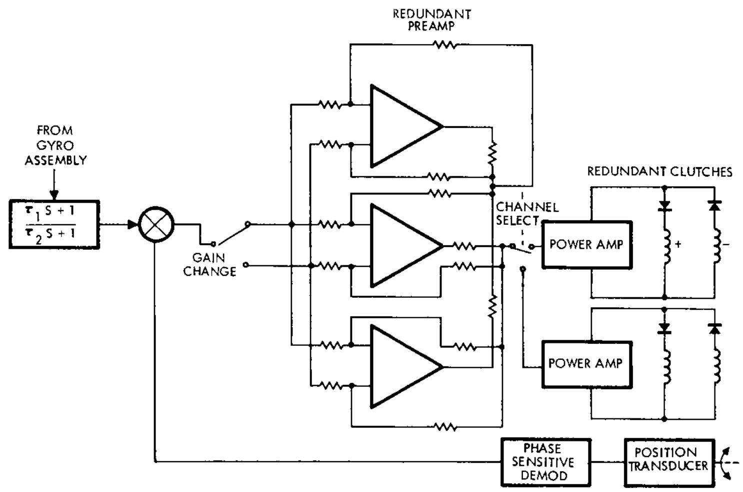


Figure 38. Thrust Vector Control Circuitry

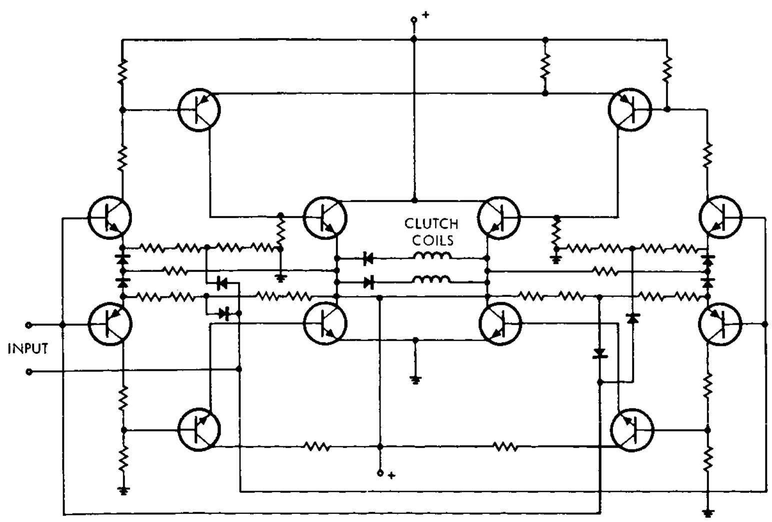


Figure 39. Fail-Safe Power Amplifier

PRELIMINARY SPECIFICATION

Guidance and Control Electronics

Function

Process signals from C and S, sensor electronics, and gyro electronics and provide actuating signals to the jet valve actuators and control signals to the thrust vector control actuator clutches. It also supplies the necessary regulated power required in the antenna drive electronics.

Physical Characteristics

Size	7 x 6 x 11 inches	Power	50 vdc	50 vac
Weight	13 pounds	Peak	10 watt	28 watt
		Minimum	5 watt	20 watt

General Information

The valve control electronics consists of pitch, roll, and yaw, channels identical except for feedback integration in roll to prevent spinup if the roll gyro should fail.

The thrust vector control electronics consists of a pitch and a yaw channel which are identical in every respect. By application of a discrete signal the forward gain of both channels can be changed by a factor of 6 to accommodate varying engine thrust levels.

The assembly also includes a valve driver to select the high or low jet thrust system by switching in the proper pressure regulator.

G and C Electronics Inputs

Canopus sensor
Fine sun sensor
Coarse sun sensor
Pitch, roll and yaw gyros
C and SS signals
Canopus sensor signals
High and low jet thrust signal

G and C Electronics Outputs

12 pitch, roll and yaw jet control signals
Pitch and yaw thrust vector control actuator signals
High and low jet thrust signal

Figure 40. Preliminary Specification — Guidance and Control Electronics

the worst case, it will present only a slight signal to the clutch winding. Since any failure errors will be small in comparison to the maximum possible output signal only two amplifiers are needed driving redundant clutches to provide a completely redundant system. Should a failure occur in one of the amplifiers or clutches, the other amplifier and clutch will provide the necessary torque to the output shaft. The channel select switch allows selection of either power amplifier and motor to control the engine gimbal angle.

Position feedback is obtained from the AC position transducers. The outputs will be demodulated with the transistor switch demodulators which will provide a phase sensitive DC feedback signal with no offset voltage or deadband.

d. Reaction Control Assembly

The reaction control assembly selected for the Voyager spacecraft is a completely redundant stored nitrogen gas system operating at two thrust levels, and incorporates the capability for electrical resistance heating of the gas immediately upstream of the 12 nozzles. The system is illustrated schematically in Figure 42.

In the normal mode of operation, the reaction control assembly produces torques in pure couples for both low and high level thrust, 0.2 and 3 pounds respectively. The two redundant tankage and feed systems each supply two identical sets of six nozzles, i.e. each set consists of one each plus and minus pitch, plus and minus yaw, and plus and minus roll nozzles. In this way each couple is supplied half from one system and half from the other. The valves, nozzles, heaters, and pressure switches are capable of operating at both the high and low thrust levels.

As sketched in Figure 41 each thruster assembly consists of a heat exchanger portion which operates on resistance heating, and the two nozzles. As shown, two gas feed lines enter the thruster along with a two-wire heater lead connected to the heater element.

The thrust level is controlled by operation of the regulator output selection valves in parallel redundancy. If a selector valve fails to open, the parallel valve can maintain normal operation. Series valves are not considered necessary, since it is intended that all valves incorporate redundant seats to protect against a failure to close. Should a nozzle feed valve fail to close, its supply tank will be depleted along with one third of the redundant system's tank. The disturbance torque from the failed valve will cause the opposite direction valve couple to actuate. Since one of the valves of the couple is in the failing system, together with the failed valve, gas will exhaust three times as fast as in the redundant system. This is the basis for loading three times the required gas, one and a half times in each tank. For a valve failed open at the start of the mission, sufficient gas will remain to complete the entire mission. Following such a failure, operation in the failed axis will be limited to one nozzle rather than pure couples.

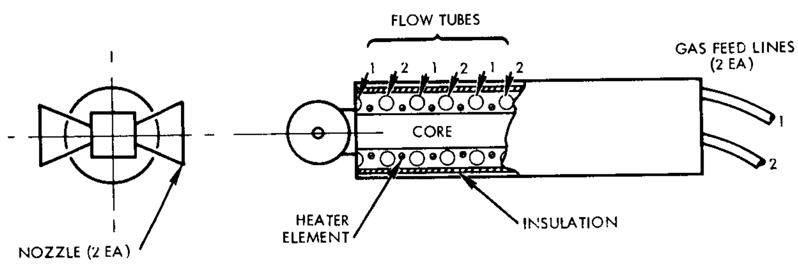


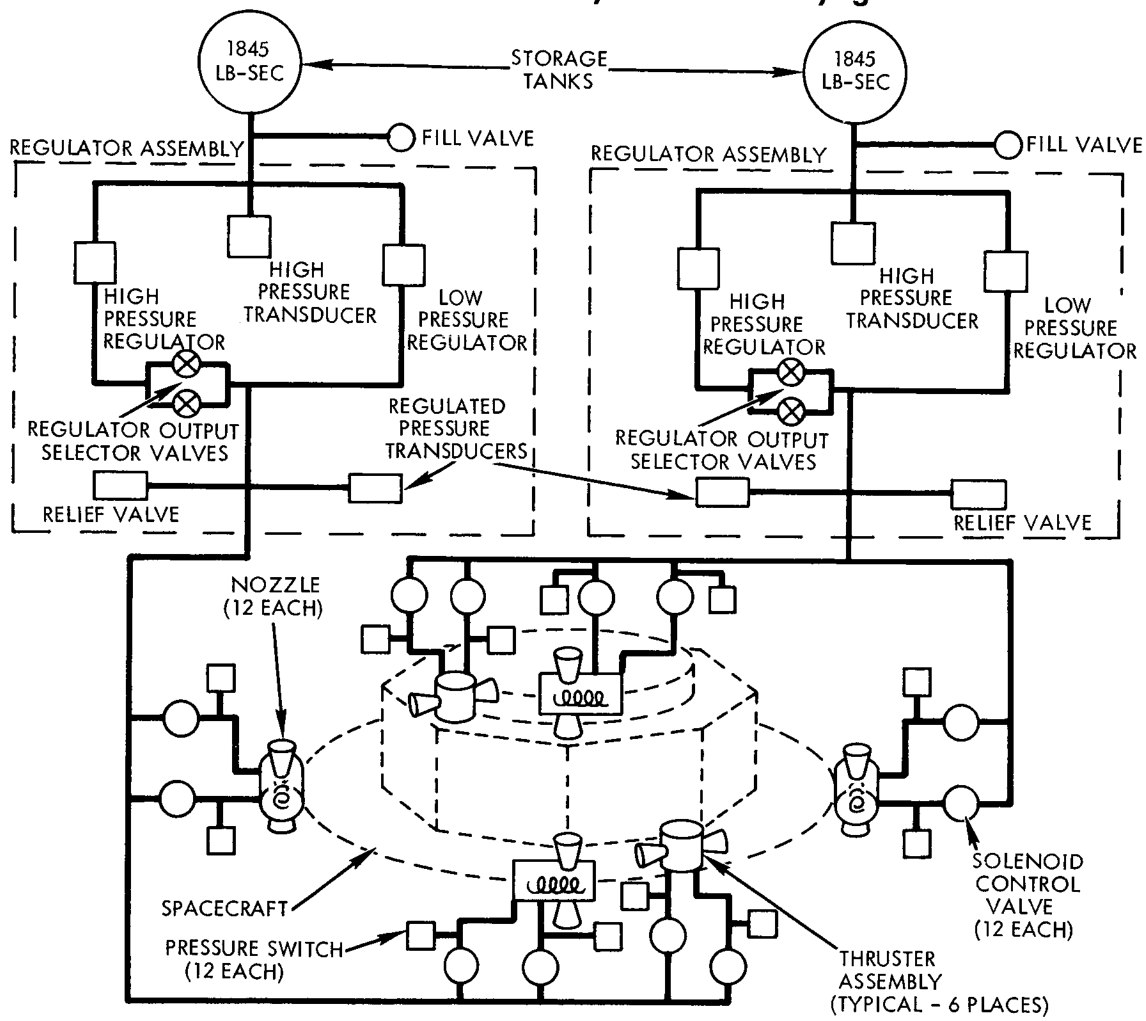
Figure 41. Heater Thruster Assembly

The line length between the nozzle feed valves and the thruster assemblies will be kept to a minimum since for short pulses long lines result in metering the gas flow at the control valves rather than at the thruster assemblies. This would not allow for the proper flow control necessary to obtaining the higher impulse available in the heated gas system. If detailed analysis shows that the temperatures of the valves can become too low, foil-type low powered heaters (3 watts each) will be provided to maintain valve temperatures above -50° F.

Resistance heating of the nitrogen in the thruster results in a total system weight savings of about 24 pounds over a cold nitrogen system. This weight savings is effected by an increase in specific impulse by about a factor of two during the cruise mode. The thruster assembly includes a resistance element with which to convert electrical energy into heat; a heat exchanger in which the gaseous nitrogen propellant is heated; and nozzles through which the heated propellant is expanded. There are

PRELIMINARY SPECIFICATION

Reaction Control Assembly Schematic (Voyager)



Performance Characteristics

Low level thrust:	0.2 pound
Low level regulated pressure:	39 psia
High level thrust:	3 pounds
High level regulated pressure:	630 psia
Specific impulse:	
Minimum:	60 seconds
Maximum:	120 seconds
Total impulse capability:	3690 pound/second
Minimum impulse bit:	5×10^{-4} pound/second
Total heater power requirements: (used when power is available)	128 watts
Nozzle chamber pressure:	
Low thrust:	28 psia
High thrust:	420 psia

Physical Characteristics

Propellant:	Electro-resistively heated nitrogen (1500°F)
Total subsystem weight:	140 pounds
Gas weight:	49 pounds
Initial storage pressure:	3000 psia
Solenoid valve power requirements:	20 watts each peak
Valve heater power requirements:	3 watts each

Figure 42. Preliminary Specification — Reaction Control Assembly Schematic (Voyager)

two gas feed tubes, corresponding to the two nozzles per assembly, entering the heater. These tubes have an inside diameter of 0.120 inch and a wall thickness of 0.013 inch and are of 304 stainless steel. The two-wire heater lead entering the heater connects to the Balco heater element. The core, around which the heater element and gas feed tubes are wound, is a copper slug 0.5 inch in diameter and 2 inches long. The tubes and heater wire comprise the heat exchanger and are imbedded in insulation.

The solenoid valves and pressure regulators will employ the redundant seat design technique. The use of redundant seats has been shown to keep leakage negligible, in addition to providing the added reliability of continued operation in the presence of one valve seat (open) failure. Redundant seating adds little to component weight and is presently being used in other spacecraft developed at TRW.

e. Actuator

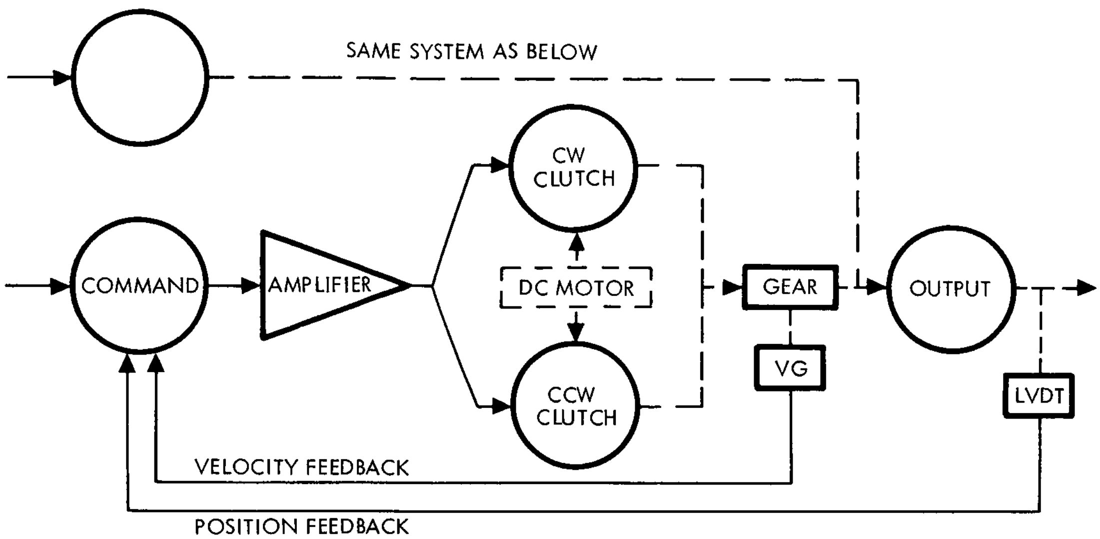
The gimbal actuators are electromechanical clutch actuators similar to those used in the Apollo service module engine system, but scaled down to meet the requirements of this application. A detailed discussion is given in Section 2.6.1. Counter-rotating magnetic particle clutches are driven by DC motors which transmit power through precision gear trains to a high efficiency ball screw. The ball screw converts the rotary motion of the motors and clutches to the linear motion of the output shaft. A preliminary specification and a block diagram of the actuators are provided in Figure 43.

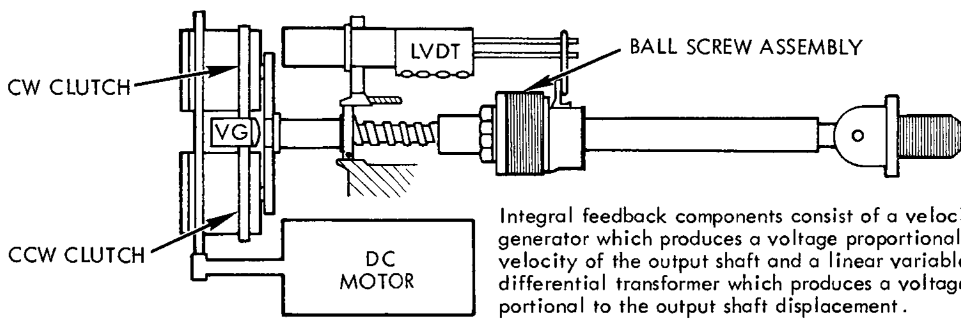
Shaft extension or retraction is commanded by excitation of the appropriate clutch. The actuator is a hermetically sealed unit and possesses a parallel power train system to the ball bearing jack screw, thus enabling the use of redundant actuators.

The prime movers for the drive mechanisms are intermittent-duty, 29-volt DC motors. The motors will be designed to operate on a duty cycle corresponding to the worst case velocity correction of the spacecraft. The prime considerations in the design and fabrication of the motors will be reliability and magnetic cleanliness.

PRELIMINARY SPECIFICATION

Thrust Vector Control Actuators





Integral feedback components consist of a velocity generator which produces a voltage proportional to the velocity of the output shaft and a linear variable differential transformer which produces a voltage proportional to the output shaft displacement.

Function

Control direction of engine thrust in pitch and yaw during trajectory correction, orbit insertion and orbit trim maneuvers in response to signals from the gyro reference assembly

Characteristics

Gimbal rate	10 degree/second
Gimbal acceleration	50 degree/second^2
Gimbal travel	±6 degree
Input power	400 watts minute 800 watts average 1300 watts peak
Weight	36 pound
Total unit midstroke length	16 inches
Control power	1.25 watts maximum

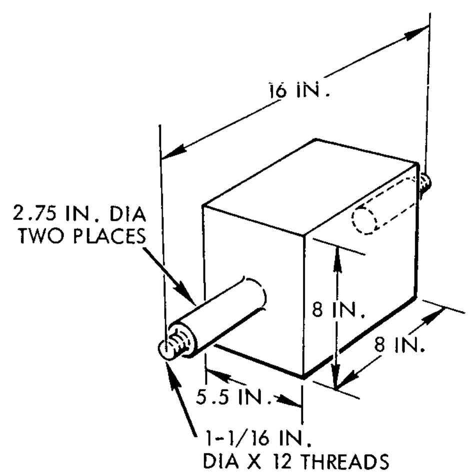


Figure 43. Preliminary Specification — Thrust Vector Control Actuators

The clutches are normally disengaged, except for a bias to reduce backlash, providing no torque output to the ball screw. Application of a control current to the appropriate clutch couples the motor to the ball screw in either a clockwise or counterclockwise direction, the amount of torque being proportional to the control current applied. The control power required to command full output torque is 1.25 watts per clutch.

The main body of the actuator will be closed in a hermetically sealed container. The hermetic seal will be extended to enclose the output shaft by using nested ripple welded bellows. Swivel joints at either end of the shaft will be enclosed within bellows.

Four linear variable differential transformers attached to the output shaft will provide redundant position feedback for closed loop operation of the actuator.

f. Medium and High-Gain Antenna Drives

The high gain antenna drives consist of gimbal assemblies operated by sealed "wabble" gear drives, digital shaft encoders, and non-contacting coaxial joints which conduct the RF signals through rotating gimbal members. A single-axis gimbal drive is used for the medium-gain antenna, and a two-axis gimbal drive in a single unit is used for the high-gain antenna. (See Section 2.6.3.) Preliminary specifications for the antenna drives are presented in Figures 47 and 48.

Figure 44 shows the basic gimbal elements. The yoke is equipped with a pad for mounting the gimbal assembly to the spacecraft and contains the bearings which support the hinge axis gimbal trunnions. For the medium-gain antenna drive, the antenna is mounted directly to the hinge axis gimbal. The high-gain antenna drive is provided with an additional axis of rotation as shown.

The antenna drive electronic circuitry is shown in Figure 45. The desired antenna angle relative to spacecraft coordinates will be received from the C&S and entered into a storage register. The storage register will be compared with the feedback position register in the digital comparator. If the two registers do not compare in every bit, the comparator will feed a signal of proper polarity into the driver gate. The

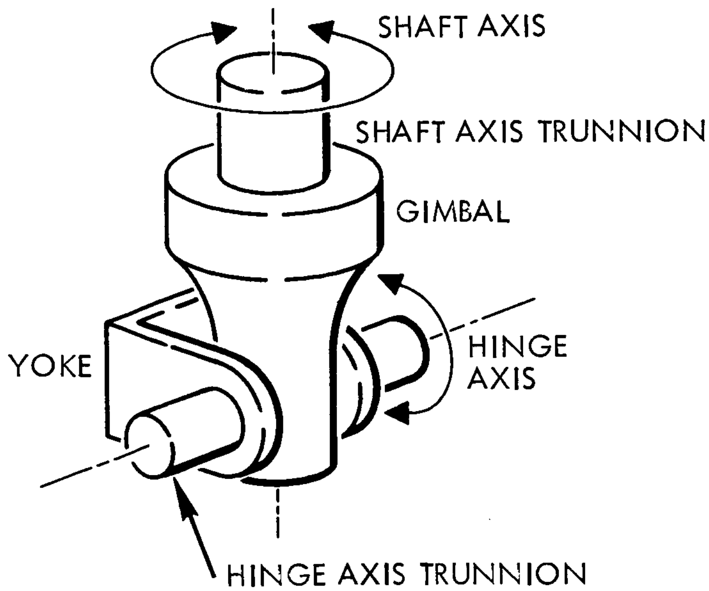


Figure 44. Basic Two Axis Gimbal Elements

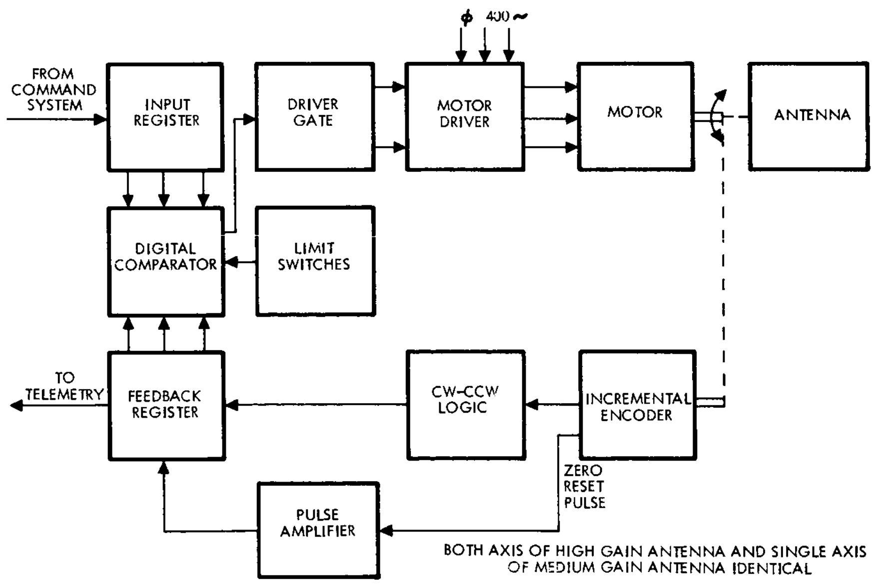


Figure 45. Antenna Drive Implementation Block Diagram

gate will then turn on the motor drivers which will apply full power to the antenna drive motor. The polarity of the input to the driver gate will dictate the direction of rotation of the motor.

Feedback information is obtained from the incremental encoder mounted to the antenna shaft. The output pulses from the encoder are entered into the feedback register, which will constantly accumulate the absolute antenna angle. This information is then used for comparison into the digital comparator and for telemetry.

A zero reset capability is provided by the encoder. When the encoder passes through null, a pulse is emitted which is amplified and fed into the reset input of the register to assure a zero feedback indication.

The contribution to antenna pointing error by the electronics drive will be less than 0.1°. Since the least significant bit of the register used will correspond to 0.1° error, any error this big will cause the antenna to be driven until the two registers are identical, at which time power to the motor is removed. The antenna will then stop within 0.05° from the time the power was removed.

The drives for the gimbal are sealed wabble gear units similar to those developed for the OGO solar array (Figure 46) and OPEP drives. During the OGO program these drives were subjected to thorough environmental and performance testing and operated more than 10,000 hours in vacuum. OGO I was launched September 5, 1964, and the drives are still operating when commanded.

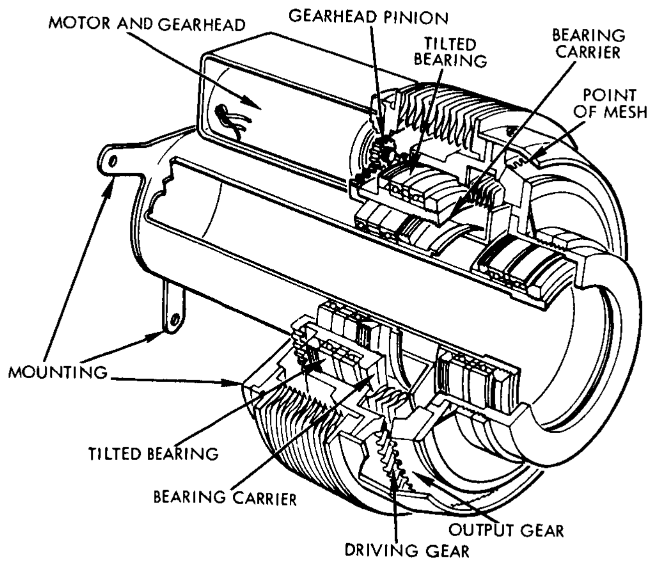
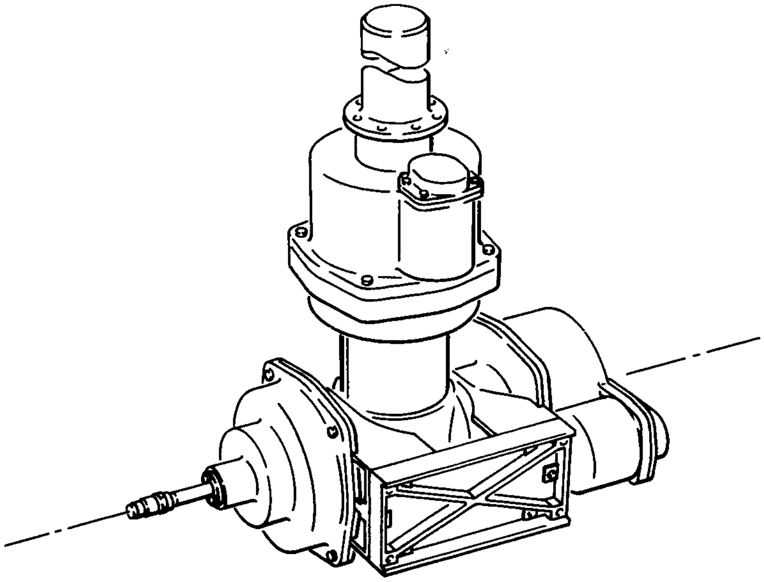


Figure 46. Sealed OGO Drive Mechanism

PRELIMINARY SPECIFICATION

High-Gain Antenna Two Axis Gimbal



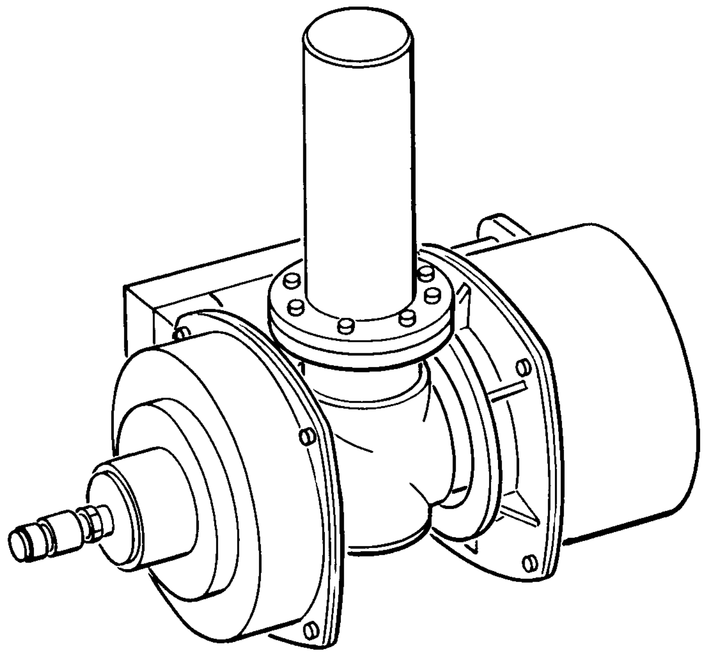
Performance Characteristics

Gimbal travel	
Hinge axis:	±90° plus stowage
Shaft axis:	360° unrestricted
Gimbal slew rates	
Slew:	5.3 mr per second/minute each axis
Angular acceleration:	0.5 mr per second^2
Output torque (maximum)	
Hinge axis:	8000 inch/pound
Shaft axis:	1100 inch/pound
Input voltages	
Drives:	50 volt, 2ϕ400 cps square wave
Heaters:	28vdc
Power	
Hinge axis drive:	3 watts average, 26 watts peak
Shaft axis drive:	8 watts average, 7 watts peak
Heaters:	5 watts
Gimbal angle encoder:	To indicate gimbal angles within 0.10 degrees
Number of gimbal	
Axes:	Two
R-F path:	Two noncontacting rotary joints
Gimbal drives	
Seals:	High speed elements sealed in pressurized inert atmosphere
Stalled conditions:	Drives to withstand stalled conditions without damage
Magnetic fields:	Nonmagnetic materials to be used where possible
Weight:	30 pound maximum

Figure 47. Preliminary Specification—High Gain Antenna Two Axis Gimbal

PRELIMINARY SPECIFICATION

Medium-Gain Antenna
Single-Axis Gimbal Drive



Performance Characteristics

Gimbal travel:	±90° stowage
Gimbal slow rates:	0.5 mr per second^2
Angular acceleration:	0.5 mr per second^2
Output torque (maximum):	1100 inch pound
Input voltage	
Drive:	50 vac, 2Φ400 cps square wave
Heater:	29 vdc
Power	
Drive:	7 watts maximum
Heaters:	5 watts
Gimbal angle encoder:	To indicate gimbal angles within 0.10 degrees
Number of gimbal axes:	One
R-F path:	One noncontacting rotary joint
Gimbal drives	
Seals:	High speed elements sealed in pressurized inert atmosphere drive to withstand stalled conditions without damage
Magnetic materials:	Nonmagnetic materials to be used where possible
Weight:	17 pound maximum

Figure 48. Preliminary Specification—Medium-Gain Antenna Single-Axis Gimbal Drive

All rolling and sliding surfaces of the motor, gearhead, associated gears and ball bearings are hermetically sealed in pressurized inert gas except for one pair of gears and one ball bearing pair specially designed to operate at low speed in vacuum. For the Voyager antenna drive, this bearing pair is replaced by gimbal bearings. All exposed slow-moving elements are plated with low-shear precious metal and impregnated with molydisulphide. All high-speed elements employ radiation resistant lubricants and operate in a sealed pressurized inert atmosphere.

Sealing is accomplished by two bellows installed between the unmoving parts of the mechanism and the driving gear. The unique feature of the drive mechanism is the use of a pair of specially-cut wabble gears for the output stage. Action of the bearing carrier and a tilted bearing internal to the unit produces a non-rotating conical nutation (or wabble) of the driving gear at the end of the main bellows. This motion causes rotation of the output gear and shaft by sequential engagement of a limited number of gear teeth. The prime movers for the drive mechanisms will be two-phase, 400 cps, servomotors.

Two gimbal position pick-offs will be required for determining the position of the antenna about each axis. Antenna position is measured by means of a shaft encoder which is a magnetic, incremental shaft encoder using a variable reluctance transducing technique. The input transducer is essentially a segmented disc mounted on the output shaft which, in conjunction with two differential pick-offs, provides a periodically variable reluctance path as a function of shaft position. This effect serves to modulate an interrogation carrier correspondingly, thus providing two 90-degree out-of-phase modulated carrier signals. Electronics convert these into a series of pulses which are also a function of angular position. Each pulse represents a binary fraction of a circle. By operating logically on the two signals in order to distinguish CW and CCW pulses, and accumulating them in an external register, net angular travel can be determined. A zero reference involving an additional pick-off and magnetic reference point is included.

To provide an RF path through the rotating members of the gimbals, RF rotary joints and an interconnecting 90-degree RF elbow fitting are required. The joints and the elbow fitting will be contained within the gimbal structure; and the non-contacting rotating elements of the joint will be mounted such that no additional bearings other than the gimbal bearings will be required to maintain concentricities.

A small cable wrap-up assembly is required on the hinge axis of the high-gain antenna drive to accommodate wires from the shaft axis drive and position pick-off. This assembly consists of wires grouped together in a flat ribbon which are wound around the hinge axis trunnion and shielded from direct exposure to space. Motion about the hinge axis is limited to prevent damage to the antenna, drives, and spacecraft. A set of limit switches interrupt the drive motor circuit at appropriate hinge axis positions.

To minimize stiffening of the cable wrapup and thickening of the sealed drive lubricants at extreme low temperatures, the antenna drive employs thermostatically controlled heaters and thermal insulation. In addition, the drive is thermally isolated from the spacecraft.

g. Terminator and Limb Detector

The function of the terminator and limb crossing sensors is to sense the crossings of the terminator plane and the limb plane (the plane tangent to Mars' surface at the point on the planet closest to the sun) while the spacecraft is in an orbit about Mars. (See Figure 49).

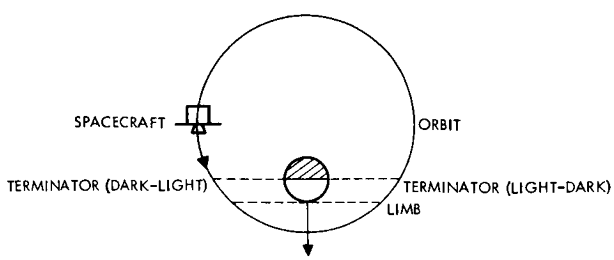


Figure 49. Relationship of Spacecraft, MARS Limb & Terminator Crossings

A pair of identical sensors are used to detect the limb and terminator crossings. One sensor will detect crossings on the dark-light side of the planet, and the other sensor will detect crossings on the light-dark side. Each sensor is mounted with its axis at 90 degrees cone angle on the spacecraft. Their orientation in clock angle on the spacecraft are determined by the expected range of clock angles for the limb and terminator of Mars. For the sample mission they are located at clock angles of 105 and 274 degrees respectively.

Each sensor consists of a cadmium sulfide photocell mounted in the focal plane of a 1.5-inch diameter f/1.5 lens. The photocell will be specially constructed in the form of a long narrow grid on a flat substrate measuring about 0.08 by 5.4 inches. The resulting field of view will be approximately 2 by 90 deg. The speed of the lens and the size of the photocell are designed to:

- provide a sensor field of view which brackets the range of clock angles of Mars' limb and terminator during the mission
- provide adequate sensitivity to produce a detectable signal at the crossing points.

Each sensor has an active detector and a "dummy" detector which is shielded from light. The two detectors are connected in a bridge, providing compensation for changes in the temperature of the unit. Thermal design of the unit keeps the average temperature of the unit about 0°F. This operating temperature provides high light sensitivity and relatively low change in sensitivity with temperature.

The field of view of each sensor is oriented such that it is normal to the spacecraft roll axis. The slit width is chosen to be such that when one-half of the slit is on each side of the terminator plane, a detectable signal is produced by the photocell. It is then found that when the limb plane is approximately half-way across the slit the same signal level is achieved. The slit width and the cone angle of the slit center may be varied by small amounts, if necessary, to produce detection thresholds at the true limb and terminator crossing points.

The accuracy of the sensors will be affected by the following items:

a) Uncertainties in the Mars orbit

b) Uncertainties of Mars' surface and albedo

c) Changing orbital parameters during the mission

d) Detector drift and noise

e) Threshold level drift

f) Non-homogeneity of the photocell.

The over-all accuracy which results from the effects of these items is estimated to be ±0.7 deg. neglecting attitude uncertainty. The type of signal will be a discrete binary signal which indicates whether the sensor field of view intersects or does not intersect the illuminated portion of Mars. Figure 50 presents a preliminary specification for the limb crossing sensors.

h. Accelerometer

A preliminary specification for the accelerometer is given in Figure 51. The accelerometers being considered are the Bell Model VII, the Honeywell GG177B, and the Systron Donner 4310, all three of which utilize a minimum number of moving parts (a flexure supported proof mass), as compared to accelerometers with motors, bearings, and pumps.

2.5 Parameters and Performance Summary

2.5.1 Performance Parameters

The Guidance and Control Subsystem performance is compared with the subsystem performance requirements (Section 2.2) in Table 22.

PRELIMINARY SPECIFICATION

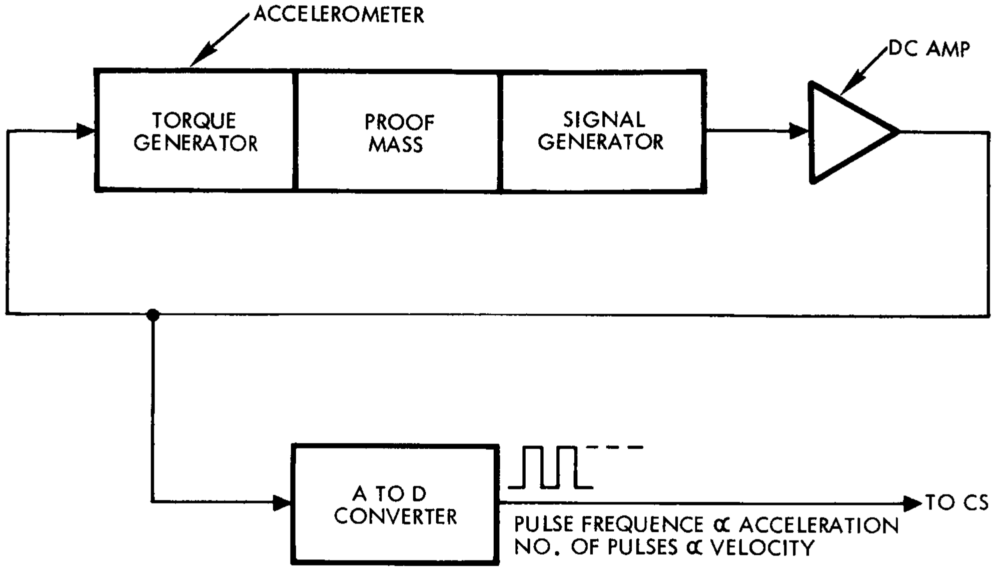
Limb and Terminator Crossing Detector

Number:	2 identical sensors are required; one to sense "morning crossings", the other to sense evening crossings"
Type:	Cadmium sulfide photocell detector with objective lens and signal amplifier
Size:	1.75 inch diameter at lens end by 4 inch long, by 1.75 x 6 inches at "base" end
Weight:	0.6 pound
Power:	200 milliwatts per sensor
Components (per sensor):	2 CdS photocells 1 1.5 inch diameter wide-angle f/1.5 lens 8 1/8 watt metal film resistors 2 precision wire-wound resistors, 1/8 watt 2 uA702A integrated circuits 2 silicon transistors 2 ceramic capacitors 1 tantalum capacitor
Field of view:	100 degrees by 2 degrees
Accuracy:	±0.7 degrees for each crossing

Figure 50. Limb and Terminator Crossing Detector

PRELIMINARY SPECIFICATION

Accelerometer



Function

Provide spacecraft thrust axis velocity information to the computing and sequencing and subsystem. This information is used to turn off the engine for velocity corrections and injection into mars orbit

Accelerometer Description

Weight	1.0 pound
Size	Accelerometer and electronics: 1.75 diameter x 1.75 inch
	Pulse rate converter: 0.5 x 2.5 x 1.375 inch
Power	3 watts
Range	±2.3 g's
Pulse Quantization	0.015 ft/(second-pulse)
Maximum Pulse Rate	5000 pulses/second
Accuracy	≤ ±0.1 percent

Figure 51. Preliminary Specification — Accelerometer

Table 22. Comparison of Subsystem Requirements of Guidance and Control Subsystem Performance

Mode	Requirement	G&CS Performance
Initial Attitude Hold	Maintain separation attitude ±10 deg	For a maximum initial rate of 1.5 deg/sec, separation attitude is maintained within ±8.5 deg
Cruise Mode	Maintain pointing accuracy of ±0.5 deg, 3σ per control axis	±0.5 deg, 3σ, per control axis
Trajectory Correction	Thrust vector pointing accuracy of ±0.76 deg, 3σ, per control axis	±0.8 deg, 3σ, per control axis
Flight Spacecraft - Flight Capsule Separation	Maintain pointing accuracy of ±0.75 deg, 3σ, per control axis	±0.71 deg, 3σ, per control axis
PSP Pointing	a. Maintain pointing accuracy of ±0.5 deg, 3σ, per control axis b. Maintain angular rate less than 10 deg/hr during photographic operations	a. ±0.49 deg, 3σ, per control axis b. 2 deg/hr

Estimated weight of the subsystem is presented in Table 23.

Estimates of gas consumption given in Table 24 are based on a cold gas nitrogen system with a specific impulse of 60 seconds. The thrust level of the reaction jet system is 0.2 pound during cruise and 3 pounds during all other Voyager mission phases. Table 24 includes the gas required to nullify the effects of solar radiation torque, gravity forces, and PSP reaction forces. The initial attitude hold gas consumption was computed assuming an initial rate of 1.5 degrees per second. The sun-Canopus acquisition mode and the inertial and maneuver mode gas requirements were based on 6 operations before capsule separation and 4 operations after capsule separation.

Table 23. Guidance and Control Estimated Weights

Gyro Reference Assembly	10.0
Accelerometer	1.0
G and C Electronics	13.0
Canopus Sensor (2)	12.0
Fine Sun Sensor Coarse Sun Sensor (4)	1.0
Earth Detector	0.25
Reaction Control Ass'y	140.0
High Gain Antenna Drive	32.0
Medium Gain Antenna Drive	17.0
TVC Actuators (2)	36.0
Limb and Terminator Crossing Detector (2)	1.2
Antenna Drive Electronics	5.0
TOTAL	268.5 lb

Table 24. Voyager Reaction Control System Gas Requirements

Mode	Gas Consumption (lb)
1. Initial Attitude Hold	1.5
2. Cruise Mode:	
a. Mars Transit	2.0
b. Orbit	7.0
3. Sun - Canopus Acquisition (10 times)	5.3
4. Inertial and Maneuver Mode	4.2
5. Capsule Separation	0.5
Total	20.5

2.5.2 Reliability Assessment

The guidance and control subsystem reliability block diagram illustrating some of the redundant and backup components is shown in Figure 52. In addition to the redundancies shown, several of the components have circuit level redundancy, in particular the control electronics and the thrust vector control drives. See Section 2.4.4 for a discussion of these redundancies. In general, the redundancies follow the same approach that was used during Task A, but additional studies, especially with respect to redundant gyros, are required before the system configuration can become firm. The reliability assessments in Figure 52 are based on a mission time corresponding to encounter plus one month and the environmental factors indicated in Table 25. The indicated reliability for the system as configured in Figure 52 is 0.93.

Table 25. Mission Phases, Time, and Environmental Factors (k)

Mission Phase	Description	Time (hours)	k Factor
1	Liftoff, boost and injection into interplanetary trajectory	0.3	1000
2	Reference acquisition, cruise and midcourse corrections:		
	Reference acquisition and cruise	4280	1
	Midcourse correction	0.125	50
3	Deboost	0.109	100
4	Orbit trim and Mars cruise:		
	Orbit trim	0.026	50
	Cruise	723	1

2.5.3 Error Summaries

a. Sources

Pointing requirements for the Voyager mission involve interaction among the guidance and control subsystem, the high-gain antenna, the PSP, and the vehicle structure. The sources of error include attitude reference errors, guidance and control subsystem limit cycle errors, the articulated component drive errors, alignment errors, and thrust vector

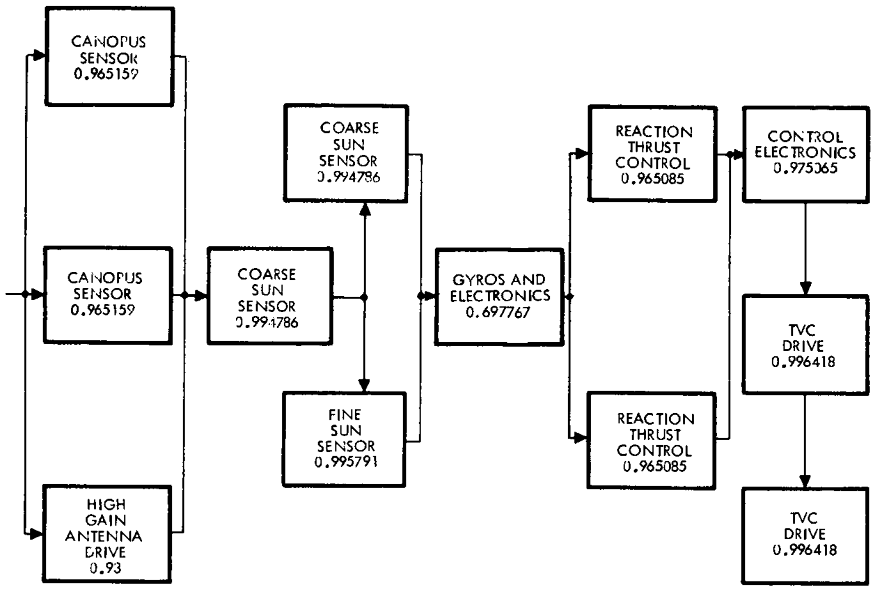


Figure 52. Guidance and Control Subsystem Reliability Diagram

pointing errors caused by thrust misalignment and center of mass offset. To minimize the alignment errors, the gyro assembly, Canopus sensors, fine sun sensor, and accelerometer are mounted on a common fixture attached to primary structure.

b. High Gain Antenna Pointing

The high gain antenna must be pointed to within ±1 degree to limit the pointing loss to 1 db, with 99 percent probability. This pointing requirement, reflected into a single-axis error, results in an allowable error of 0.7 degree (3σ). A preliminary 3σ error budget is given below.

Attitude reference accuracy	±0.1 deg.
Limit cycle error	±0.499 deg.
Antenna drive quantization	±0.3 deg.
Drive electronics error	±0.1 deg.
Antenna drive pickoff error	±0.15 deg.
Antenna boresight error	±0.2 deg.
Alignment (to spacecraft reference)	±0.25 deg.

Neglecting the fact that limit cycle and quantization errors are not gaussian, root-sum-squaring the errors leads to a value of 0.692 deg.

c. Midcourse Velocity Corrections

The thrust vector pointing accuracy requirement during midcourse corrections is ±0.76 deg, 3σ, per axis. The single axis pointing error budget is shown below for this case.

Attitude reference accuracy	±0.1 deg.
Gyro torquing error	±0.2 deg.
Gyro alignment error	±0.1 deg.
Gyro drift error (0.4°/hr. for 1 hr.)	±0.4 deg.
Limit cycle error (at gyro uncage)	±0.25 deg.
TVC limit cycle error	±0.2 deg.
Thrust vector pointing error from center of mass uncertainty (0.5 inch)	±0.56 deg.
Control error from thrust misalignment	±0.04 deg.
RSS	0.8 deg.

The RSS total of 0.8 degree, 3σ, (again neglecting the impropriety of calling the RSS "3σ") does not quite meet the thrust vector pointing requirement.

The largest contribution to the pointing error results from the 0.5-inch uncertainty in the location of the center of mass. Reducing the uncertainty would result in an acceptable 3σ pointing error of 0.76 degree. Further improvement in the pointing accuracy can be achieved at the expense of TVC complexity. One technique would be to add error integral compensation of the attitude gyro signal, which would eliminate the attitude errors required to point the thrust vector through the spacecraft center of mass. Although this mechanization would significantly improve the pointing accuracy during deboost when the attitude error control gain is unity, only a small improvement would be obtained during low thrust operation (midcourse corrections and orbit trim), when the control gain is increased by a factor of six. Another method of improving the pointing accuracy would be to sample the TVC actuator position and increment the attitude reference (gyro) by the measured TVC actuator deflection. For steady state conditions, and using the error integrator, this scheme would essentially eliminate the thrust vector pointing error from center of mass uncertainty. The requirement for 0.76 degree pointing accuracy is still being evaluated. If this requirement becomes firm, methods of meeting it will be studied in more detail.

d. Orbital Injection and Orbital Trim

Because of the lower attitude control gain during deboost, the error caused by center of mass offsets are larger at this time than during midcourse corrections. The thrust vector pointing error is increased to ±1.15 degree, 3σ, per control axis at the start of this event and reduces to ±0.86 degree at termination. The error during orbit trim is also greater than that of midcourse correction. Although the attitude control gain during orbit trim maneuvers is at the high value, the control moment arm is reduced after capsule separation, and this increases the thrust vector pointing error to ±1.13 degrees, 3σ, per control axis. The error for the orbit trim with the capsule is ±0.68 degree, 3σ, each axis.

e. Capsule Separation

For the reorientation maneuver prior to capsule separation, the error sources are shown below.

Attitude reference accuracy	±0.1 deg.
Gyro torquing error	±0.2 deg.
Gyro alignment error	±0.1 deg.
Gyro drift error	±0.4 deg.
Limit cycle errors	±0.25 deg.
Gyro uncage capsule release	±0.25 deg.
Capsule to spacecraft alignment	±0.4 deg.

As before, the pointing accuracy requirement of ±0.75 degree (3σ) is met by reducing the control deadband to ±0.25 degree. The RSS total computed from the above data is ±0.71 degree.

f. PSP Pointing

The PSP, Mode 1, pointing requirement of ±0.5 degree, 3σ imposes a stringent constraint on the guidance and control subsystem. For this case, the primary error sources are:

Attitude reference accuracy	±0.1 deg.
Limit cycle error	±0.25 deg.
PSP drive accuracy	±0.25 deg.
Camera boresight error	±0.05 deg.
PSP alignment to spacecraft	±0.25 deg.

The RSS error of ±0.49 degree, 3σ, is within the requirement. For high gain antenna pointing, sun-Canopus acquisition and cruise mode operation, a ±0.5 degree limit cycle amplitude is acceptable. For velocity correction maneuvers, capsule separation and PSP pointing, the accuracy requirements can be achieved by decreasing the control deadband from ±0.5 to ±0.25 degree by command from the C&S. To minimize reaction control gas consumption, it is desirable to provide two control deadbands, a wide range for normal cruise and a smaller range for precise pointing.

2.6 Design Alternatives

2.6.1 TVC Implementation

Three different means of actuating the engine for thrust vector control have been investigated. While the results do not indicate an obvious choice and further study is warranted, a magnetic particle clutch actuator was chosen primarily because a unit for a similar application is being developed for the Apollo service module, and it offers a reliability advantage over the other electromechanical actuators considered.

The thrust vector control actuator was sized based on the following requirements:

• Low thrust power:	force	54 lbs
	rate	3.35 in/sec
	acceleration	16.7 in/sec^2
• High thrust power:	force	148 lbs
	rate	0.134 in/sec
• Stroke		4.1 inches

These requirements are based on the actuators correcting the initial trim offset rapidly during the engine low thrust period and operating under a low frequency, low amplitude duty cycle during the high thrust period.

The three methods of implementation of the TVC investigated were the magnetic particle clutch actuator, a DC torque motor-driven actuator, and a hydraulic actuator. A comparison of the various parameters affecting the final actuator selection is shown in Table 26, and discussed in the paragraphs which follow.

a. Magnetic Particle Clutch Actuator

The magnetic clutch actuator features a parallel power train and redundancies in essentially all elements except the ball screw assembly, thus enabling a high reliability assessment. All of the moving parts of the actuator including the motors are enclosed within a hermetic seal, permitting the use of normal lubricating techniques. Since the actuator output is controlled by the clutches, the control power required is small, and simple electronics can be used.

Since the DC motors will be either shunt wound or series wound, no permanent magnet materials will be employed in the motor. Selection of the rotor and field materials of these motors will involve the consideration of low retentivity to minimize the non-operating residual magnetic field of the motors. Shielding may be required if the magnetic field of the motors when they are not operating is not otherwise satisfactory and if the magnetic field specifications must be maintained during velocity corrections.

The friction in the power train of this actuator will maintain the engine in a fixed position in the presence of relatively large gimbal torques even when the actuator is not energized.

In order to provide a common basis for comparison, the reliabilities of the three thrust vector control systems considered were computed based on failure rate data from vendors where available, data based on similar components, and on the ground rules used on the LEMDE program for K factors as shown below. The times used were those for Voyager.

Table 26. Thrust Vector Control Actuation Trade-Offs

System	Weight (lb)	Power (watts)	Reliability	State of Development	Comments	Advantages	Disadvantages
Magnetic Particle Clutch Actuator	36 Size 2 pkgs. 5.5x8.5x8 with 2.5" dia. ext. Total unit midstroke length 16"	400 STBY 800 NORM. 1300 PEAK 0.7 w/coil control power	.99827 for Actuator and electronics based on LEMDE reliability rules and Voyager times.	Similar unit developed for Apollo service module thrust vector control actuators.	Actuator length must be reduced from 22" to 16" to fit Voyager application. Actuator stroke must be increased from 3.5" to 4.1" to fit Voyager application. Actuator motor and clutches will be reduced to improve power and weight efficiency at the reduced requirements for Voyager	1. Control power is small (~1.4 watts) simplifying electronics 2. Holding capability when deenergized 3. Development status for space applications.	1. Relatively large power requirements 2. Magnetic properties
D.C. Torque Motor Driven Ball Bearing Jackscrew	30 Size 2 pkgs. 5.5" dia. x 6 with 2.5" dia. ext. Total unit midstroke length 16"	10 STBY 250 NORM. 450 PEAK (including elect.)	.9799 for Actuator and electronics based on LEMDE reliability rules and Voyager times. (Actuator alone .99971)	Similar unit developed for LEMDE throttle actuator	Actuator motors must be larger to accommodate higher speed requirements. Actuator stroke must be increased from 0.7" to 4.1" to fit Voyager application.	Development status for space applications.	1. Electronics must control motor power. 2. D.C. torque motors have large permanent magnetic fields 3. If the load requirements increase appreciably this type of actuator becomes impractical 4. Little holding power when deenergized
Hydraulic Actuators	27	400 STBY 600 NORM. 800 PEAK	.99827 for Actuator and electronics.	Motor pump developed for Minuteman Wing VI Stage II		1. Control power is small 2. Simple electronics 3. Large torques easily obtained.	1. Long term space storage uncertainty 2. Little holding power when deenergized 3. Development status for space application.

K Factor Table

Mission Phase	Electronic K	Mechanical K'
Launch and boost	10	0.1
Cruise	1	0.1
Velocity corrections	2000	200

Using the above K factors and the estimated failure rates of the mechanical and electrical components, and computing redundancy where applicable, the reliability of the clutch actuator system is computed to be 0.99827. Several factors led to the choice of a scaled down version of the Apollo service module actuator rather than using the actuator unmodified. These factors are listed in the following table.

Item	Apollo	Voyager
Stroke, in.	3.5	4.1
Overall length, in.	22	16
Output force, lb	650	225
Actuator weight (each actuator), lb	30	18

Scaling down the output force capability of the actuator permits the choice of smaller clutches and smaller motors operating in a more efficient region. The Voyager loads on the Apollo service module actuator result in maximum motor operating efficiencies of about 43 percent; however, the proper choice of motor could result in motor operating efficiencies of between 65 and 70 percent. Thus, a considerable savings in power and weight can be achieved by scaling down the Apollo actuator.

b. DC Torque Motor Actuator

A redundant torque motor-driven ball screw actuator similar to the LEMDE throttle actuator was investigated for the Voyager TVC implementation. This actuator employs three redundant DC torque motors directly driving a ball screw actuator hermetically sealed around the output shaft by means of a bellows assembly. The brushes and commutators of the DC torque motors are protected within the controlled atmosphere inside the seal. A redesign of the LEMDE actuator would be required to accommodate the differences in requirements for the Voyager application as shown in the table below.

Item	LEMDE	Voyager
Stroke, in.	0.7	4.1
Rate, in/sec	0.875	3.35

Two factors reduce the operating power requirement of this actuator over that of the magnetic particle clutch actuator. The motors of this actuator drive the efficient ball screw actuator directly, while the motors of the Apollo type actuator drive the ball screw actuator through magnetic particle clutches and an additional gear train. The torque-speed capabilities of the DC torque motors more closely match the load requirements at both thrust levels than do the torque-speed curves of the clutch actuator motors. Consequently at low speed, high torque conditions much power is lost in slippage of the clutches.

The torque motors required to provide the TVC power requirements have a 1.5 lb-ft stall torque, are 5.15 inches in diameter and weigh 1.5 pounds, an increase over the existing torque motors from 0.52 lb-ft torque, 3.625 inches diameter, and 1.1 pounds. The overall size of the actuator would become 5.5 inches diameter by 6 inches long, with 2.75-inch diameter extensions at either end. The method of mounting can readily be made identical to the method of mounting the magnetic particle clutch actuator. The swivel joints at either end would be enclosed within the hermetic seal by means of bellows.

This actuator has several disadvantages compared to the clutch type actuator. The large magnetic field associated with DC torque motors will be difficult to control to the requirements of Voyager. The control amplifiers for this actuator are relatively high powered, increasing their complexity. If the actuator load or performance requirements are increased appreciably in the future, the size of torque motors required to handle the requirements may increase beyond practical limits.

Because of the high efficiency of the ball screw, this actuator can be easily backdriven. Therefore, when the actuator is not excited, the engine will be relatively free to move within the gimbal limits.

The reliability of this actuator system was computed, based on using the exact LEMDE system, to be 0.9799.

c. Hydraulic Actuators

Hydraulic actuators were also considered for the TVC implementation. The actuators and pump were sized to handle the maximum force and rate simultaneously, although they never occur together in practice. Theoretically, considerable power could be conserved by employing an accumulator to handle the initial high rate. Two factors limit the practical reduction of power. The minimum practical operating pressure of presently developed servo valves is 450 psi. The minimum practical leakage flow for two servo actuators is approximately 0.10 gpm. The Minuteman Wing VI Stage II pump comes nearest of any developed pump to meeting the requirements. This pump will supply sufficient flow to meet the maximum force and rate requirements simultaneously. If a hydraulic system were selected as the TVC actuation system, further investigations would be made to find, develop, or modify an existing pump to supply less flow at a corresponding reduction in power. The Minuteman pump provides 0.9 gpm flow. A pump with a flow of 0.4 gpm would provide sufficient flow at a considerable savings in power. A dual system is provided in order to achieve a reliability comparable to the other systems considered. The system, estimated to weigh 27 pounds, would consist of two hydraulic power supplies, 3.5 x 4.5 x 7 inches, and four actuators, 1.5 inches diameter x 16 inches long with a 1.5 x 2 x 2 inch servo valve mounted on the side.

A hydraulic system results in the lightest weight. However, there are no known hydraulic systems being developed for appropriate space applications and thus the hydraulic system would require special development to prevent external leakage. The external leakage could be virtually eliminated by completely enclosing the system within an hermetic seal. Hermetic sealing of the actuators could be accomplished by a double O-ring seal and bellows arrangement. Another disadvantage of this system is that the engine is not rigidly held in position when the actuators are not energized.

The reliability of this actuator system is estimated to be .99827, assuming the drive electronics are identical to the magnetic particle clutch actuator system.

2.6.2 Reaction Control Assembly

Tradeoff factors for the reaction control assembly include weight, reliability, development status and feasibility, and electrical power availability. Weight estimates for the primary implementation choices were based on providing 3,690 lb-sec of total impulse, which is three times the basic mission requirement of 1,230 lb-sec. Gross weight for the recommended system, a heated gaseous nitrogen system described in Section 2.4.4, is estimated at 140 pounds. Its major contender on a weight basis is a monopropellant hydrazine thruster system functionally identical to the recommended system. The gross weight of that system is estimated at 70 pounds.

a. Heated Nitrogen vs. Monopropellants

Recommendation of the heated nitrogen system is based on advantages of development status and simplicity, hence reliability. Stored gas systems for spacecraft attitude control are generally credited with offering maximum reliability because of functional simplicity and proven flight performance. This is the basis for general preference for stored gas systems over liquid monopropellant and bipropellant systems where total impulse requirements are relatively low.

In addition to these broad considerations, there are specific drawbacks to a monopropellant system. Both hydrazine and hydrogen peroxide thrusters have been operated at a thrust level as low as 2 pounds and experimental units have been run at one pound thrust. However, there is little evidence of successful monopropellant thruster operation at a thrust of less than one pound where the engine has been exhausted to a vacuum. The implication here is the need for a significant development program on low-thrust monopropellant engines. Typical problems expected to show up in such a program are:

- Combustion stability, particularly for the low duty cycle, pulse type operation required here,

- Acceptable repeatability, i.e., pulse amplitudes can vary by integral factors,
- Plugging the nozzle throats, whose small size may readily be blocked with particles from the catalyst beds.

In contrast, stored gas systems have demonstrated flight performance at this thrust level and much lower, e.g., 0.02 pound for the Vela satellite reaction control thrusters. As for the implementation of heater-thruster assemblies, similar units have been designed, developed, and incorporated into the velocity correction thruster for the Vela 3 satellite, and have successfully operated in space.

An additional problem with monopropellant thrusters is that they conventionally require close coupling of the valve, decomposition chamber, and nozzle. Since the nozzles will be located at the outer periphery of the spacecraft solar array, the propellant feed lines, as well as the valves, would require heating to maintain the propellant above its freezing temperature (about 35° F). Depressing the freezing temperature is impractical since the expected ambient low temperature is about -180° F. An alternative is to locate valves and decomposition chambers within the temperature controlled cabin, and feed four-foot lines out to the nozzles. This, too, amounts to considerable performance degradation; line cooling would reduce specific impulse to values approaching 100 seconds, which is in the neighborhood of the heated nitrogen average performance.

b. Functional Mechanization

Aside from the type of thruster, stored gas or monopropellant, two system mechanizations in addition to that selected were considered. The first would require separate valves and nozzles for low and high level thrust, giving a total of 24 valves and 24 nozzles. The system would incorporate redundant tankage and feed systems, gas jets would operate in pure couples with the two nozzles in a given couple deriving their propellant from separate tanks. Sufficient propellant would be stored for three times the basic mission requirement to provide for failure of a single solenoid valve in the open position. This system was discarded because of the excessive number of valves, nozzles, pressure switches, and associated circuitry. The selected system, which will utilize half as many of all these components, to be operated at both low and high thrust levels, is believed to be superior with respect to simplicity, reliability, and weight.

A second alternate mechanization would look schematically exactly like the one selected. The difference would lie in the system operation and weight (of gas and tankage). In this system, one half of the components would be used for low thrust, nominally, and the other half for high thrust, nominally. For a given axis, gas jets would fire singly rather than in couples, and each half of the system would be capable of either high or low thrust operation. In this way, the two system halves would be redundant, since either one could perform both high and low thrust operation. This capability would be provided by selection of either high or low pressure regulator output just as in the selected system. The main advantage of this alternate system is that a factor of two, rather than three, times the basic mission requirement for gas could be used to contend with the worst failure mode of a solenoid valve (open). The big problem with this system is the added complexity of switching logic required. In the event of a valve failure, control signals must be rerouted from one set of valves to the other, the regulator output valve requires active control compared to its normally de-energized state, and two-level switching must be provided for the valve drivers during depletion of gas from one tank through a "failed-open" valve. Following gas depletion, the switching levels of the active half of the system must revert to the design values for normal cruise operation. Since these switching requirements are only a part of the added complexity in electronic logic required with this mechanization, the associated degradation in reliability does not appear to be justified by the weight savings, about 36 pounds.

c. Heated vs. Cold Gas

In comparison with a cold gaseous nitrogen system, a weight savings of about 24 pounds is possible with heated nitrogen. Since electrical power is to be furnished to the heaters on an "as available" basis, the only cost for this weight savings is the added complexity of a priority switching provision to cut the electrical power supply in or out as necessary. At this stage in the program, it is expected that sufficient excess power will be available during almost the entire cruise mode, and probably during Mars orbit, as the result of varying power demands of spacecraft equipment during the mission. Operating the gas heaters on or off depending upon power availability deletes the need for any additional electrical power handling equipment such as additional solar panel surface area or storage batteries.

It is reasonable to expect that as the gas heaters are turned on and off during the mission, resulting in specific impulse change, thrust level would also change. This would require modulation of gas jet pulse width to maintain the attitude control design limit cycle and to benefit from the higher specific impulse. However, this added complexity to the control electronics will not be necessary. Flight proof test data for the Vela 3 heater-thruster assembly verified analysis, where a constant nozzle thrust level was observed over a 1000°F variation in chamber gas temperature. (See STL Report Number 2409-6023-RU000, Volume I, June 1965, prepared under Contract AF04(695)-36.) Since there is no reason to expect different performance in a Voyager heated-gas thruster, it is expected that no pulse width modulation will be required as compensation for heater switching.

The higher specific impulse with heating results from the higher gas temperature, giving an increase in gas characteristic exhaust velocity since it is theoretically proportional to the square root of absolute gas temperature. Power consumption of the heater assemblies is estimated at 128 watts total.

2.6.3 Antenna Drive

An alternate drive is presented for possible use as the medium gain antenna drive which would enhance the over-all spacecraft reliability by providing drives with different failure modes. The drive incorporates a stepper motor in combination with a nonreversing worm gear. The stepper motor is driven by a switching amplifier which has as its input a number of pulses proportional to the angular travel required.

The drive is shown in Figure 53. Table 27 gives qualitative information on the major differences between the selected and alternate drives.

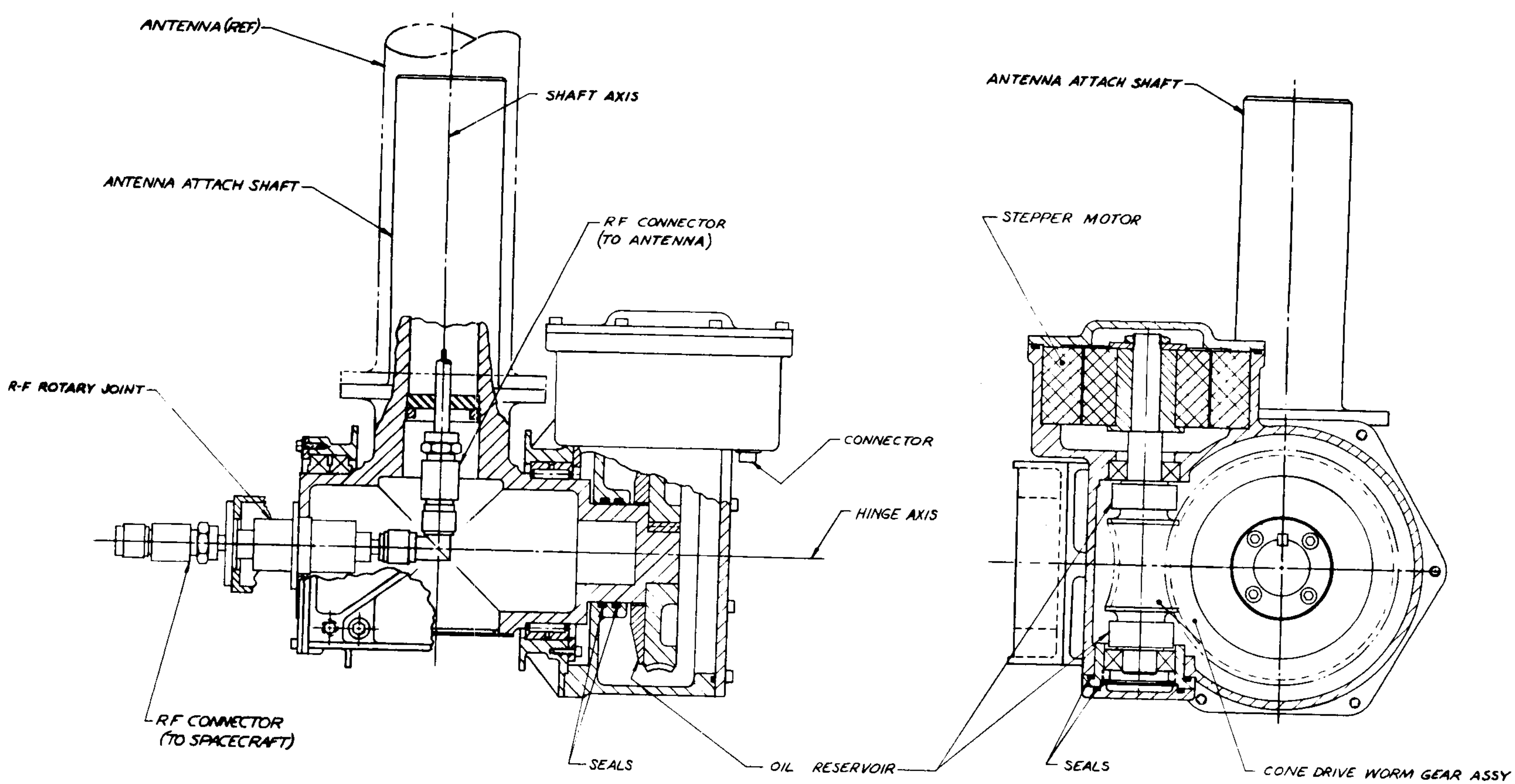


Figure 53. Alternate Antenna Drive

Table 27. Comparison — Alternate Antenna Drives

Category	Primary Drive	Alternate Drives	Comments
Mechanical design	• Spurgear reduction, wabble drive output	• Single cone drive worm gear • Non-reversible because of gear inefficiency	
Sealing	• Motor, bearings, all gearing except last stage, etc., in metal hermetically sealed container • Filled with dry inert gas	• Viton A for static seals • Two Butyl O-ring shaft seals • Filled with dry inert gas	
Lubrication	• Petroleum - ester base • Brayco NPT-4 oil for bearings and gears • Brayco KK-949B grease additional bearing lubricant • Porous reservoirs with Brayco NPT-4 oil	• Silicones - low vapor pressures • GE F-50 oil for bearings • GE F-300 Grease for worm • M_nS_2 burnished in worm • Porous reservoirs with GE F-50	Petroleum base lubricants have superior lubricating qualities
Motor	• AC servo motor • Low magnetic field • Well developed component	• DC stepper motor • Good environmental tolerance	AC and DC torque motors also considered for alternate

Table 27. Comparison — Alternate Antenna Drives (Continued)

Category	Primary Drive	Alternate Drives	Comments
System implementation (circuitry)	• DTA conversion, power amplifier required	• Digital word to pulse train conversion - power switching • Discrete step motion, can operate open loop	
General	• Reliability less dependent on lubricants • Successfully used on OGO	• Reliability less dependent on seals • Sealing and lubricating scheme successfully used on Mariner-Ranger • Lubrication demonstrated at Lockheed Missile & Space Co.	

3. S-BAND RADIO SUBSYSTEM

3.1 General Description

The elements of the S-band radio subsystem are listed below. The subsystem, in conjunction with the DSN stations, implements the functions of doppler and angle tracking, telemetry, ranging, and spacecraft command. A block diagram showing the arrangement and interconnections of the subsystem is shown in Figure 54.

- S-band receiver
- Receiver selector
- 1-watt transmitter and power monitor
- Modulator-exciter
- Power amplifier, power supply, and RF power monitor
- Transmitter selector
- 4-port hybrid ring and power monitors
- Circulator switch
- Diplexer
- High-gain antenna
- Medium-gain antenna
- Low-gain antenna

3.2 Requirements and Constraints

The following constraints are imposed on the S-band radio subsystem by the Voyager mission:

Two identical planetary vehicles are to be launched together on a single Saturn V launch vehicle. The separation between planetary vehicle arrival dates at Mars is to be not less than ten days. The radio subsystem of each vehicle shall accommodate, in conjunction with the DSN stations, uplink and downlink communications from prelaunch through end of mission.

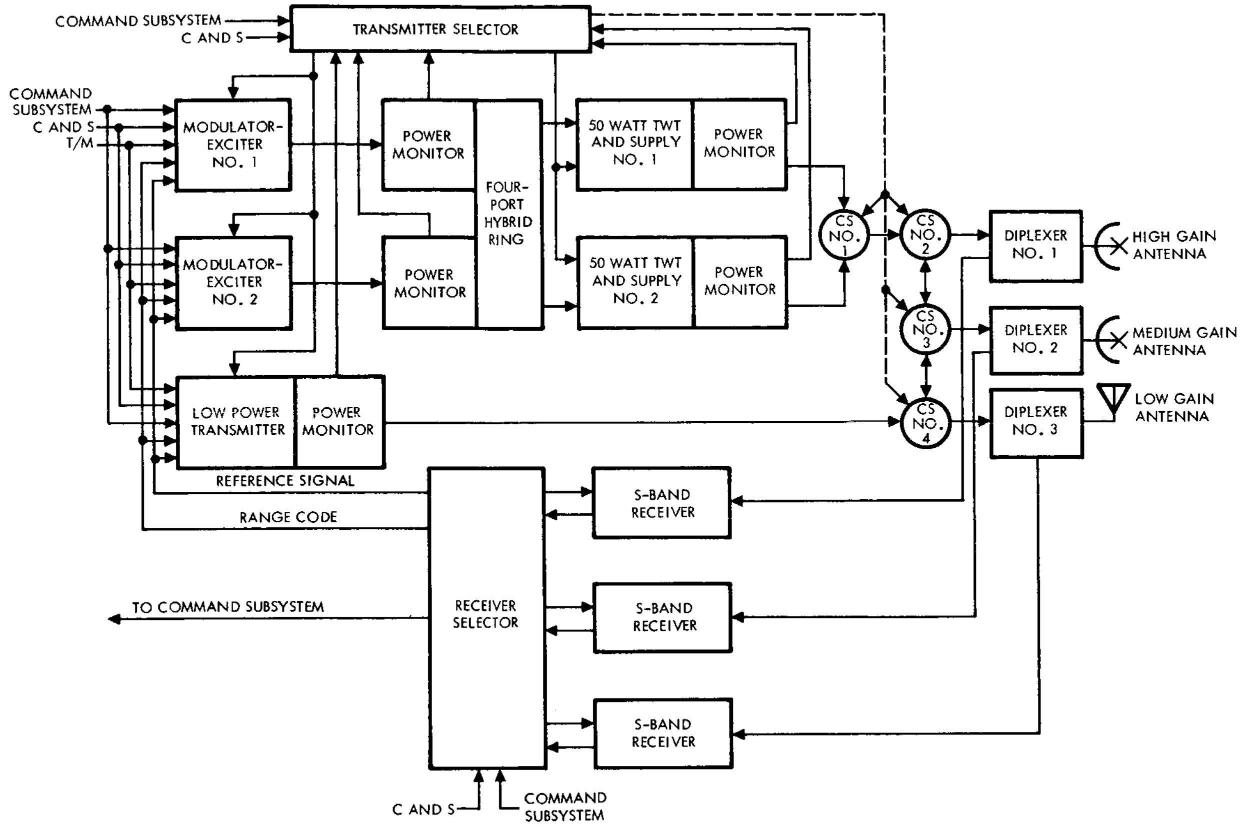


Figure 54. S-Band Radio Subsystem Block Diagram

Communications will be maintained from prelaunch to at least encounter plus two months. The capability to maintain communications to encounter plus six months is a design goal. Communication distances corresponding to these events are as follows:

- At encounter: 80×10^6 km minimum
  180×10^6 km maximum
- Encounter + 2 months: 150×10^6 km minimum
  265×10^6 km maximum
- Encounter + 6 months: 310×10^6 km minimum
  390×10^6 km maximum

The flight spacecraft is fully attitude stabilized, using the sun and Canopus as reference objects, except during maneuvers.

3.2.1 Subsystem Requirements

The S-band radio subsystem operating modes will be sequenced automatically for the nominal mission. Link performance is defined in accordance with the requirements of JPL documents IOM 3393-19-65 and IOM 3393-25-65. The S-band radio subsystem will be compatible with the planned capabilities of the DSN as defined by JPL Document EPD-283, dated 15 September 1965.

Uplink communications with each planetary vehicle will be accommodated from injection through end of mission by a low-gain, nonsteerable antenna. From prelaunch to interplanetary orbit injection, downlink telemetry is required from each planetary vehicle within the view capability of the DSN.

From interplanetary orbit injection to injection plus 30 days, the capability for simultaneous doppler tracking, telemetry, and command of both vehicles is required on a continuous basis when within view of the DSN. Simultaneous turnaround ranging capability is required (except during command operation).

From injection plus 30 days to Mars encounter minus five days, the capability for simultaneous doppler tracking, turnaround ranging (except during command operation), telemetry, and command is required for each vehicle, with DSN operation with two vehicles on a time-shared basis.

From Mars encounter minus five days to Mars encounter plus eight days, the capability for simultaneous doppler tracking, turnaround ranging (except during command operation), telemetry, and command is required for two planetary vehicles. From Mars encounter to termination of orbital operations, continuous telemetry from both planetary vehicles is required. Doppler tracking, turnaround ranging, and command of each vehicle will be on a time-shared basis.

The telemetry data rate of each planetary vehicle will not exceed 15,000 bits/sec. The peak total telemetry rate from both vehicles will not exceed 15,000 bits/sec. The signal characteristics and radiated power of the downlink transmission will be such that the telemetry bit error rate for reception by DSN stations does not exceed 5×10^{-3}.

The capability for uplink command and downlink telemetry will exist during roll maneuvers following sun acquisition. Continuous communication during a trajectory correction maneuver is not required. (The data storage subsystem is required to store engineering data during maneuvers.)

Checkout and monitoring of RF system parameters on the launch pad by transmission of the spacecraft RF signal will not require a mechanical connection between the spacecraft and the nose fairing.

3.2.2 Design Requirements

The nominal RF power radiated from each spacecraft will be 50 watts.

The power supply for the RF power amplifier will draw unregulated DC power, not exceeding 150 watts. All other elements of the subsystem will draw regulated AC power.

The pointing angles of directional antennas will be automatically updated, as required, throughout the mission.

Selected S-band radio subsystem measurements will be telemetered in support of DSN flight operations. These will include subsystem mode status, receiver signal strength, and receiver loop stress.

The subsystem will incorporate redundancy and alternate modes of operation such that in the event of subsystem element failures communication can be maintained.

3.3 Functional Interfaces

3.3.1 Electrical Input Signals

The DSIF/DSN system provides a signal in the 2115 ± 5 mc band. The maximum bandwidth required for this signal is 3.3 mc.

The telemetry subsystem supplies a composite telemetry signal to the three transmitter phase modulators. The bandwidth of this signal is DC to 500 kc. The relative amplitudes of the data and sync signals are adjusted in the telemetry subsystem to provide the desired relative modulation indices.

The computing and sequencing (C and S) subsystem supplies all sequencing commands required for a nominal mission. These are listed in Table 28. Control signals are pulses of 50 ms duration.

Table 28. Command Inputs

Function	Primary Source of Command	Backup Source of Command	Type Control	Notes
Transmitter-Antenna Selection				
Low power, low gain	Ground	None	Discrete	3
Low power, high gain	Ground	None	Discrete	3
High power, high gain	C&S	Ground	Discrete	1, 3
High power, medium gain	C&S	Ground	Discrete	1, 3
High power, low gain	C&S	Ground	Discrete	1, 2, 3
Low power, medium gain	Ground	None	Discrete	3
PA No. 1	Ground	None	Discrete	3
PA No. 2	Ground	None	Discrete	3
Mod. Exciter No. 1	Ground	None	Discrete	3
Mod. Exciter No. 2	Ground	None	Discrete	3
Receiver Selection				
Maximum gain	C&S	Ground	Discrete	
Maximum coverage	C&S	Ground	Discrete	2
Noncoherent override - on	Ground	None	Discrete	
Noncoherent override - off	Ground	None	Discrete	
Receiver No. 1 disconnect	Ground	None	Discrete	
Receiver No. 2 disconnect	Ground	None	Discrete	
Receiver No. 3 disconnect	Ground	None	Discrete	
All receivers on	Ground	None	Discrete	
Range Code				
On	Ground	None	Discrete	
Off	Ground	None	Discrete	

Notes 1. Interlocked until preset time after separation

2. Special mode selected in the event of loss of attitude control

3. Transmitter selector provides automatic switching for RF power loss in all transmission modes (see 3.4.4-b)

The command subsystem provides signals to select alternate modes in case of an abnormal mission and provides backup of C and S controls. The signals are also listed in Table 28 .

The S-band radio subsystem requires both AC (50 vrms ±2%) and DC (50 vdc ±1%) primary power. The power amplifiers use DC; all other subsystem elements use AC.

The guidance and control subsystem provides the controls for positioning the high- and medium-gain antennas. The pointing error is held to less than 0.82 degree in each control axis.

3.3.2 Electrical Output Signals

The power amplifiers provide 50 watts in the 2295 ± 5-mc band. The maximum channel bandwidth is 3.3 mc.

The low-power transmitter provides 1 watt in the 2295 ± 5 mc band. The maximum channel bandwidth is 3.3 mc.

Table 29 lists data monitoring points. Telemetry points are isolated through a high impedance from the internal circuits such that terminals can be shorted without degrading subsystem operation.

3.3.3 Mechanical Interfaces

The low-gain antenna is mounted on the end of a 14-foot deployable boom with a single hinge attachment at the edge of the solar array structure.

The medium-gain antenna is located at the edge of the solar array such that a single gimbal provides ±90 degrees of unobstructed line of site (LOS). Provisions are made to support the unbalanced mass of the antenna during launch accelerations.

The high-gain antenna is located at the edge of the solar array such that a double axis gimbal provides 3π stearadian unobstructed LOS plus stowage. This is accomplished by providing a 360-degree shaft axis capability and a ±90 degree hinge axis capability, plus stowage.

3.3.4 Thermal Interfaces

The TWT dissipates approximately 80 watts of heat and requires a maximum baseplate temperature of 165°F.

Table 29. Telemetry Monitoring Points

Unit and Function	Type*	Sampling Rate**	No. of Channels	Priority***
Low Power Transmitter				
Power output	A	1/4 min.	1	1
Voltage (2)	A	1/8 min.	2	2
Modulator Exciter (2)				
Power output	A	1/4 min.	2	1
Voltage (2)	A	1/8 min.	4	2
Power Amplifier (2)				
Power output	A	1/4 min.	2	1
Helix current	A	1/8 min.	2	2
Temperature (2)	A	1/8 min.	4	3, 4
Voltage (2)	A	1/8 min.	4	3, 4
S-Band Receiver (3)				
In-lock	D	1/24 sec.	3	1
Signal strength	A	1/24 sec.	3	2
Loop stress	A	1/1.2 sec.	3	1
VCO temperature (freq)	A	1/8 min.	3	2
Temperature	A	1/8 min.	3	3
Voltage (2)	A	1/8 min.	6	3
Receiver Selector				
Mode (5)	D	1/4 min.	5	1
Voltage (2)	A	1/8 min.	2	3
Transmitter Selector				
Mode (8)	D	1/4 min.	8	1
Voltage (2)	A	1/8 min.	2	3
RF Switches (4)				
State (8)	D	1/4 min.	8	1

*A = Analog; D = Discrete

**Sampling rate at 234 bits/sec, all engineering telemetry mode

***1. Required for performance of flight operations
2. Required to verify performance of specific subsystem functions
3. Required to indicate effect of the space environment on subsystem performance
4. Required to evaluate components previously unexposed to a space environment

All antennas are designed to withstand a thermal space environment of -250 to +300°F, which is well in excess of the range they will experience during direct solar heating and space cold-soaking conditions. The gimbal drive mechanisms incorporate heaters to limit low temperature extremes.

The medium- and high-gain antennas, when deployed, are well out of the engine's plume field. Radiation and convective heating effects are small in comparison to direct solar heating. The antennas can withstand temperatures in excess of 300°F, which will provide an adequate thermal safety factor. The maximum stabilization temperature of the low-gain antenna is estimated at 150°F during engine firing.

3.3.5 Launch Vehicle Shroud Interface

RF windows are required in the metallic wall of the shroud to permit transmission. The required windows are 2.5 inches in diameter with a teflon shield or insert. The window acts as a short section of circular waveguide operating in the TE_{11} circularly polarized waveguide mode, providing broad pattern coverage about the side of the vehicle in which is is mounted. One window is adjacent to the stowed low-gain antenna to permit on-stand and down-range telemetry and tracking to separation. Similar windows provide coupling to the high- and medium-gain antennas for on-stand checkout.

3.3.6 Pyrotechnics Interface

All antennas are deployed by a releasing mechanism actuated by redundant pyrotechnic actuator. The exposure device cuts a pin that releases a clevis holding the antennas. The gimballed antennas are then under pointing control of their respective gimbal drive systems. The low-gain antenna is mechanically deployed to its fully-extended position.

3.4 Design Description

3.4.1 Functions

The Voyager spacecraft S-band radio subsystem performs the following functions:

a) Receives RF signals transmitted to the spacecraft from the DSIF stations

b) Coherently translates the received RF signal by a fixed ratio

c) Demodulates the received RF signal and sends the detected command subcarrier signal to the spacecraft command subsystem

d) Phase-modulates the transmitter with the demodulated ranging signal from the turn-around ranging channel and/or a composite telemetry signal

e) Transmits a modulated RF signal to the DSIF stations using the translated RF signal (b, above) or an independent frequency source

f) Provides appropriate antenna patterns for the functions described in a) and e), above.

3.4.2 Subsystem Organization

As shown in Figure 54, the transmitter portion consists of two modulator-exciters cross-strapped via the four-port hybrid ring to drive two redundant 50-watt power amplifiers. These power amplifiers can be connected to any of three antennas (low, medium, or high-gain) via the circulator switches. A low-power transmitter is provided primarily for launch mode telemetry, but it can also be connected to any antenna for failure mode communications. The relative modulation index for the two-channel telemetry signal is set in the telemetry subsystem. However, the absolute index of the composite telemetry signal is set in the modulator-exciter and low-power transmitter. This absolute index is switched in these units when simultaneous turnaround ranging and telemetry is being transmitted. The transmitter selector switches power amplifiers, modulator-exciters, and low power transmitters in case of failure. It also provides control signals to the circulator switches for transferring RF signals to the various antennas.

The receiver portion consists of three S-band receivers and a receiver selector. Each receiver is connected to one antenna (via a diplexer). The receiver selector provides the logic for selecting the receiver to provide signals to the modulator-exciters, low-power transmitter, and command subsystem.

3.4.3 Mission Operation

At launch the low-power transmitter and S-band receivers are turned on. The low-power transmitter is connected to the low-gain antenna and provides 234.4 bits/sec spacecraft telemetry independent of the launch vehicle communication system. DSIF Station 71 (Kennedy Space Center) receives from both spacecraft during the initial powered flight phase. During the parking orbit, Station 72 (Ascension), if in view, and the 85-foot antenna sites (via the acquisition aid antenna) in view, receive signals from both planetary vehicles. Between separation and sun acquisition, the guidance and control subsystem holds spacecraft attitude within ± 10 degrees of injection attitude, thus maintaining acceptable look angles to the spacecraft low-gain antenna. Shortly after injection into the Mars trajectory, the DSIF station in view establishes two-way lock with both planetary vehicles using the acquisition aid antenna and frequency multiplexed transmission. Both spacecraft are receiving and transmitting over their low-gain antennas. During the first pass after injection, the DSIF station switches reception from the acquisition aid antenna to the main 85-foot dish. Frequency-multiplexed uplink communication continues via the acquisition aid antenna for the entire first pass. Doppler data is obtained to establish ephemerides. The second DSIF station to acquire after injection uses the 85-foot dish for both up and down links with frequency-multiplexed 10-kw transmission.

At approximately 15 hours after launch, the 50-watt transmitter is turned on and connected to the low-gain antenna. This remains the normal cruise mode for 75 days; viz: receive-transmit on low-gain antenna, 50-watt power, 234.4 bits/sec telemetry. Turnaround ranging mode is actuated by ground command and operates simultaneously with 234.4 bits/sec telemetry. The DSIF stations use 85-foot antennas and 10-kw transmitters for continuous two-way communication with both spacecraft. If both spacecraft remain within the beamwidth of a single 85-foot antenna, then the uplink can be operated in a multiplexed mode; otherwise, separate sites are required. After 30 days, two-way communications with both spacecraft is maintained on a time-shared basis.

After approximately 75 days, both spacecraft switch transmission and reception to high-gain antennas. This provides communication via the 10-kw, 85-foot sites until Mars encounter. The telemetry bit rate remains at 234.4 bits/sec until encounter. For ranging measurements after approximately 15 days, the spacecraft is switched by ground command to the high-gain antenna. For ranging after about 150 days, a 100-kw ground transmitter is required with 210- or 85-foot antennas.

As the first spacecraft approaches Mars five days from encounter (E - 5 days), the DSIF sites are switched from the 85-foot to the 210-foot facility for this spacecraft. The second spacecraft continues to operate with the 85-foot sites until E - 6 hours. Continuous two-way communication is thus provided to both spacecraft for the period of E -5 to E + 10 days. Although the 210-foot facility need not be used until after the bit rate is increased, it is activated at E -5 days for the first spacecraft to provide a transition period. After encounter, the first spacecraft transmits at 7500 bits/sec to support the high data rate planetary science. At E -6 hours for the second spacecraft, two-way communications is switched from the 85-foot to the 210-foot sites. The 210-foot facility also continues to receive from the first spacecraft. After encounter, the second spacecraft also transmits at 7500 bits/sec. The 210-foot sites then receive simultaneously from both spacecraft at this bit rate until the end of the mission. The uplink operates on a time-shared basis using the 100-kw transmitters on the 210-foot antennas. If only a single spacecraft is operational during the early Mars orbital phase, the spacecraft data rate is increased to 15,000 bits/sec.

For maneuvers, the normal sequence is to slew the spacecraft high-gain antenna to the position required so as to be earth pointing when in the attitude-maneuver orientation. Verification of the high-gain antenna angles over the low- or medium-gain antenna is accomplished prior to maneuver. (The low-gain antenna is used for the early maneuvers; the medium-gain antenna for the later maneuvers.) The spacecraft maneuver-enable command is then transmitted and is received over either the low- or medium-gain antenna. After maneuver attitude execution has been confirmed over the high-gain antenna, the motor-burn-enabling command from the ground station is received over the high-gain antenna. After the propulsion maneuver, the spacecraft automatically goes through Sun-Canopus acquisition and reorients the high-gain antenna toward earth. The radio subsystem is returned to cruise mode and data recorded during the maneuver is transmitted to earth, time-multiplexed with real-time science and engineering data.

The radio subsystem also provides backup capability 1) to receive commands on the low-gain antenna to encounter plus six months, 2) to transmit 7.3 bits/sec telemetry on the low-gain antenna to worst-case encounter range, and 3) to transmit 7.3 bits/sec telemetry to encounter plus six months, using the 1-watt, low power transmitter on the high-gain spacecraft antenna.

3.4.4 Modes of Operation

a. Antenna Transfer

The low- and high-power transmitters can transmit over the low-, medium-, or high-gain antennas. The switch positions for connecting any transmitter to any antenna are given in Table 30.

Table 30. Antenna Transfer Switch Positions

RF Power Source	Antenna	Circulator Switch Position			
		CS No. 1	CS No. 2	CS No. 3	CS No. 4
PA No. 1	Low	CW	CCW	CCW	CW
	Medium	CW	CCW	CW	–
	High	CW	CW	–	–
PA No. 2	Low	CCW	CCW	CCW	CW
	Medium	CCW	CCW	CW	–
	High	CCW	CW	–	–
Low Power Transmitter	Low	–	–	–	CCW
	Medium	–	–	CCW	CW
	High	–	CCW	CW	CW

b. Transmitter Transfer

The transmitter consists of two redundant power amplifiers, two redundant modulator-exciters, and a low-power transmitter. On-board power monitors and control logic (subject to C and S or command override) are used to switch power amplifiers, exciters, and the low-power transmitter.

The following combinations can also be selected by ground command:

- Modulator-Exciter No. 1, Power Amplifier No. 1
- Modulator-Exciter No. 2, Power Amplifier No. 1
- Modulator-Exciter No. 1, Power Amplifier No. 2
- Modulator-Exciter No. 2, Power Amplifier No. 2
- Low-Power Transmitter

The transmitter select logic (Table 31) provides automatic failure correction for loss of RF power.

Table 31. Transmitter Selector Logic

Initial Condition	Malfunction	Backup Condition
1-watt, low-gain antenna	Loss of power	50-watt, low-gain antenna*
1-watt, high-gain antenna	Loss of power	50-watt, high-gain antenna
1-watt, high-gain antenna	Loss of sun or Canopus lock	1-watt, low-gain antenna**
50-watt, high-gain antenna	Loss of power	Switch power amplifiers***
50-watt, high-gain antenna	Loss of sun or Canopus lock	50-watt, low-gain antenna

*Inhibited by interlock for 15 hours after launch

**Inhibited by C&S for fixed time interval during maneuvers

***Provided that modulator exciter output is normal; if not, then modulator-exciters are switched

c. Receiver Transfer

Three receivers are provided; one is permanently connected to each antenna. All receivers are operated continuously with the output of one selected by logic in the receiver selector (subject to C and S and command subsystem override). There are two basic modes:

1) Maximum Coverage. The receiver that is in-lock and provides maximum area coverage is selected to provide signals to the command demodulators, the modulator-exciters, and low-power transmitter. The priority of the logic is:

- Use the receiver connected to the low-gain antenna if it is in-lock.
- Use the receiver connected to the medium-gain antenna if it is in-lock and condition above is not satisfied.
- Use the receiver connected to high-gain antenna if it is in-lock and both conditions above are not satisfied.

2) Maximum Gain. The receiver that provides maximum antenna gain and is in-lock is selected to provide signals to the command demodulators, modulator-exciters, and low-power transmitter. The prority of the logic is:

- Use the receiver connected to the high-gain antenna if it is in-lock.
- Use the receiver connected to the medium-gain antenna if it is in-lock and the condition above is not satisfied.
- Use the receiver connected to the low-gain antenna if it is in-lock and both conditions above are not satisfied.

A guidance and control override is used to switch to the maximum coverage mode if the vehicle loses sun or Canopus lock. The override is inhibited during normal maneuvers. Provision is also made to turn off any receiver on ground command.

d. Coherent — Noncoherent Operation

When no S-band receiver is in-lock, the transmitted carrier frequency is controlled by a crystal oscillator. When one of the S-band receivers is in-lock, the transmitted carrier frequency is 240/221 times the received carrier frequency.

e. Telecommunication Modes

The radio subsystem is capable of simultaneous two-way doppler, command, and telemetry. Turnaround ranging can operate simultaneously with the above signals, but with degraded performance margins. Normally, there is no requirement for simultaneous ranging and command; however, simultaneous ranging and 234.4 bits/sec telemetry is provided as a normal mode.

f. Telemetry Bit Rates

There are six telemetry bit rates:

1) 15,000 bits/sec (orbital operations)
2) 7,500 bits/sec (orbital operations)
3) 3,750 bits/sec (orbital operations)
4) 1,875 bits/sec (orbital operations)
5) 234.4 bits/sec (launch, cruise, maneuver)
6) 7.3 bits/sec (emergency mode)

3.4.5 Subsystem Elements

a. High-Gain Antenna

The high-gain antenna (Figure 55) is a horn-fed circular paraboloid with 2 degrees of pointing freedom. Polarization is right-hand circular. It is designed to provide a nominal gain of 34 db at the downlink frequency of 2295 mc. The reflector has a 9.5-foot diameter aperture which is illuminated by a conical-horn, focal-point feed. Pointing accuracy constraints have led to an 18-db edge illumination taper on the high-gain antenna for main beam broadening without appreciable loss in gain. With the 18-db taper, the calculated half-power beamwidth is 3.4 degrees. The flared aperture of the feedhorn provides proper illumination of the reflector with minimum aperture blockage and mismatch between reflector and feed. Left-hand circular polarization is excited within the short length of circular waveguide feeding the conical horn by a two-arm Archimedian spiral element.

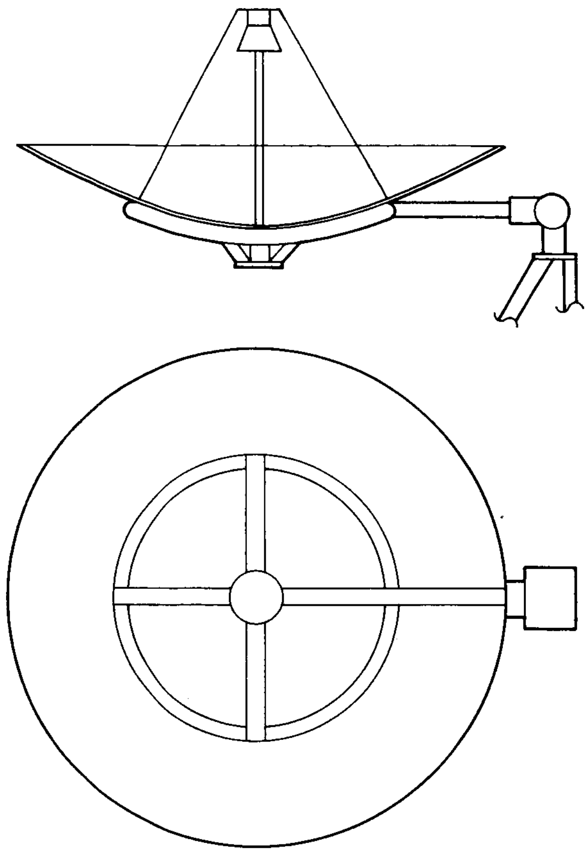


Figure 55. High-Gain Antenna

A balun transformer is incorporated in the transmission line to match the feed. The support of the paraboloid is an arm type tubing assembly, and the entire assembly is mounted to the spacecraft through the use of a double-gimballed actuator and drive mechanism.

The actuator and drive mechanism allow positioning the paraboloid main beam through 3π steradians (360-degree shaft-axis motion, $\pm$90-degree hinge-axis motion) as well as allowing stowage of the antenna within the spacecraft fairing envelope. Antenna pattern coverage is limited only by the blockage due to the spacecraft itself. The RF transmission path is required to traverse two axes of rotation in the gimbal assembly, requiring the use of noncontacting type rotary joints which are integral parts of the actuator and gimbal assembly.

The reflector is fabricated from type 2024 aluminum sheet, 0.03 inch thick and perforated to reduce weight. The feed horn is made from the same material and is supported by a column which incorporates the feed line and is guyed for rigidity.

The boom yoke assembly is fabricated from heavy wall type 6061 aluminum tubing and is pinned to the gimbal assembly which is mounted on the spacecraft structure.

b. S-Band Medium-Gain Antenna

The medium-gain antenna (Figure 56) is a single-gimballed, horn-fed paraboloid with an elliptical aperture having major and minor axis dimensions of 84 and 36 inches, respectively. The antenna polarization is right-hand circular, and provides a 28-db gain at the downlink frequency of 2295 mcs. Except for the differences in sizes and shapes, the medium-gain antenna design concepts are identical to those of the high-gain antenna. The elliptical aperture provides a fan-shaped beam

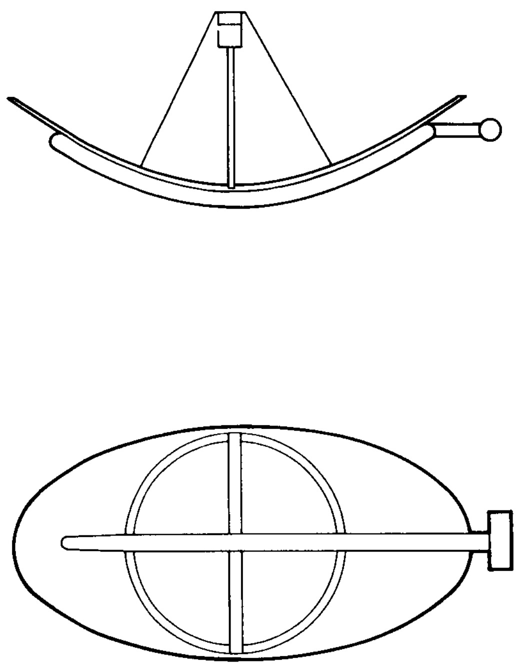


Figure 56. Medium-Gain Antenna

having its wide dimension normal to the scan direction. The single axis motion of this antenna provides both stowage and repointing capability. The lower mass of the medium-gain antenna structure, the shorter moment arm, and the elimination of one axis of rotation allows the gimbal-actuator assembly to be reduced in size. The actuator incorporates a single integral RF rotary joint. The feed horn for the elliptical paraboloid is the most critical element of the medium-gain antenna design. Although the method of excitation of the circularly polarized wave will be identical to that used for the high-gain antenna feed horn, a feed horn with an elliptical aperture is required for proper illumination of the unsymmetrically-shaped reflector. A guyed column is used to support the feed horn.

The secondary pattern beamwidths of 4 by 10 degrees, at the half-power level, allow a maximum pointing error of 6 degrees in the fixed plane direction, one degree in the updated plane direction.

c. S-Band Low-Gain Antenna

The low-gain antenna is a boom-mounted, right-hand circularly polarized cup turnstile. The antenna is cylindrical in shape with a diameter of 3.5 inches and a height of 2 inches. The turnstile, or crossed dipole elements, are mounted on a split tube balun which is integral with the feed transmission line. Proper phasing of the elements to obtain circular polarization is accomplished by adjusting the relative lengths of the orthogonal elements. Radiation pattern coverage of the antenna is slightly greater than hemispherical, with a 10-db pattern beamwidth of approximately 190 degrees, and symmetrical about the element axis. The boresight gain, with matched polarization, is greater than 5 dbi, with corresponding axial ratios of less than 2 db over the uplink to downlink bandwidth. At the design center frequency, the axial ratio is held to 0.5 db on axis, with off-axis axial ratios not exceeding 3 db over the hemisphere.

From prelaunch to injection, the antenna is in the stowed configuration, pointing in the downrange direction at a clock angle of 75 degrees and a cone angle of 90 degrees under the shroud. To maintain the spacecraft-to-ground communication during this phase of the mission, an RF window is placed in the shroud near the antenna aperture. A coupler between the shroud window and the low-gain antenna provides the RF interface required to maintain the powered flight telemetry link.

After spacecraft separation, the 14-foot boom is erected and locked into a fixed position such that the low-gain antenna main beam is pointed in a cone angle of zero degrees. The boom is tilted to a cone angle of one-degree outboard to prevent solar cell shadowing. A simple prewrapped, flexible, coaxial cable is used across the actuator to maintain the RF transmission path.

d. Preliminary Specifications, S-Band Radio Subsystem

The preliminary specifications for the subsystem are shown in Figures 57 through 66.

3.5 Parameters and Performance Summary

3.5.1 Communications Link Parameters

The values of the communications system parameters that determine the over-all system performance are summarized in Tables 32 through 38, and Figures 67 and 68.

3.5.2 Link Performance Definitions

a. Typical Communications Profile

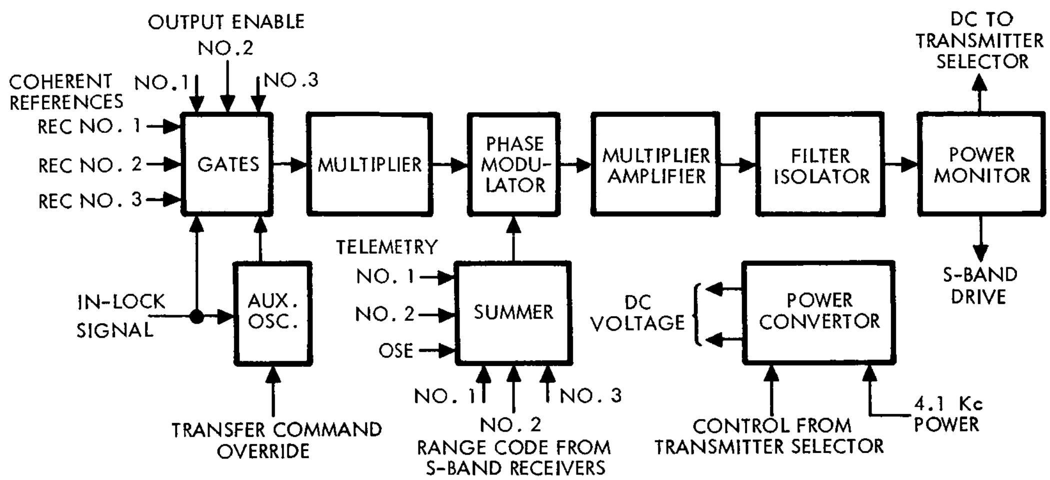
To determine the link performance, it is necessary that the critical events in the mission be identified and the parameters affecting the communication capability be determined for each of these events. Figure 69 shows a communication range and cone angle versus time from launch for the nominal trajectory, which is used to determine the the mission profile shown in Figure 70. Three midcourse correction maneuvers, a capsule vehicle separation maneuver, and the deboost maneuver at encounter are indicated, with typical times and ranges for each. Table 39 gives a more detailed time breakdown for each of the maneuver sequences and shows the spacecraft antennas available during each phase of the maneuver. The times given in this table are not critical, from the communication point of view, but are used in plotting the performance margin for the various maneuvers. The ranges (particularly at encounter) indicated in Figures 69 and 70 do not represent maximum values. The maximum range values and the associated cone angles are:

	Range	Cone Angle
• Encounter	1.80×10^8 km	42^o
• Encounter plus 1 month	2.65×10^8 km	35^o
• Encounter plus 6 months	3.9×10^8 km	12^o

The link performance at these maximum ranges is discussed in Paragraph 3.5.3.

PRELIMINARY SPECIFICATION

I Modulator – Exciter



Function

Provides RF drive to both power amplifiers through the four-port hybrid; phase modulates the telemetry and ranging signals onto the S-band carrier. The modulator-exciter operates from either a coherent reference (VCO) signal, supplied by the S-band receivers or from the auxiliary (crystal) oscillator. The choice is made from the presence or absence of the in-lock signal from the receiver.

Characteristics

Frequency	2295 ± 5 mc
Power output	100 mw minutes
Phase stability	6° peak - auxiliary mode 12° peak - coherent mode
Frequency stability	
a) Short term	10^{-7} 20 minutes } Auxiliary mode
b) Long term	10^{-6} 12 hours } Auxiliary mode
Modulation bandwidth	1.8 mc
Peak modulation	±4 radians
Modulation sensitivity	(six inputs isolated by at least 30 db)
a) Telemetry inputs	1 radian ±5 percent/volt peak
b) Ranging inputs	2 radians ±5 percent/volt peak
Spurious emission	Any spurious less than 50 db below unmodulated carrier
RF input impedance	50 ohms - VSWR less than 1.2:1
RF output impedance	50 ohms - VSWR less than 1.4:1

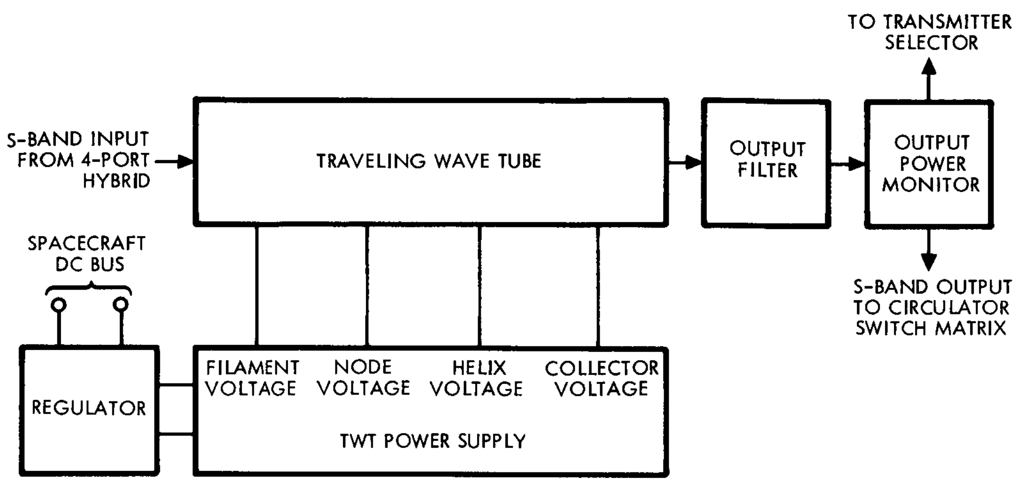
Physical Characteristics

Weight	3.0 pounds
Size	90 cubic inches
Power	2 watts

Figure 57. Preliminary Specification - Modulator–Exciter

PRELIMINARY SPECIFICATION

II RF Power Amplifier with DC-DC Converter



Function

Amplifies the S-band signal from the modulator-exciter to a level of 50 watts for tansmisssion from the spacecraft to earth. The power amplifier includes a traveling wave tube (TWT), the power supply circuits required to operate the TWT from a low voltage DC bus and provide calibrated telemetry voltages, and RF power monitor and an RF output filter.

Characteristics

Frequency	2295 ±5 mc
Power output	50 watts
RF gain saturated	35 db
Noise figure	35 db
RF impedance	50 ohms
VSWR	1.2:1 maximum
Turn on time	90 seconds
DC voltage input	50 ±0.5 vdc
Spurious output	Less than -80 dbm in any cps band in the range of 2110-2120 mc
Harmonics	60 db below unmodulated carrier

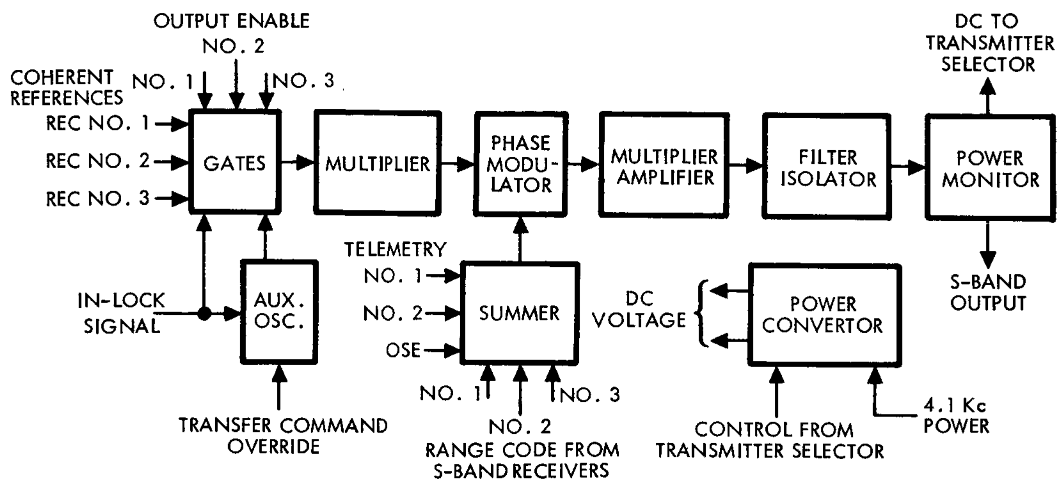
Physical Characteristics

Weight	7.8 pounds
Power	150 watts
Size	200 cubic inches

Figure 58. Preliminary Specification — II RF Power Amplifier with DC-DC Converter

PRELIMINARY SPECIFICATION

III Low Power Transmitter



Function

Transmits telemetry during the boost and early portion of the mission and phase modulates (PM) the telemetry or ranging signal onto the S-band carrier. The low power transmitter operates from either a coherent reference (VCO) signal, supplied by the S-band receivers or from auxiliary (crystal) oscillator. The choice is made from the presence or absence of the in-lock signal from the receiver.

Characteristics

Frequency	2295 ±5 mc	
Power output	1 watt minute	
Phase stability	6° peak - auxiliary mode 12° peak - coherent mode	
Frequency stability		
a) Short term	10^{-7} 20 minutes	Auxiliary mode
b) Long term	10^{-6} 12 hours	
Modulation bandwidth	1.8 mc	
Peak modulation	±4 radians	
Modulation sensitivity	(six inputs isolated by at least 30 db)	
a) Telemetry	1 radian ±5 percent/volt peak	
b) Ranging	2 radians ±5 percent/volt peak	
Spurious Emission	Any spurious less than 50 db below unmodulated carrier	
RF input impedance	50 ohms - VSWR less than 1.2:1	
RF output impedance	50 ohms - VSWR less than 1.4:1	

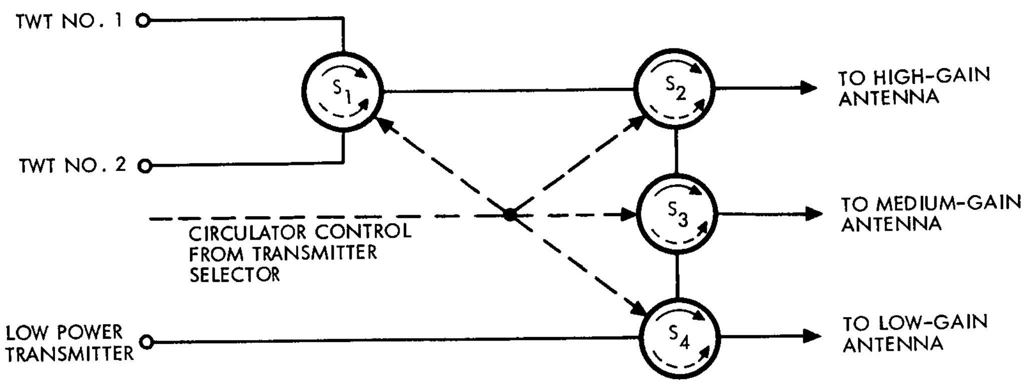
Physical Characteristics

Weight	3.5 pounds
Size	90 cubic inches
Power	10 watts

Figure 59. Preliminary Specification — III Low Power Transmitter

PRELIMINARY SPECIFICATION

IV RF Switches



Function

Switch any of three transmitters to any of three antennas. Consist of four ferrite circulator sections, each controlled independently. The connection of transmitters to antennas is controlled by the sense of circulation of the sections. One transmitter will be on at a time.

Characteristics

Frequency	2295 ± 5 mc
Insertion loss	0.25 maximum/section
Isolation	25 db minimum/section
Power handling capability	100 watts (shall be capable of switching with RF power applied)
Impedance	50 ohms
VSWR	1.1:1 maximum at each port
Control signal	5 vdc at 20 ma

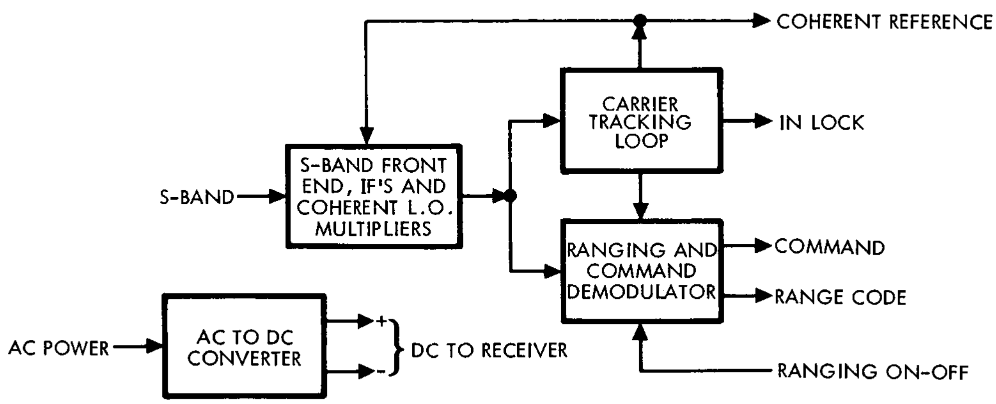
Physical Characteristics

Weight (4 switches)	7.3 lb maximum
Power (4 switches)	1 watt
Size (4 switches)	36 cubic inches

Figure 60. Preliminary Specification — IV RF Switches

PRELIMINARY SPECIFICATION

VI S-Band Receiver



Function

Receives transmission from DSIF, phase tracks the RF carrier, provides coherent reference for modulator-exciter, and demodulates composite command subcarrier and PN range code.

Characteristics

Frequency	2115 ± 5 mc
Noise figure	10 db
Carrier tracking loop	$2 B_L$ = 32 cps (for SNR-6 db)
Predetection bandwidth	4.5 kc
Ranging channel RF bandwidth	3.3 mc
Dynamic range	-145 to -50 dbm

Physical Characteristics

Weight	5 pounds
Power	2.5 watts
Size	150 cubic inches

Figure 61. Preliminary Specification — VI S-Band Receiver

PRELIMINARY SPECIFICATION

Transmitter Selector and Receiver Selector

Transmitter Selector

Function

Controls the selection of one of two mod-exciters, one of two power amplifiers, or the low power transmitter, as well as the choice of antennas through circulator switches by means of preset logic and commands.

Characteristics

Inputs

Power monitors	From mod-exciters and power amplifiers
Ground commands	From command subsystem
On-board commands	From computing and Sequencing subsystem

Outputs

Mod-exciter on/off	To mod-exciter
Power amplifier or low power transmitter on/off	To power amplifier
Circulator switches on/off	To circulator switches
Auxiliary oscillator override	To mod-exciters and low power transmitter

Physical Characteristics

Weight	1.0 pounds
Power	0.8 watts
Size	36 cubic inches

Receiver Selector

Function

Selects the outputs of one of three receivers, based on preset logic and commands.

Characteristics

Inputs

In lock	From receivers
Ranging on/off	From command subsystem
Ground override	From command subsystem
Mode select	From C & S and command subsystem

Outputs

Ranging on/off	To receivers
Enable outputs	To receivers

Physical Characteristics

Weight	1.0 pounds
Power	0.8 watts
Size	36 cubic inches

Figure 62. Preliminary Specification — Transmitter Selector and Receiver Selector

PRELIMINARY SPECIFICATION

Diplexer and Four-Port Hybrid

Diplexer

Function

Allow a common antenna to be used for transmission and reception by directing the transmitted signal at 2295 mc to the antenna and the received signal at 2115 mc to the receiver. The diplexer will consist of two bandpass filter sections and a matching junction.

Characteristics

Frequency	2295 ± 5 mc 2115 ± 5 mc	Transmit Receive
Passband insertion loss	0.36 db maximum	Transmit and receive
Impedance	50 ohms	
Passband VSWR	1.2:1 maximum	
Isolation	80 db minimum	(At both transmit and receive frequencies)
Power handling capability	100 watts	

Physical Characteristics

Weight	1.3 pound maximum
Volume	50 cubic inches

Four-Port Hybrid

Function

Provides one-half of the S-band power from either of two modulator-exciters to each of the TWT power amplifiers.

Characteristics

Frequency	2295 ± 5 mc
RF Power capability	500 mw
Insertion loss	3.5 db maximum
Isolation	25 db (between input ports)
Impedance	50 ohms
VSWR	1.3:1 maximum

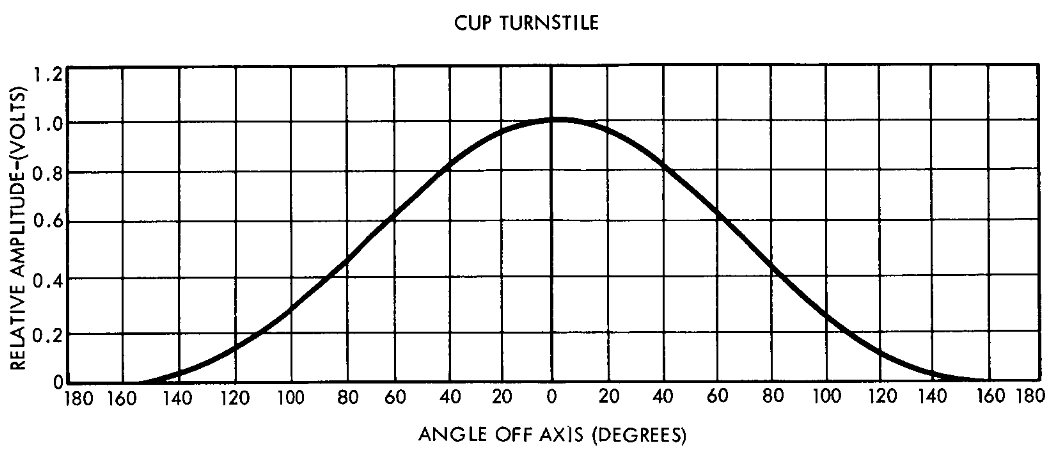
Physical Characteristics

Weight	0.6 pounds
Volume	30 cubic inches

Figure 63. Preliminary Specification — Diplexer and Four-Port Hybrid

PRELIMINARY SPECIFICATION

S-Band Low Gain Antenna



Antenna Characteristics

Configuration	A boom-mounted cup-backed turnstile antenna.

Function

To provide omnidirectional pattern coverage for command reception at maximum range for all nominal spacecraft attitudes.

Performance Characteristics

Maximum gain (relative to circular isotropic)	5.0 ±0.5 db at 2115 mc
	4.2 ±0.5 db at 2295 mc
Polarization	Right-hand circular
Ellipticity on axis	1.0 ± 1.0 db at 2115 mc
	3.0 ± 2.0 db at 2295 mc
Impedance	50 ohm nominal
VSWR	1.5:1 maximum
10 db beamwidth	180 degrees minimum

Physical Characteristics

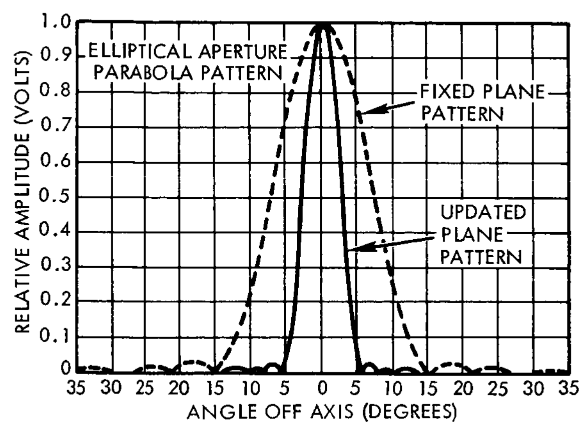
Antenna size	3.5 diameter x 2.0 inch
Total weight	0.55 pounds
Connector	Type N male aluminum
Coaxial cable	Semi-rigid aluminum and RF-142/U
Mounting	Single hinge point
Boom length	14 feet

A boom-mounted backed turnstile antenna.

Figure 64. Preliminary Specification — S-Band and Low Gain Antenna

PRELIMINARY SPECIFICATION

S-Band Medium Gain Antenna



Antenna Characteristics

Configuration	A single-gimballed medium-gain elliptical paraboloid reflector with an elliptical conical horn feed illuminator.

Function

To provide a highly reliable redundant antenna for transmission of signals from maximum range at reduced data rate.

Performance Characteristics

Maximum gain (relative to circular isotropic)	$28.0^{+0.25}_{-0.50}$ db at 2295 mc
	$27.5^{+0.25}_{-0.50}$ db at 2115 mc
Polarization	Right-hand circular
Ellipticity on axis	1.0 ± 1.0 at 2295 mc
	2.0 ± 1.0 at 2115 mc
Impedance	50 ohm nominal
VSWR	1.2:1 maximum at 2295 mc
	1.4:1 maximum at 2115 mc
Half-power beamwidth	4.0 ± 1.0 degrees
	10.0 ± 1.0 degrees

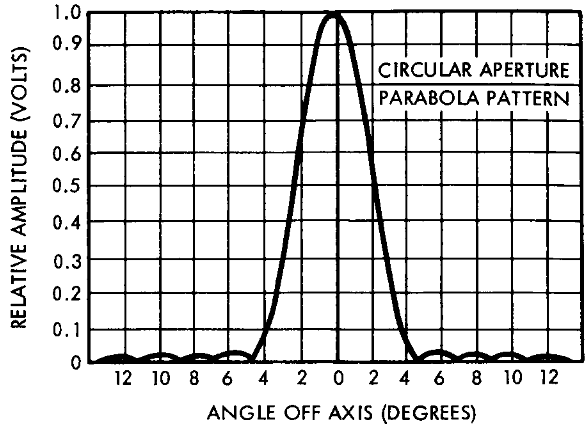
Physical Characteristics

Antenna size	94 x 36 inches
Focal length	30.65 inch
Total weight	13.44 pounds
Connector	Type N male aluminum
Coaxial cable	Semi-rigid aluminum 3/8 O.D.
Mounting	Single axis gimbal drive

Figure 65. Preliminary Specification — S-Band Medium Gain Antenna

PRELIMINARY SPECIFICATION

S-Band High Gain Antenna



Antenna Characteristics

Configuration	A double-gimballed high-gain circular paraboloidal reflector with a conical horn feed illuminator.

Function

To provide a means of focusing the transmitted r.f. signals into a collimated beam for communication from maximum range at all nominal spacecraft attitudes.

Performance Characteristics

Maximum gain (relative to circular isotropic)	$34.0^{+0.25}_{-0.50}$ db at 2295 Mc
	$33.5^{+0.25}_{-0.50}$ db at 2115 Mc
Polarization	Right-hand circular
Ellipticity on axis	1.0 ± 1.0 db at 2295 Mc
	2.0 ± 1.0 db at 2115 Mc
Impedance	50 ohm nominal
VSWR	Less than 1.2:1 at 2295 Mc
	Less than 1.4:1 at 2115 mc
Half-power beamwidth	3.4 ± 0.3 degrees

Physical Characteristics

Antenna size	114 x 114 inches
Focal length	45.6 inches
Weight	
Antenna assembly	32.70 pounds
Boom assembly	22.55 pounds
Total	55.25 pounds
Connector	Type N male aluminum
Coaxial cable	Semi-rigid aluminum 3/8 O D
Mounting	2 axis gimbal drive

Figure 66. Preliminary Specification — S-Band High Gain Antenna

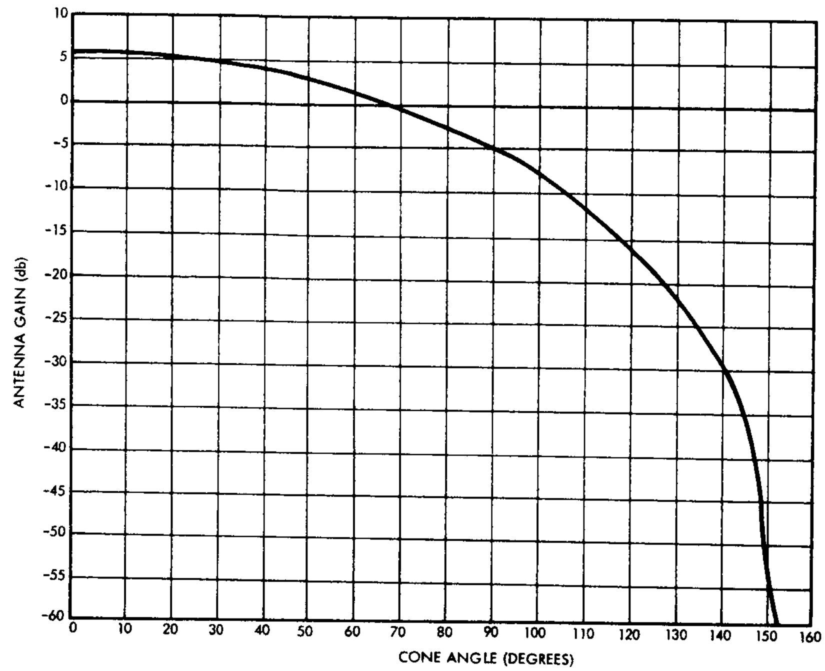


Figure 67. Low-Gain Antenna Gain vs. Cone Angle for 2295 Mc.

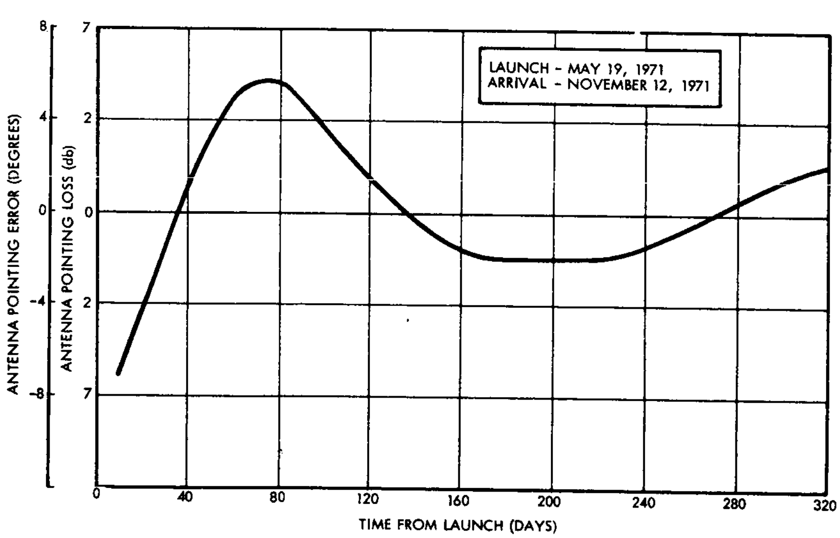


Figure 63. Medium-Gain Antenna Loss vs. Time for Nominal Trajectory

Table 32. Spacecraft Radio Transmission Parameters[1] (2295 mc ± 5 mc)

Mode	Parameter	Low-Gain Antenna Value	Low-Gain Antenna Tolerance	Medium-Gain Antenna Value	Medium-Gain Antenna Tolerance	High-Gain Antenna Value	High-Gain Antenna Tolerance
High Power Mode	Total Transmitted Power (50 w)	47.0 dbm	+1.0 -0.0 db	47.0 dbm	+1.0 -0.0 db	47.0 dbm	+1.0 -0.0 db
	Transmission Circuit Loss[2]	2.65 db	±0.8 db	2.30 db	±0.8 db	2.05 db	±0.8 db
	Antenna Gain[3]	4.2 db	±0.50 db	28.0 db	+0.25 -0.5 db	34.0 db	+0.25 -0.50 db
	Antenna Pointing Loss	Figure	-	Figure	-	0.4 db	+0.4 -0.3 db
Low Power Mode	Total Transmitted Power (1 w)	30.0 dbm	+1.0 -0.0 db	30.0 dbm	+1.0 -0.0 db	30.0 dbm	+1.0 -0.0 db
	Transmission Circuit Loss	1.6 db	±0.8 db	1.95 db	±0.8 db	2.40 db	±0.8 db

[1] These parameters were obtained from the requirements listed in this section.

[2] Includes all circuitry between the output of the amplifier and the input to the antenna.

[3] Referenced to perfectly circular isotropic with matched polarization, pattern maximum.

Table 33. Radio Reception Parameters for Standard DSIF Station

	Early Flight Acquisition			Tracking, Diplexed, Maser 85-foot diameter antenna			Tracking, Diplexed, Maser 210-foot diameter antenna		
	Value	Tolerance	JPL Doc. No.	Value	Tolerance	JPL Doc. No.	Value	Tolerance	JPL Doc. No.
Antenna gain[1]	21.0 db	±1.0 db	MC-4-310A	53.0 db	+1.0 -0.5 db	EPD-283	61.7 db	+1.0 -0.5 db	Note 2
Antenna pointing loss	0.0 db	0.0 db	MC-4-310A	0.0 db	±0.0 db	MC-4-310A	0.5 db	--	Note 3
Circuit loss[4]	0.5 db	±0.2 db	TM-33-83	0.2 db	±0.1 db	TM-33-83	0.2 db	±0.1 db	
Effective system noise temp.[5]	270°K	±60°K	MC-4-310A	55°K	±10°K	TM-33-83	30°K	±5°K	Note 2
Antenna ellipticity	<1.5 db	--	MC-4-310A	0.7 db	±0.3 db	MC-4-310A	0.3 db	±0.2 db	
Carrier PLL noise bandwidth ($2B_{LO}$)	12, 48, 152 cps	+0.5 -0.0 db	MC-4-310A	12, 48, 152 cps	+0.5 -0.0 db	MC-4-310A	12, 48, 152 cps	+0.5 -0.0 db	MC-4-310A
Carrier threshold SNR in $2B_{LO}$									
a. One-way doppler tracking	0.0 db	±0.0 db	MC-4-310A	0.0 db	±0.0 db	MC-4-310A	0.0 db	±0.0 db	MC-4-310A
b. Two-way doppler tracking	2.0 db	±1.0 db	MC-4-310A	2.0 db	±1.0 db	MC-4-310A	2.0 db	±1.0 db	MC-4-310A
c. Telemetering	6.0 db	+0.5 -1.0 db	MC-4-310A	6.0 db	+0.5 -1.0 db	MC-4-310A	6.0 db	+0.5 -1.0 db	MC-4-310A

[1] To matched polarization

[2] Worst-case values were specified in EPD-283

[3] Computed from specified beamwidth, 0.1° (EPD-283), and specified pointing error, 0.02° (EPD-283).

[4] Circuit losses include diplexer, switch, and waveguide losses.

[5] Includes contributions due to antenna zenith temperature, circuit losses, diplexer, low-noise amplifier, and follow-on receiver.

[6] When the carrier threshold SNR in $2B_{LO}$ is +3.8 db on the earth-to-spacecraft link, +2.0 db is required to overcome the ground receiver degradation.

Table 34. DSIF Radio Transmission Parameters (2115 ± 5 mc)

Parameter	Early Flight Acquisition			Tracking, Diplexed, 10 KW, 85-foot Antenna			Tracking, Diplexed, 100 KW, 210-foot Antenna [1]		
	Value	Tolerance	Source	Value	Tolerance	Source	Value	Tolerance	Source
Total transmitted power[2]	70.0 dbm	$^{+0.5}_{-0.0}$ db	MC-4-310A	70.0 dbm	$^{+0.5}_{-0.0}$ db	MC-4-310A	80.0 dbm	--	
Transmission[3] circuit loss	0.5 db	±0.2 db	MC-4-310A	0.4 db	±0.1 db	MC-4-310A	0.4 db	±0.1 db	
Antenna gain[4]	20.0 db	±2.0 db	MC-4-310A	51.0 db	$^{+1.0}_{-0.5}$ db	MC-4-310A	60.6 db	$^{+1.0}_{-0.5}$ db	Note 5
Antenna ellipticity	<1.5 db	--	MC-4-310A	1.0 db	±0.5 db	MC-4-310A	0.3 db	±0.2 db	

[1] This capability does not exist at present; however, it is planned for Voyager (EPD-283).

[2] Miltiplexed operation reduces transmitter power by 4.56 db (EPD-283).

[3] Circuit loss includes diplexer, switch, and waveguide losses.

[4] To matched polarization.

[5] Worst-case parameter was specified in EPD-283.

Table 35. Spacecraft Radio Reception Parameters (2115 ± 5 mc)[1]

Parameter	Low-Gain Antenna		Medium-Gain Antenna		High-Gain Antenna	
	Value	Tolerance	Value	Tolerance	Value	Tolerance
Antenna gain[2]	5.0 db	±0.5 db	27.5 db	+0.25 -0.5 db	33.5 db	+0.25 -0.5 db
Antenna pointing loss	Fig.	--	Fig.	--	0.4 db	+0.4 -0.3 db
Receiving Circuit Loss[3]	1.26 db	±0.2 db	1.26 db	±0.2 db	1.36 db	±0.2 db
Effective System Noise Temperature[4]	2610°K	+1.12 -2.18 db	2610°K	+1.12 -2.18 db	2610°K	+1.12 -2.18 db
Carrier PLL noise bandwidth ($2B_{LO}$)	20 cps	+0.46 -0.41 db	20 cps	+0.46 -0.41 db	20 cps	+0.46 -0.41 db
Carrier threshold SNR in $2B_{LO}$						
a. Two-way Doppler tracking[5]	3.8 db	--	3.8 db	--	3.8 db	--
b. Command Reception[6]	8.0 db	±1.0 db	8.0 db	±1.0 db	8.0 db	±1.0 db

[1] These parameters have been obtained from specifications listed in this section.

[2] Antenna gains referenced to perfectly circular isotropic.

[3] Includes all circuitry between the antenna and the input to the receiver.

[4] Includes contribution due to antenna temperature, circuit losses, and noise figure at input to preselector.

[5] 3.8 db SNR is required for a 2.0 db ground receiver degradation.

[6] Based upon command threshold definition.

Table 36. Telemetry Parameters

				Source
1.	Subcarrier modulation method	Square wave PSK data channel plus square wave PN sync channel		
2.	Transmission rates	15,000; 7,500; 3,750; 1,875; 234-3/8; 7.3 b/sec		
3.	Required $\frac{ST}{N/B}$ for bit error probability of $P_e^b = 5 \times 10^{-3}$	Theoretical	$5.2 \frac{db \cdot cps}{b/sec}$	
		Carrier degradation	1.0 db	
		Subcarrier degradation	0.5 db	
		Bit sync degradation	0.1 db	
		Filtering of square wave	0.9 db	
		Total	$7.7 \frac{db \cdot cps}{b/sec}$ ±1.0 db	
4.	Modulation indices			
	a. For bit rates 15,000; 7,500; 3,750; 1.875 b/sec $\theta_D = 1.25$ RAD ±5%, $\theta_S = 0.32$ RAD ±5%			
	b. For bit rate 234-3/8 b/sec (cruise mode) $\theta_D = 1.2$ RAD ±5%, $\theta_S = 0.4$ RAD ±5%			
	c. For bit rate 234-3/8 b/sec (launch mode) $\theta_D = 0.87$ RAD ±5%, $\theta_S = 0.3$ RAD ±5%			
	d. For bit rate, 7.3 b/sec $\theta_D = 0.76$ RAD ±5%, $\theta_S = 0.71$ RAD ±5%			
5.	Modulation Losses			
	a. Bit rates 15,000; 7,500; 3,750; 1,875 b/sec			
	Carrier modulation loss	10.47 db $^{+1.51}_{-1.90}$ db		
	Data modulation loss	0.91 db $^{+0.19}_{-0.24}$ db		
	Sync modulation loss	20.06 db $^{+1.87}_{-2.29}$ db		
	b. Bit rate 234-3/8 b/sec (cruise mode)			
	Carrier modulation loss	9.53 db $^{+1.31}_{-1.56}$ db		
	Data modulation loss	1.32 db $^{+0.22}_{-0.31}$ db		
	Sync modulation loss	17.01 db $^{+1.64}_{-1.89}$ db		
	c. Bit rate 234-3/8 b/sec (launch mode)			
	Carrier modulation loss	4.24 db $^{+0.48}_{-0.52}$ db		
	Data modulation loss	2.72 db ±0.36 db		
	Sync modulation loss	14.42 db $^{+0.82}_{-0.90}$ db		
	d. Bit rate 7.3 b/sec			
	Carrier modulation loss	5.20 db $^{+0.58}_{-0.65}$ db		
	Data modulation loss	5.64 db $^{+0.64}_{-0.69}$ db		
	Sync modulation loss	6.52 db $^{+0.71}_{-0.76}$ db		
6.	Sync channel threshold SNR in $2B_{LO}$	18.4 db ±1.0 db		TR-32-495
7.	Sync channel effective noise bandwidth	0.5 cps ±0.4 db		MC-4-310A

Table 37. Command Parameters

Parameter	Value	Source
1. Subcarrier modulation method	Sine wave PSK command channel plus square wave PN sync	
2. Transmission rate	1 bit/sec	
3. Required carrier SNR in 20 cps bandwidth at command threshold	8.0 db ±1.0 db	MC-4-310A
4. Required command channel ST/N/B for bit error probability $P_e^b = 1 \times 10^{-6}$	15.8 db ±1.0 db	Section 7.5
5. Required sync channel SNR at threshold	13.7 db ±1.0 db	Section 7.5
6. Sync channel effective noise bandwidth	2.0 cps ±0.8 db	Section 7.5
7. Modulation indices	$\theta_D = 0.725$ RAD ±10%, $\theta_S = 0.55$ RAD ±10%	
8. Modulation losses		
a. Carrier modulation loss	2.57 db $^{+0.27}_{-0.28}$ db	
b. Data modulation loss	7.77 db $^{+0.51}_{-0.54}$ db	
c. Sync modulation loss	6.82 db $^{+0.53}_{-0.56}$ db	

Table 38. Range Code Parameters

Parameter	Value	Source
1. Subcarrier modulation method	Square wave PSK	
2. Clock rate	Approximately 500K bit/sec	
3. Required range code SNR in $2B_{LO} = 0.8$ cps (threshold)	+22 db ±1.0 db	MC-4-310A
4. Required carrier SNR in $2B_{LO}$ (threshold)	6.0 db $^{+0.5}_{-1.0}$ db	MC-4-310A
5. Uplink modulation losses ($\theta_u = 1.25$ radians)		
a. Carrier	10.02 db $^{+1.42}_{-1.75}$	
b. Range code	0.46 db ±0.18 db	
6. Downlink modulation losses with telemetry (234-3/8 bits/sec)[1]		
a. Carrier	-6.33 db $^{+0.93}_{-1.25}$ db	
b. Ranging	-7.79 db $^{+2.17}_{-3.67}$ db	
c. Data	-7.09 db $^{+0.99}_{-1.29}$ db	
d. Sync	-18.19 db $^{+1.29}_{-1.67}$ db	
7. Loop noise bandwidth		
a. Carrier	12 cps $^{+0.45}_{-0.0}$ db	MC-4-310A
b. Range code	0.8 cps $^{+0.46}_{-0.0}$ db	MC-4-310A
8. Sync channel loss during acquisition[2]	12 db	

[1] Based on effective range code modulation (θ_R') at maximum range of 0.70 radians and modulation loss due to uplink noise ($\sigma_n^2 = 0.246$ rad^2) of 1.07 db, with $\theta_D = 0.74$, $\theta_S = 0.25$

[2] Occurs during acquisition of X component.

Table 39. Maneuver Sequence (approximate times in minutes)

	Antennas Available	First Midcourse	Second Midcourse	Third Midcourse	Deboost	Capsule Separation
Maneuver preparations	Low Medium High	35	37	58	68	68
Point high-gain antenna	Low Medium	15	15	15	15	15
Verify antenna position	Low Medium	5	5.4	9.5	11.5	11.5
Enable the maneuver	Low Medium	5	5.4	9.5	11.5	11.5
Spacecraft maneuver	None	38	38	38	38	38
Verify spacecraft position	High only	5	5.4	9.5	11.5	11.5
Enable burn	High only	5	5.4	9.5	11.5	11.5 (enable separation)
Burn	High only	5	5	5	5	--
Reorient spacecraft and reacquire sun/ Canopus	None	46	46	46	46	46
Post maneuver sequence (verification)	Low Medium High	5	5.4	9.5	11.5	11.5

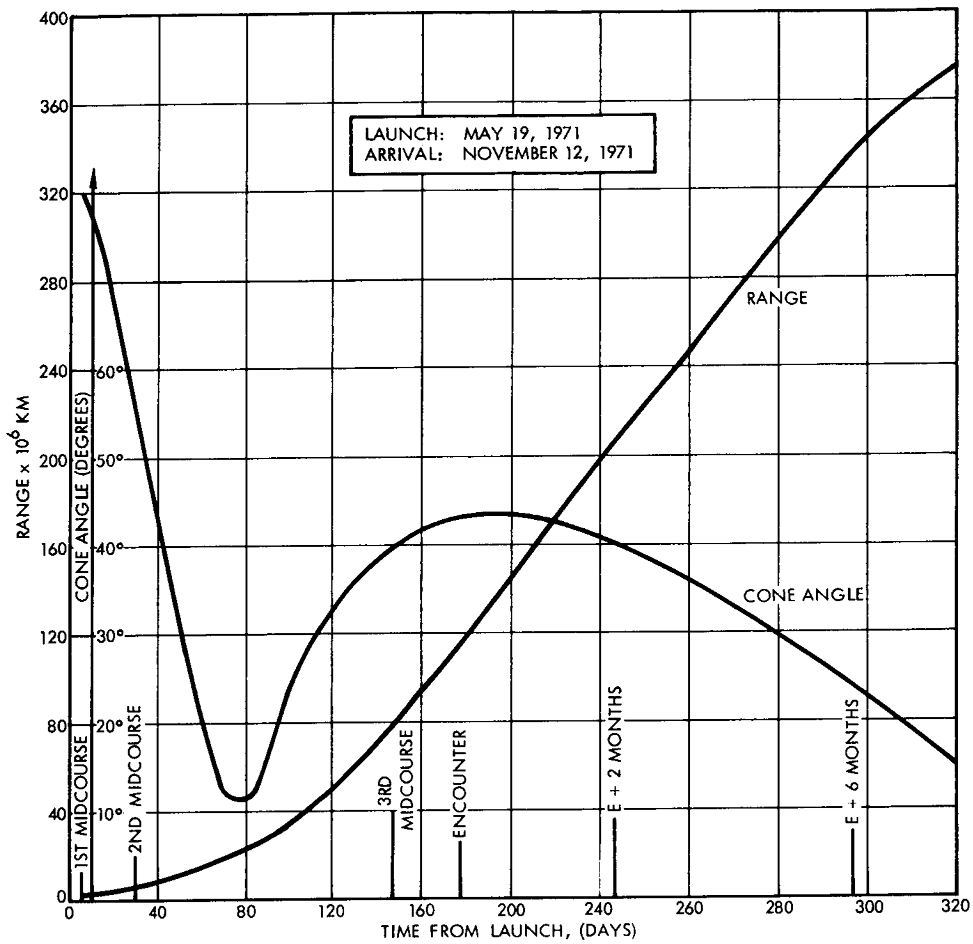


Figure 69. Communications Distance and Cone Angle vs. Time for Nominal Trajectory

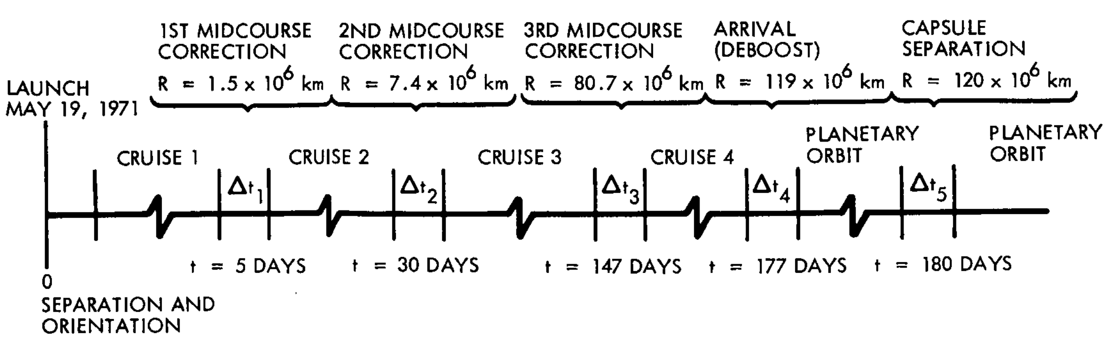


Figure 70. Typical Voyager Mission Maneuver Profile

b. Modulation Loss

The modulation loss for a given channel is that fraction of the total received power which does not appear as useful power in that channel due to the modulation process. For the telemetry link, where square-wave subcarriers are used for both the data and sync channels, these losses are given by:

- Carrier modulation loss = $\cos^2\theta_D \cos^2\theta_S$
- Data modulation loss = $\cos^2\theta_S \sin^2\theta_D$
- Sync modulation loss = $\cos^2\theta_D \sin^2\theta_S$

For the command link, which has a sinusoidal data subcarrier and square-wave sync subcarrier, the losses are given by:

- Carrier modulation loss = $J_0^2(\theta_D) \cos^2\theta_S$
- Data modulation loss = $2J_1^2(\theta_D)\cos^2\theta_S$
- Sync modulation loss = $J_0^2(\theta_D)\sin^2\theta_S$

where θ_D and θ_S are the modulation indices due to the data and sync channels, respectively.

The carrier modulation loss for the turnaround ranging link with and without telemetry is a function of the signal-to-noise ratio received at the spacecraft and is treated in Phase IA Report, Volume 5, Appendix F. Tolerances on the modulation loss are computed by assuming that θ_D and θ_S are independent.

c. Performance Margin

The performance margin for each mode of operation is the nominal available signal power in a given channel (carrier, data, or sync), minus the nominal power required in that channel for threshold. Since the various channels do not necessarily threshold together, their performance margins may differ. When they differ, the performance margins for the controlling channel are plotted.

d. Nominal Received Signal Level

The nominal received signal level for a given channel is the total received power determined from the nominal values in the telecommunications design control table (item 11), minus the nominal modulation loss for that channel.

e. Nominal Threshold Signal Level

The nominal threshold signal level is the signal level required to satisfy some minimum performance criteria as determined from the nominal values for noise spectral density, bandwidth, communication efficiency, etc., in the telecommunications design control table.

f. Parameter Tolerances

In tabulating the parameter values in the telecommunications design control tables presented in 3.5.3, the positive and negative tolerances have been assigned in a manner such that tolerances appearing in the negative column always denote a decrease in the signal-to-noise ratio and those appearing in the positive column denote an increase.

3.5.3 Link Performance Summary

The link performance for various communication modes is summarized in this section in the form of telecommunications design control tables and plots of performance margin versus time or range. The plots are presented only for the nominal trajectory, while the design control tables include performance calculations for worst-case (maximum communication range) trajectories.

a. Spacecraft-to-Earth Link

Figure 71 presents the telemetry link performance as a function of range for most possible modes of operation. The low- and medium-gain antenna pointing losses are not included, since such losses are dependent on the particular trajectory. For the nominal trajectory, the performance margins versus time from launch (for communication modes of major interest) are shown in Figure 72. The nominal mode of operation is indicated as a heavy line. Antenna pointing errors for the low- and medium-gain antennas are included in this figure. The performance margin and sum of the adverse tolerances presented in Figures 71 and 72 are for the data channel, since this is the

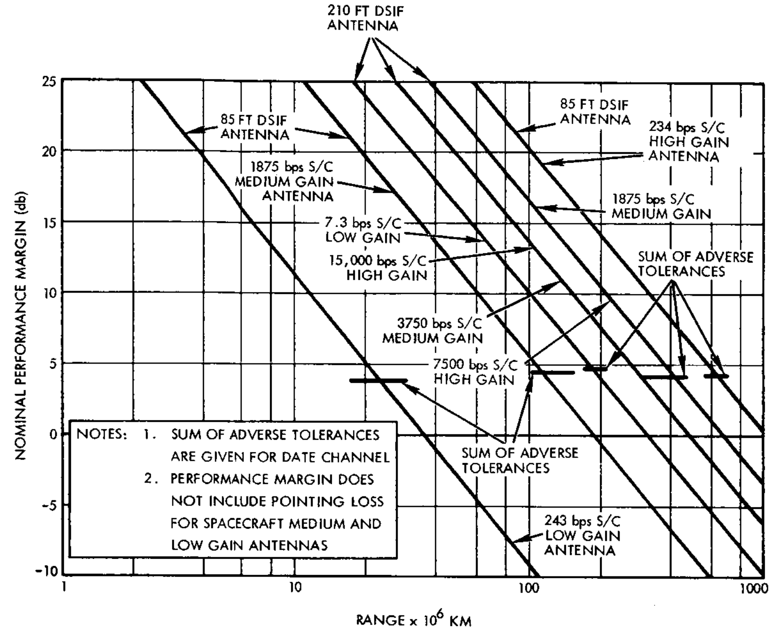


Figure 71. Spacecraft-to-Earth Performance vs. Range

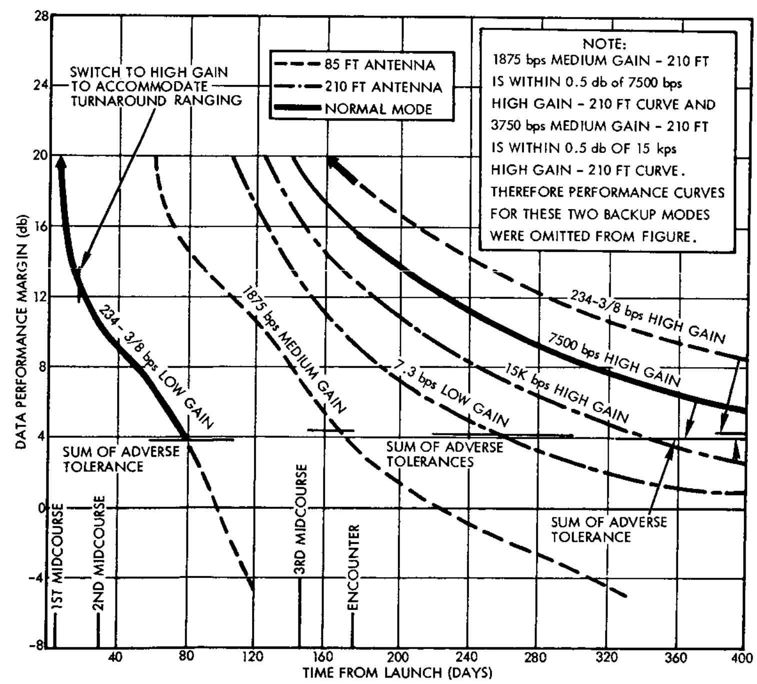


Figure 72. Spacecraft-to-Earth Channel Performance Margin vs. Time for Nominal Trajectory

controlling channel for bit rates above 234 bits/sec. For the bit rates of 234 and 7.3 bits/sec, the nominal performance margin for all three channels is the same; however, the sum of adverse tolerances is somewhat higher for the sync channel.

Figure 72 shows that for the nominal trajectory it will be necessary to switch to high-gain antenna transmit approximately 75 days after launch (if turnaround ranging is desired this will occur about 15 days from launch). The 15 kb/sec capability will exist to about encounter plus 5 months with the 210-foot receiving antenna. For the remainder of the nominal mission 7.5 kb/sec capability using the 210-foot antenna will be available. Figure 72 also illustrates that the medium-gain antenna and 7.3 bit/sec transmission rates are adequate backup modes for the mission.

The telemetry channel performance for the worst-case encounter (worst-case trajectory) and the worst-case encounter plus 6 months conditions are presented in Tables 40 through 42. Tables 40 and 41 indicate that nominal modes of operation will be possible even for the worst-case trajectory. Table 42 shows that the emergency bit rate capability, 7.3 bits/sec, using the low-gain antenna, will exist to worst-case encounter.

Figures 73 and 74 present telemetry channel performance versus time for the early portion of the mission, while Tables 43 and 44 give performance parameters used.

Station 71, Kennedy Space Center, is able to receive telemetry to 0-degree elevation. The acquisition aid DSIF station mode of operation can be used to about 40 minutes after injection, as indicated in Figure 64. By this time the DSIF station is able to switch reception to the 85-foot dish.

Performance margins for the first midcourse and the deboost maneuvers are given in Figures 75 and 76, respectively. Maneuver sequence times are given in Table 39. and maneuver ranges are presented in Figure 70.

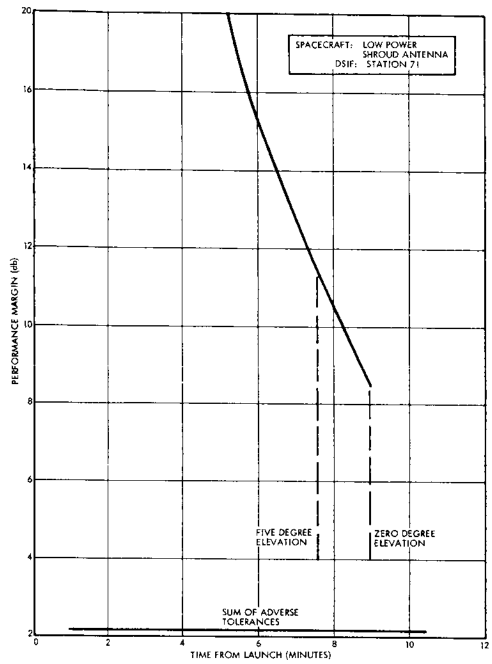


Figure 73. Launch Mode Telemetry Performance Margin vs. Time

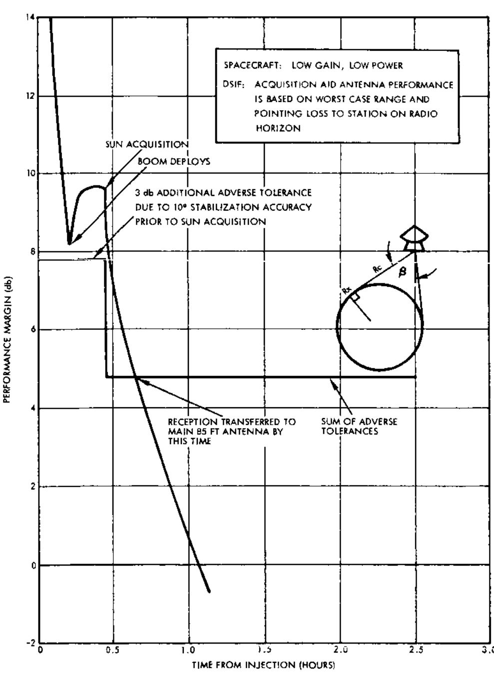


Figure 74. Injection Mode Telemetry Performance Margin vs. Time

Table 40. Telecommunications Design Control

Channel: Spacecraft-to-Earth
Mode: 234-3/8 bits/sec, Spacecraft High-Gain Antenna, 85-foot DSIF Antenna

No.	Parameter	Value	Tolerance +	Tolerance -	Source
1	Total transmitter power	47.0 dbm	1.0	0.0	Note 1
2	Transmitting circuit loss	-2.05 db	0.8	0.8	
3	Transmitting antenna gain	34.0 db	0.25	0.5	
4	Transmitting antenna pointing loss	-0.4 db	0.4	0.3	
5	Space loss (worst case Eng.) 2295 Mc $R = 180 \times 10^6$ km	-264.77 db	0.0	0.0	
6	Polarization loss ϵ(DSIF) = 1.5 db ϵ(S/C) = 2 db	-0.13 db	-	-	Note 2
7	Receiving antenna gain	53.0 db	1.0	0.5	Note 3
8	Receiving antenna pointing loss	0.0 db	0.0	0.0	
9	Receiving circuit loss	-0.2 db	0.1	0.1	
10	Net circuit loss	-180.55 db	2.55	2.2	
11	Total received power	-133.55 dbm	3.55	2.2	
12	Receiver noise spectral density (N/B) T System = $55^oK \pm 10^oK$	-181.2 dbm/cps	0.9	0.7	
13	Carrier modulation loss	-9.53 db	1.31	1.56	
14	Received carrier power	-143.08 dbm	4.86	3.76	
15	Carrier APC noise bandwidth ($2B_{LO}$ = 12 cps)	10.8 db·cps	0.5	0.0	
	Carrier Performance Tracking (one-way)				
16	Threshold SNR in $2B_{LO}$	0.0 db	0.0	0.0	
17	Threshold carrier power	-170.4 dbm	1.4	0.7	
18	Performance margin	27.32 db	6.26	4.46	
	Carrier Performance Tracking (two-way)				
19	Threshold SNR in $2B_{LO}$	2.0 db	1.0	1.0	
20	Threshold carrier power	-168.4 dbm	2.4	1.7	
21	Performance margin	25.32 db	7.26	5.46	
	Carrier Performance				
22	Threshold SNR in $2B_{LO}$	6.0 db	0.5	1.0	
23	Threshold carrier power	-164.4 dbm	1.9	1.7	
24	Performance Margin	21.32 db	6.76	5.46	
	Data Channel				
25	Modulation loss	-1.32 db	0.22	0.31	
26	Received data subcarrier power	-134.87 dbm	3.77	2.51	
27	Bit rate (1/T) (234-3/8 bits/sec)	23.7 db·cps	0.0	0.0	
28	Required ST/N/B ($Pe^b = 5 \times 10^{-3}$)	7.7 db	1.0	1.0	
29	Threshold subcarrier power	-149.8 dbm	1.9	1.7	
30	Performance margin	14.93 db	5.67	4.21	
	Sync Channel				
31	Modulation loss	-17.01 db	1.64	1.89	
32	Received sync subcarrier power	-150.56 dbm	5.19	4.09	
33	Sync. APC noise bandwidth ($2B_{LO}$ = 0.5 cps)	-3.01 db·cps	0.4	0.4	
34	Threshold SNR in $2B_{LO}$	18.4 db	1.0	1.0	
35	Threshold subcarrier power	-165.81 dbm	2.3	2.1	
36	Performance margin	15.25 db	7.49	6.19	

Notes: 1. Sources for spacecraft parameters are specifications given in section 3.0 of this volume.
2. Only maximum polarization loss was computed.
3. Sources for the DSIF parameters are indicated in tables 3.6 and 3.7.

Table 41. Telecommunications Design Control

Channel: Spacecraft-to-Earth
Mode: 7,500 bits/sec, Spacecraft High-Gain Antenna, 210-foot DSIF Antenna

No.	Parameter	Value	Tolerance +	Tolerance -	Source
1	Total transmitter power	47.0 dbm	1.0	0.0	Note 1
2	Transmitting circuit loss	-2.05 db	0.8	0.8	
3	Transmitting antenna gain	34.0 db	0.25	0.5	
4	Transmitting antenna pointing loss	-0.4 db	0.4	0.3	
5	Space loss (worst case E + 6 months) 2295 Mc R = 390 x 10 km	-271.5 db	0.0	0.0	
6	Polarization loss ϵ(DSIF) = 0.5 db ϵ(S/C) = 2 db	-0.09 db	-	-	Note 2
7	Receiving antenna gain	61.7 db	1.0	0.5	Note 3
8	Receiving antenna pointing loss	-0.5 db	-	-	
9	Receiving circuit loss	-0.2 db	0.1	0.1	
10	Net circuit loss	-179.04 db	2.55	2.2	
11	Total received power	-132.04 dbm	3.55	2.2	
12	Receiver noise spectral density (N/B) T System = $30^{o}K \pm 5^{o}K$	-183.8 dbm/cps	0.8	0.7	
13	Carrier modulation loss	-10.47 db	1.51	1.9	
14	Received carrier power	-142.51 dbm	5.06	4.1	
15	Carrier APC noise bandwidth ($2B_{LO}$ = 12 cps)	10.8 db·cps	0.5	0.0	
Carrier Performance Tracking (one-way)					
16	Threshold SNR in $2B_{LO}$	0.0 db	0.0	0.0	
17	Threshold carrier power	-173.0 dbm	1.3	0.7	
18	Performance margin	30.49 db	6.36	4.8	
Carrier Performance Tracking (two-way)					
19	Threshold SNR in $2B_{LO}$	2.0 db	1.0	1.0	
20	Threshold carrier power	-171.0 dbm	2.3	1.7	
21	Performance margin	28.49 db	7.36	5.8	
Carrier Performance					
22	Threshold SNR in $2B_{LO}$	6.0 db	0.5	1.0	
23	Threshold carrier power	-167.0 dbm	1.8	1.7	
24	Performance Margin	24.49 db	6.86	5.8	
Data Channel					
25	Modulation loss	-0.91 db	0.19	0.24	
26	Received data subcarrier power	-132.95 dbm	3.74	2.44	
27	Bit rate (1/T) (7.5 k bits/sec)	38.75 db·cps	0.0	0.0	
28	Required ST/N/B ($Pe^{b} = 5 \times 10^{-3}$)	7.7 db	1.0	1.0	
29	Threshold subcarrier power	-137.35 dbm	1.8	1.7	
30	Performance margin	4.60 db	5.54	4.14	
Sync Channel					
31	Modulation loss	-20.06 db	1.87	2.29	
32	Received sync subcarrier power	-152.10 dbm	5.42	4.49	
33	Sync APC noise bandwidth ($2B_{LO}$ = 0.5 cps)	-3.01 db·cps	0.4	0.4	
34	Threshold SNR in $2B_{LO}$	18.4 db	1.0	1.0	
35	Threshold subcarrier power	-168.41 dbm	2.2	2.1	
36	Performance margin	16.31 db	7.62	6.59	

Notes: 1. Sources for spacecraft parameters are specifications given in section 3.0 of this volume
2. Only maximum polarization loss was computed
3. Sources for the DSIF parameters are indicated in tables 3.6 and 3.7

Table 42. Telecommunications Design Control

Channel: Spacecraft to DSIF

Mode: 7.3 bits/sec, Spacecraft Low-Gain Antenna, 210-foot DSIF Antenna

No.	Parameter	Value	Tolerance +	Tolerance -	Source
1	Total transmitter power	47.0 dbm	1.0	0.0	Note 1
2	Transmitting circuit loss	-2.65 db	0.8	0.8	
3	Transmitting antenna gain	4.2 db	0.5	0.5	
4	Transmitting antenna pointing loss	-1.5 db	0.0	0.0	
5	Space loss (worst case Encounter) 2295 MC R = 180 x 10^6 km	-264.77 db	0.0	0.0	
6	Polarization loss ϵ(DSIF) = 0.5 db ϵ(S/C) = 4 db	-0.28 db	-	-	Note 2
7	Receiving antenna gain	61.7 db	1.0	0.5	Note 3
8	Receiving antenna pointing loss	-0.5 db	-	-	
9	Receiving circuit loss	-0.2 db	0.1	0.1	
10	Net circuit loss	-204.0 db	2.4	1.9	
11	Total received power	-157.0 dbm	3.4	1.9	
12	Receiver noise spectral density (N/B) T System = 30°K ± 5°K	-183.8 dbm/cps	0.8	0.7	
13	Carrier modulation loss	-5.20 db	0.58	0.65	
14	Received carrier power	-162.2 dbm	3.98	2.55	
15	Carrier APC noise bandwidth ($2B_{LO}$ = 12 cps)	10.8 db·cps	0.5	0.0	
	Carrier Performance Tracking (one-way)				
16	Threshold SNR in $2B_{LO}$	0.0 db	0.0	0.0	
17	Threshold carrier power	-173.0 dbm	1.3	0.7	
18	Performance margin	10.8 db	5.28	3.25	
	Carrier Performance Tracking (two-way)				
19	Threshold SNR in $2B_{LO}$	2.0 db	1.0	1.0	
20	Threshold carrier power	-171.0 dbm	2.3	1.7	
21	Performance margin	8.8 db	6.28	4.25	
	Carrier Performance				
22	Threshold SNR in $2B_{LO}$	6.0 db	0.5	1.0	
23	Threshold carrier power	-167.0 dbm	1.8	1.7	
24	Performance Margin	4.8 db	5.78	4.25	
	Data Channel				
25	Modulation loss	-5.64 db	0.64	0.69	
26	Received data subcarrier power	-162.64 db	4.04	2.59	
27	Bit rate (1/T) (7.3 bits/sec)	8.65 db	0.0	0.0	
28	Required ST/N/B ($Pe^b = 5 \times 10^{-3}$)	7.7 db	1.0	1.0	
29	Threshold subcarrier power	-167.45 dbm	1.8	1.7	
30	Performance margin	4.81 db	5.84	4.29	
	Sync Channel				
31	Modulation loss	-6.52 db	0.71	0.76	
32	Received sync subcarrier power	-163.52 dbm	4.11	2.66	
33	Sync APC noise bandwidth ($2B_{LO}$ = 0.5 cps)	-3.01 db·cps	0.4	0.4	
34	Threshold SNR in $2B_{LO}$	18.4 db	1.0	1.0	
35	Threshold subcarrier power	168.41 dbm	2.2	2.1	
36	Performance margin	4.89 db	6.31	4.76	

Notes:
1. Sources for spacecraft parameters are specifications given in section 3.0 of this volume
2. Only maximum polarization loss was computed
3. Sources for the DSIF parameters are indicated in tables 3.6 and 3.7

Table 43. Telecommunications Design Control

Channel: Spacecraft-to-Earth
Mode: 234-3/8 (bits/sec), DSIF 71 (launch mode)

No.	Parameter	Value	Tolerance +	Tolerance −	Source
1	Total transmitter power	30.0 dbm	1.0	0.0	Note 1
2	Transmitting circuit loss	-1.6 db	0.8	0.8	
3	Transmitting antenna gain (worst case)	-6.0 db	-	-	
4	Transmitting antenna pointing loss (included in item 3)	-	-	-	
5	Space loss 2295 Mc R = 1500 km	-163.2 db	-	-	
6	DSIF antenna linearly polarized				
	Polarization loss ϵ(spacecraft) = 4 db	-5.4 db	-	-	Note 2
7	Receiving antenna gain	25.0 db	-	-	EPD-283
8	Receiving antenna pointing loss	0.0 db	0.0	0.0	
9	Receiving circuit loss (included in item 7)	-	-	-	
10	Net circuit loss	-151.2 db	0.8	0.8	
11	Total received power	-121.2 dbm	1.8	0.8	
12	Receiver noise spectral density (N/B)	-163.83 dbm/cps	-	-	EPD-283
	T System = 3000°K (worst case)				
13	Carrier modulation loss	-4.24 db	0.48	0.52	
14	Received carrier power	-125.44 dbm	2.28	1.32	
15	Carrier APC noise bandwidth ($2B_{LO}$ = 152 cps)	21.82 db · cps	0.5	0.0	EPD-283
Carrier Performance Tracking (one-way)					
16	Threshold SNR in $2B_{LO}$	0.0 db	0.0	0.0	
17	Threshold carrier power	-142.01 dbm	0.5	0.0	
18	Performance margin	16.57 db	2.78	1.32	
Carrier Performance Tracking (two-way)					
19	Threshold SNR in $2B_{LO}$	2.0 db	1.0	1.0	
20	Threshold carrier power	-140.01 dbm	1.5	1.0	
21	Performance margin	14.57 db	3.78	2.32	
Carrier Performance					
22	Threshold SNR in $2B_{LO}$	6.0 db	0.5	1.0	
23	Threshold carrier power	-136.01 dbm	1.0	1.0	
24	Performance Margin	10.57 db	3.28	2.32	
Data Channel					
25	Modulation loss	-2.72 db	0.36	0.36	
26	Received data subcarrier power	-123.92 dbm	2.16	1.16	
27	Bit rate (1/T) (234-3/8 bits/sec)	23.7 db	0.0	0.0	
28	Required ST/N/B ($P_e^b = 5 \times 10^{-3}$)	7.7 db	1.0	1.0	
29	Threshold subcarrier power	-132.43 dbm	1.0	1.0	
30	Performance margin	8.51 db	3.16	2.16	
Sync Channel					
31	Modulation loss	-14.42 db	0.82	0.90	
32	Received sync subcarrier power	-135.62 dbm	2.62	1.70	
33	Sync APC noise bandwidth ($2B_{LO}$ = 0.5 cps)	-3.01 db cps	0.4	0.4	
34	Threshold SNR in $2B_{LO}$	18.4 db	1.0	1.0	
35	Threshold subcarrier power	-148.44 dbm	1.4	1.4	
36	Performance margin	12.82 db	4.02	3.1	

Notes: 1. Sources for spacecraft parameters are specifications given in section 3.0 of this volume.
2. Only maximum polarization loss was computed.
3. Sources for the DSIF parameters are indicated in tables 3.6 and 3.7.

Table 44. Telecommunications Design Control

Channel: Spacecraft-to-Earth
Mode: 234-3/8 bits/sec, Acquisition Aid Station Mode

No.	Parameter	Value	Tolerance +	Tolerance -	Source
1	Total transmitter power	30.0 dbm	1.0	0.0	Note 1
2	Transmitting circuit loss	-1.6 db	0.8	0.8	
3	Transmitting antenna gain	4.2 db	0.5	0.5	
4	Transmitting antenna pointing loss	-9.5 db	-	-	
5	Space loss 2295 Mc $R = 6 \times 10^3$ km	-175.23 db	-	-	
6	Polarization loss ϵ(S/C) = 4 db ϵ(DSIF) < 1.5 db	-0.34 db	-	-	Note 2
7	Receiving antenna gain	21.0 db	1.0	1.0	
8	Receiving antenna pointing loss	0.0 db	0.0	0.0	
9	Receiving circuit loss	-0.5 db	0.2	0.2	
10	Net circuit loss	-161.97 db	2.5	2.5	
11	Total received power	-131.97 dbm	3.5	2.5	
12	Receiver noise spectral density (N/B) T System = $270^{\circ}K \pm 60^{\circ}K$	-174.3 dbm/cps	1.1	0.9	
13	Carrier modulation loss	-4.24 db	0.48	0.52	
14	Received carrier power	-136.21 dbm	3.98	3.02	
15	Carrier APC noise bandwidth ($2B_{LO}$ = 152 cps)	21.82 db·cps	0.5	0.0	
	Carrier Performance Tracking (one-way)				
16	Threshold SNR in $2B_{LO}$	0.0 db	0.0	0.0	
17	Threshold carrier power	-152.48 dbm	1.6	0.9	
18	Performance margin	16.27 db	5.58	3.92	
	Carrier Performance Tracking (two-way)				
19	Threshold SNR in $2B_{LO}$	2.0 db	1.0	1.0	
20	Threshold carrier power	-150.48 dbm	2.6	1.9	
21	Performance margin	14.27 db	6.58	4.92	
	Carrier Performance				
22	Threshold SNR in $2B_{LO}$	6.0 db	0.5	1.0	
23	Threshold carrier power	-146.48 dbm	2.1	1.9	
24	Performance Margin	10.27 db	6.08	4.92	
	Data Channel				
25	Modulation loss	-2.72 db	0.36	0.36	
26	Received data subcarrier power	-134.69 dbm	3.86	2.86	
27	Bit rate (1/T) (234-3/8 bits/sec	23.7 db	0.0	0.0	
28	Required ST/N/B ($P_e^b = 5 \times 10^{-3}$)	7.7 db	1.0	1.0	
29	Threshold subcarrier power	-142.9 dbm	2.1	1.9	
30	Performance margin	8.21 db	5.96	4.76	
	Sync Channel				
31	Modulation loss	-14.42 db	0.82	0.90	
32	Received sync subcarrier power	-146.39 dbm	4.32	3.4	
33	Sync APC noise bandwidth ($2B_{LO}$ = 0.5 cps)	-3.01 db·cps	0.4	0.4	
34	Threshold SNR in $2B_{LO}$	18.4 db	1.0	1.0	
35	Threshold subcarrier power	-158.91 dbm	2.5	2.3	
36	Performance margin	12.52 db	6.82	5.7	

Notes: 1. Sources for spacecraft parameters are specifications given in section 3.0 of this volume
2. Only maximum polarization loss was computed
3. Sources for the DSIF parameters are indicated in tables 3.6 and 3.7

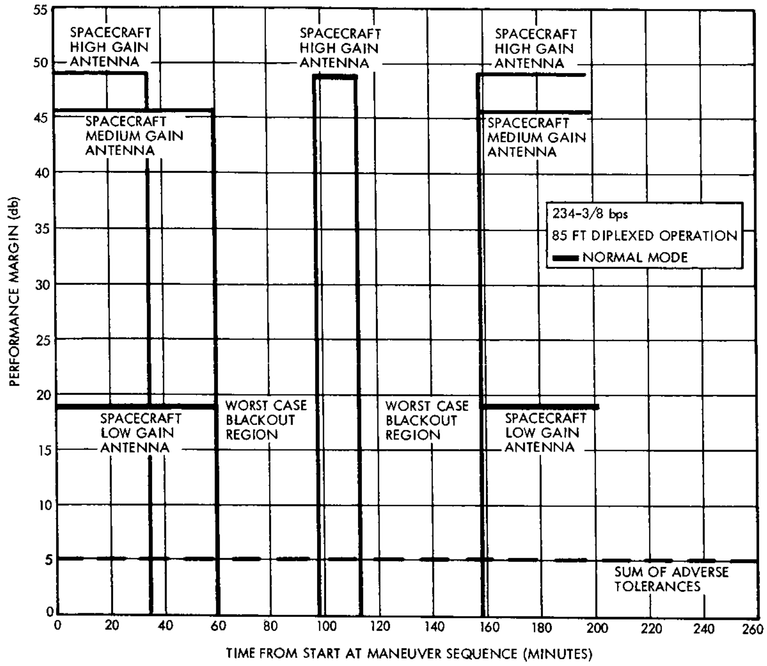


Figure 75. Spacecraft-to-Earth Performance Margin for Final Midcourse Maneuver, Nominal Trajectory

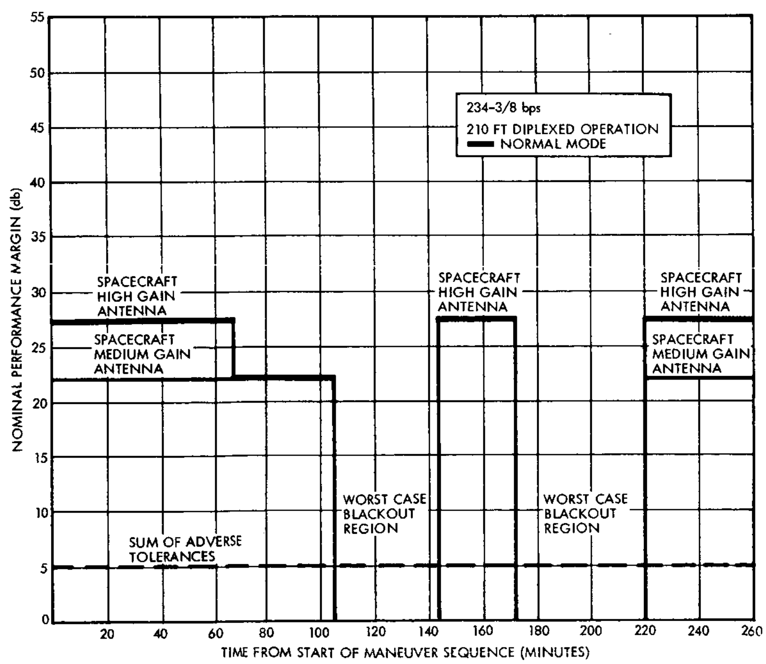


Figure 76. Spacecraft-to-Earth Performance Margin for Deboost Maneuver, Nominal Trajectory

b. Earth-to-Spacecraft Link

Figure 77 shows command performance margins versus range for most communication modes of interest. In Figure 78, the command channel performance for the nominal trajectory is presented. Low-gain antenna command reception capability is available during the entire mission. This will be possible, as indicated in Figure 77, by switching at 84 x 10^6 km ($\approx$150 days) from 85-foot diplexed 10-kw operation to either 85-foot 100-kw operation without diplexing or 210-foot diplexed 100-kw operation. Although the 210-foot diplexed 100 kw is not necessary for reliable command transmission, it is required for ranging near the end of mission. In addition, it provides a higher SNR and more rapid command synchronization acquisition. Also, the 210-foot antenna is required for telemetry reception after encounter, and it is operationally desirable to combine uplink and downlink capability in a single site (i.e., diplexed operation).

Tables 45 and 46 show command link performance at worst-case encounter, plus six months, with the 85-foot antenna, non-diplexed 100-kw transmitter, and with the 210-foot antenna, non-diplexed 100-kw transmitter, and with the 210-foot antenna, diplexed 100-kw transmitter. The command performance with medium-gain antenna for the worst-case encounter, plus six months, is given in Table 47 and indicates that the 85-foot antenna diplexed 10-kw site is adequate.

Table 48 shows that acquisition aid station modes can be used for transmitting commands during the first pass. The command link performance during the first and deboost maneuvers are shown in Figures 79 and 80.

c. Turnaround Ranging

The final three tables present turnaround ranging system (JPL Mark 1) performance. Table 49 shows that turnaround ranging with 234 bits/sec, multiplexed telemetry to worst-case encounter, plus six months, is possible with a 210-foot, 100-kw diplexed site.

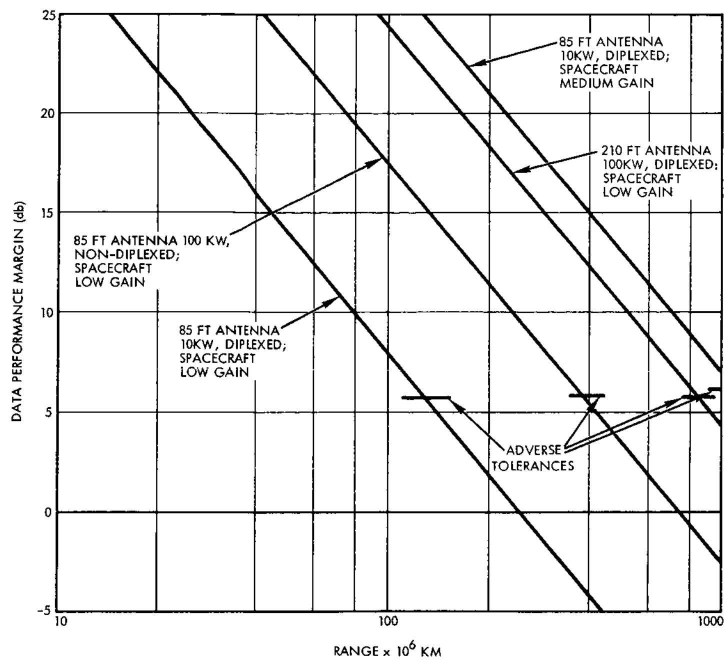


Figure 77. Earth-to-Spacecraft Performance Margin vs. Range

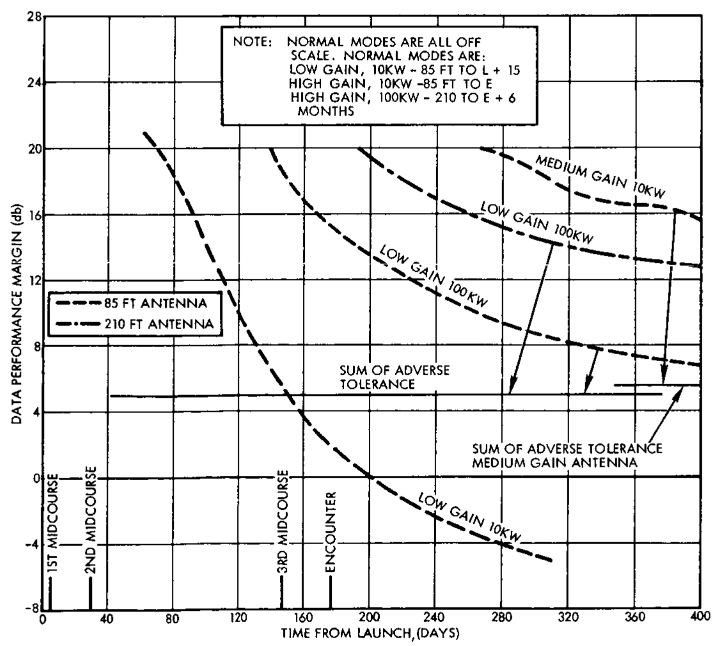


Figure 78. Earth-to-Spacecraft Channel Performance Margin vs. Time for Nominal Trajectory

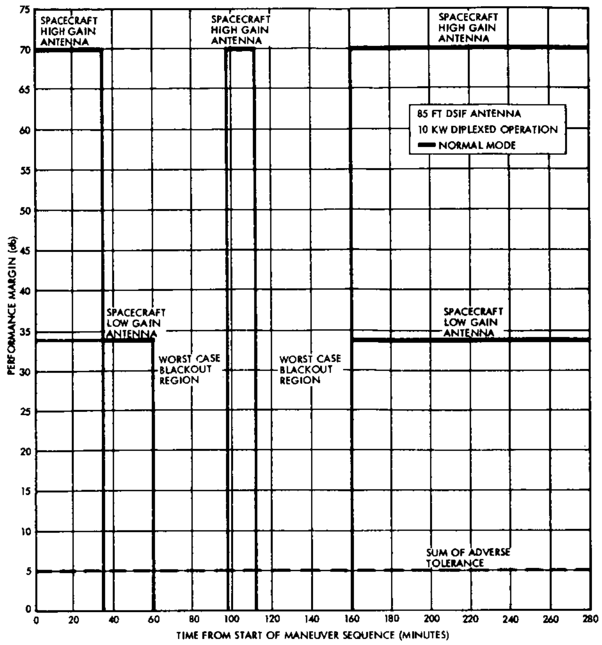


Figure 79. Earth-to-Spacecraft Link Performance Margin for First Midcourse Maneuver, Nominal Trajectory

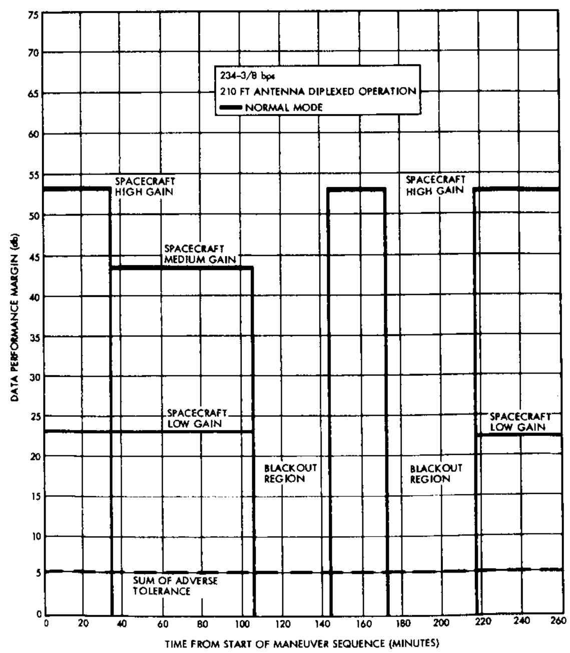


Figure 80. Earth-to-Spacecraft Performance Margin for Deboost Maneuver, Nominal Trajectory

Table 45. Telecommunications Design Control

Channel: Earth-to-Spacecraft
Mode: 85-foot DSIF Antenna, 100 kw Nondiplexed, Spacecraft Low-Gain Antenna

No.	Parameter	Value	Tolerance +	Tolerance −	Source
1	Total transmitter power	80.0 dbm	-	-	Note 3
2	Transmitting circuit loss	-0.2 db	0.1	0.1	
3	Transmitting antenna gain	53.0 db	1.0	0.5	
4	Transmitting antenna pointing loss	0.0 db	0.0	0.0	
5	Space loss (worst case E + 6 months) 2115 Mc $R = 390 \times 10^6$ km	-270.77 db	0.0	0.0	
6	Polarization loss ϵ(DSIF) = 0.5 db ϵ(S/C) = 6 db	-0.54 db	-	-	Note 2
7	Receiving antenna gain	5.0 db	0.5	0.5	Note 1
8	Receiving antenna pointing loss Cone angle = 12°	-0.5 db	-	-	
9	Receiving circuit loss	-1.26 db	0.2	0.2	
10	Net circuit loss	-215.27 db	1.8	1.3	
11	Total received power	-135.27 dbm	1.8	1.3	
12	Receiver noise spectral density (N/B) T System = 2610°K	-164.44 dbm/cps	1.12	2.18	
13	Carrier modulation loss	-2.57 db	0.27	0.28	
14	Received carrier power	-137.84 dbm	2.07	1.58	
15	Carrier APC noise bandwidth ($2B_{LO}$ = 20 cps)	13.01 db·cps	0.46	0.41	
	Carrier Performance Tracking (one-way)				
16	Threshold SNR in $2B_{LO}$	0.0 db	0.0	0.0	
17	Threshold carrier power	-151.43 dbm	1.58	2.59	
18	Performance margin	13.59 db	3.65	4.17	
	Carrier Performance Tracking (two-way)				
19	Threshold SNR in $2B_{LO}$	3.8 db	0.0	0.0	
20	Threshold carrier power	-147.63 dbm	1.58	2.59	
21	Performance margin	9.79 db	3.65	4.17	
	Carrier Performance				
22	Threshold SNR in $2B_{LO}$	8.0 db	1.0	1.0	
23	Threshold carrier power	-143.43 dbm	2.58	3.59	
24	Performance Margin	5.59 db	4.65	5.17	
	Data Channel				
25	Modulation loss	-7.77 db	0.51	0.54	
26	Received data subcarrier power	-143.04 dbm	2.31	1.84	
27	Bit rate (1/T)	0.0 db	0.0	0.0	
28	Required ST/N/B ($Pe^b = 1 \times 10^{-6}$)	15.8 db	1.0	1.0	
29	Threshold subcarrier power	-148.64 dbm	2.12	3.18	
30	Performance margin	5.60 db	4.43	5.02	
	Sync Channel				
31	Modulation loss	-6.82 db	0.53	0.56	
32	Received sync subcarrier power	-142.09 dbm	2.33	1.86	
33	Sync APC noise bandwidth ($2B_{LO}$ = 2 cps)	3.01 db·cps	0.8	0.8	
34	Threshold SNR in $2B_{LO}$	13.7 db	1.0	1.0	
35	Threshold subcarrier power	-147.73 dbm	2.92	3.98	
36	Performance margin	5.64 db	5.25	5.84	

Notes: 1. Sources for spacecraft parameters are specifications given in section 3.0 of this volume.
2. Only maximum polarization loss was computed.
3. Sources for the DSIF parameters are indicated in tables 3.6 and 3.7.

Table 46. Telecommunications Design Control

Channel: Earth-to-Spacecraft
Mode: 210 Foot DSIF Antenna, 100 kw Diplexed, Spacecraft Low-Gain Antenna

No.	Parameter	Value	Tolerance +	Tolerance -	Source
1	Total transmitter power	80.0 dbm	-	-	Note 3
2	Transmitting circuit loss	- 0.4 db	0.1	0.1	
3	Transmitting antenna gain	60.6 db	1.0	0.5	
4	Transmitting antenna pointing loss	- 0.5 db	-	-	
5	Space loss (E + 6 M worst case) 2115 Mc $R = 390 \times 10^6$ km	-270.77 db	0.0	0.0	
6	Polarization loss ϵ(DSIF) = 0.5 db ϵ(S/C) = 6 db	- 0.54 db	-	-	Note 2
7	Receiving antenna gain	5.0 db	0.5	0.5	Note 1
8	Receiving antenna pointing loss Cone Angle = 12^o	- 0.5 db	-	-	
9	Receiving circuit loss	- 1.26 db	0.2	0.2	
10	Net circuit loss	-208.37 db	1.8	1.3	
11	Total received power	-128.37 dbm	1.8	1.3	
12	Receiver noise spectral density (N/B) T System = 2610^oK	-164.44 dbm/cps	1.12	2.18	
13	Carrier modulation loss	- 2.57 db	0.27	0.28	
14	Received carrier power	-130.94 dbm	2.07	1.58	
15	Carrier APC noise bandwidth ($2B_{LO}$ = 20 cps)	13.01 db · cps	0.46	0.41	
Carrier Performance Tracking (one-way)					
16	Threshold SNR in $2B_{LO}$	0.0 db	0.0	0.0	
17	Threshold carrier power	-151.43 dbm	1.58	2.59	
18	Performance margin	20.49 db	3.65	4.17	
Carrier Performance Tracking (two-way)					
19	Threshold SNR in $2B_{LO}$	3.8 db	0.0	0.0	
20	Threshold carrier power	-147.63 dbm	1.58	2.59	
21	Performance margin	16.69 db	3.65	4.17	
Carrier Performance					
22	Threshold SNR in $2B_{LO}$	8.0 db	1.0	1.0	
23	Threshold carrier power	-143.43 dbm	2.58	3.59	
24	Performance Margin	12.49 db	4.65	5.17	
Data Channel					
25	Modulation loss	- 7.77 db	0.51	0.54	
26	Received data subcarrier power	-136.14 dbm	2.31	1.84	
27	Bit rate (1/T)	0.0 db	0.0	0.0	
28	Required ST/N/B ($P_e^b = 1 \times 10^{-6}$)	15.8 db	1.0	1.0	
29	Threshold subcarrier power	-148.64 dbm	2.12	3.18	
30	Performance margin	12.50 db	4.43	5.02	
Sync Channel					
31	Modulation loss	- 6.82 db	0.53	0.56	
32	Received sync subcarrier power	-135.19 dbm	2.33	1.86	
33	Sync APC noise bandwidth ($2B_{LO}$ = 2 cps)	3.01 db · cps	0.8	0.8	
34	Threshold SNR in $2B_{LO}$	13.7 db	1.0	1.0	
35	Threshold subcarrier power	-147.73 dbm	2.92	3.98	
36	Performance margin	12.54 db	5.25	5.84	

Notes: 1. Sources for spacecraft parameters are specifications given in section 3.0 of this volume.
2. Only maximum polarization loss was computed.
3. Sources for the DSIF parameters are indicated in tables 3.6 and 3.7.

Table 47. Telecommunications Design Control

Channel: Earth-to-Spacecraft
Mode: 85-foot DSIF Antenna, 10 kw Diplexed, Spacecraft Medium-Gain Antenna

No.	Parameter	Value	Tolerance +	Tolerance -	Source
1	Total transmitter power	70.0 dbm	0.5	0.0	Note 3
2	Transmitting circuit loss	-0.4 db	0.1	0.1	
3	Transmitting antenna gain	51.0 db	1.0	0.5	
4	Transmitting antenna pointing loss	0.0 db	0.0	0.0	
5	Space loss E + 6 months 2115 Mc R = 390 x 10 km	-270.77 db	0.0	0.0	
6	Polarization loss - (DSIF) = 1.5 db, ∈(S/C) = 5 db	-0.73 db	-	-	Note 2
7	Receiving antenna gain	27.5 db	0.25	0.5	Note 1
8	Receiving antenna pointing loss	-1.0 db	0.2	0.5	
9	Receiving circuit loss	-1.26 db	0.2	0.2	
10	Net circuit loss	-195.66 db	1.75	1.8	
11	Total received power	-125.66 dbm	2.25	1.8	
12	Receiver noise spectral density (N/B) T System = 2610°	-164.44 dbm/cps	1.12	2.18	
13	Carrier modulation loss	-2.57 db	0.27	0.28	
14	Received carrier power	-128.23 dbm	2.52	2.08	
15	Carrier APC noise bandwidth ($2B_{LO}$ = 20 cps)	13.01 db·cps	0.46	0.41	
	Carrier Performance Tracking (one-way)				
16	Threshold SNR in $2B_{LO}$	0.0 db	0.0	0.0	
17	Threshold carrier power	-151.43 dbm	1.58	2.59	
18	Performance margin	23.2 db	4.1	4.67	
	Carrier Performance Tracking (two-way)				
19	Threshold SNR in $2B_{LO}$	3.8 db	0.0	0.0	
20	Threshold carrier power	-147.63 dbm	1.58	2.59	
21	Performance margin	19.4 db	4.1	4.67	
	Carrier Performance				
22	Threshold SNR in $2B_{LO}$	8.0 db	1.0	1.0	
23	Threshold carrier power	-143.43 dbm	2.58	3.59	
24	Performance Margin	15.2 db	5.1	5.67	
	Data Channel				
25	Modulation loss	-7.77 db	0.51	0.54	
26	Received data subcarrier power	-133.43 dbm	2.76	2.34	
27	Bit rate (1/T)	0.0 db	0.0	0.0	
28	Required ST/N/B ($Pe^b = 1 \times 10^{-6}$)	15.8 db	1.0	1.0	
29	Threshold subcarrier power	-148.64 dbm	2.12	3.18	
30	Performance margin	15.21 db	4.88	5.52	
	Sync Channel				
31	Modulation loss	-6.82 db	0.53	0.56	
32	Received sync subcarrier power	-132.48 dbm	2.78	2.36	
33	Sync APC noise bandwidth ($2B_{LO}$ = 2 cps)	3.01 db·cps	0.8	0.8	
34	Threshold SNR in $2B_{LO}$	13.7 db	1.0	1.0	
35	Threshold subcarrier power	-147.73 dbm	2.92	3.98	
36	Performance margin	15.25 db	5.7	6.34	

Notes: 1. Sources for spacecraft parameters are specifications given in section 3.0 of this volume.
2. Only maximum polarization loss was computed.
3. Sources for the DSIF parameters are indicated in tables 3.6 and 3.7.

Table 48. Telecommunications Design Control

Channel: Earth-to-Spacecraft
Mode: Acquisition Aid Station Mode; Spacecraft Low-Gain Antenna

No.	Parameter	Value	Tolerance +	Tolerance -	Source
1	Total transmitter power (multiplexed)	65.44 dbm	0.5	0.0	Note 3
2	Transmitting circuit loss	-0.5 db	0.2	0.2	
3	Transmitting antenna gain	20.0 db	2.0	2.0	
4	Transmitting antenna pointing loss	0.0 db	0.0	0.0	
5	Space loss (Injection +8 hours) 2115 Mc $R = 1.33 \times 10^5$ km	-201.45 db	0.0	0.0	
6	Polarization loss ϵ(DSIF) = 1.5 db ϵ(S/C) = 6.0 db	-0.74 db	-	-	Note 2
7	Receiving antenna gain	5.0 db	0.5	0.5	Note 1
8	Receiving antenna pointing loss Cone Angle = 78°	-6.0 db	-	-	
9	Receiving circuit loss	-1.26 db	0.2	0.2	
10	Net circuit loss	-184.95 db	2.9	2.9	
11	Total received power	-119.51 dbm	3.4	2.9	
12	Receiver noise spectral density (N/B) T System = 2610°K	-164.44 dbm/cps	1.12	2.18	
13	Carrier modulation loss	-2.57 db	0.27	0.28	
14	Received carrier power	-122.08 dbm	3.67	3.18	
15	Carrier APC noise bandwidth ($2B_{LO}$ = 20 cps)	13.01 db cps	0.46	0.41	
	Carrier Performance Tracking (one-way)				
16	Threshold SNR in $2B_{LO}$	0.0 db	0.0	0.0	
17	Threshold carrier power	-151.43 dbm	1.58	2.59	
18	Performance margin	29.35 db	5.25	5.77	
	Carrier Performance Tracking (two-way)				
19	Threshold SNR in $2B_{LO}$	3.8 db	0.0	0.0	
20	Threshold carrier power	-147.63 dbm	1.58	2.59	
21	Performance margin	25.55 db	5.25	5.77	
	Carrier Performance				
22	Threshold SNR in $2B_{LO}$	8.0 db	1.0	1.0	
23	Threshold carrier power	-143.43 dbm	2.58	3.59	
24	Performance Margin	21.35 db	6.25	6.77	
	Data Channel				
25	Modulation loss	-7.77 db	0.51	0.54	
26	Received data subcarrier power	-127.28 dbm	3.91	3.44	
27	Bit rate (1/T)	0.0 db	0.0	0.0	
28	Required ST/N/B ($P_e^b = 1 \times 10^{-6}$)	15.8 db	1.0	1.0	
29	Threshold subcarrier power	-148.64 dbm	2.12	3.18	
30	Performance margin	21.36 db	6.03	6.62	
	Sync Channel				
31	Modulation loss	-6.82 db	0.53	0.56	
32	Received sync subcarrier power	-126.33 dbm	3.93	3.46	
33	Sync APC noise bandwidth ($2B_{LO}$ = 2 cps)	3.01 db · cps	0.8	0.8	
34	Threshold SNR in $2B_{LO}$	13.7 db	1.0	1.0	
35	Threshold subcarrier power	-147.73 dbm	2.92	3.98	
36	Performance margin	21.40 db	6.85	7.44	

Notes:
1. Sources for spacecraft parameters are specifications given in section 3.0 of this volume.
2. Only maximum polarization loss was computed.
3. Sources for the DSIF parameters are indicated in tables 3.6 and 3.7.

Table 49. Telecommunications Design Control

Channel: Turnaround Ranging with 234-3/8 bits/sec Telemetry

Mode: Spacecraft High-Gain Antenna, 210-Foot DSIF Antenna 100 kw Diplexed

No.	Parameter	Value	Tolerance +	Tolerance -	Source
Uplink			+	-	
1	Total transmitter power	80.0 dbm	-	-	Note 3
2	Transmitting circuit loss	-0.4 db	0.1	0.1	
3	Transmitting antenna gain	60.6 db	1.0	0.5	
4	Transmitting antenna pointing loss	-0.5 db	-	-	
5	Space loss				
	2115 Mc R = 390 x 10^6 km	-270.77 db	0.0	0.0	
6	Polarization loss				
	ϵ(DSIF) = 0.5 ϵ(S/C) = 5.0	-0.54 db	-	-	Note 2
7	Receiving antenna gain	33.5 db	0.25	0.5	Note 1
8	Receiving antenna pointing loss	-0.4 db	0.4	0.3	
9	Receiving circuit loss	-1.36 db	0.2	0.2	
10	Net circuit loss	-179.87 db	1.95	1.6	
11	Total received power	-99.87 dbm	1.95	1.6	
12	Receiver noise spectral density (N/B)				
	T system = 2610°K	-164.44 dbm/cps	1.12	2.18	
13	Receiver noise bandwidth (3.3 Mc)	65.18 db	0.45	0.42	
14	Total received SNR (in 3.3 Mc) = $1/\beta$	-0.61 db	3.52	4.2	
15	Limiter signal suppression factor				
	$\alpha_s^2 = 1/(1+\frac{4}{\pi}\beta)$	0.405	0.20	0.203	
16	Limiter noise suppression factor				
	$\alpha_n^2 = \frac{\beta}{\beta+2}$	0.37	0.16	0.23	
Downlink					
17	Total transmitter power	47.0 dbm	1.0	0.0	
18	Transmitting circuit loss	-2.05 db	0.8	0.8	
19	Transmitting antenna gain	34.0 db	0.25	0.5	
20	Transmitting antenna pointing loss	-0.4 db	0.4	0.3	
21	Space loss				
	2295 Mc R = 390 x 10^6 km	-271.5 db	0.0	0.0	
22	Polarization loss				
	ϵ(DSIF) = 0.5 db ϵ(S/C) = 2 db	-0.09 db	-	-	
23	Receiving antenna gain	61.7 db	1.0	0.5	
24	Receiving antenna pointing loss	-0.5 db	-	-	
25	Receiving circuit loss	-0.2 db	0.1	0.1	
26	Net circuit loss	-179.04 db	2.55	2.2	
27	Total received power	-132.04 dbm	3.55	2.2	
28	Receiver noise spectral density (N/B)				
	T system = 30°K ±5°K	-183.8 dbm/cps	0.8	0.7	
29	Uplink contribution to noise spectral density	0.04 db/cps	0.02	0.02	
30	Total effective noise spectral density	-183.76 dbm/cps	0.82	0.72	
31	Carrier modulation loss				
	$\theta_R' = 0.70$; $\sigma_n^2 = 0.246$, $\theta_D = 0.74$, $\theta_S = 0.25$	-6.33 db	0.93	1.25	
32	Received carrier power	-138.37 dbm	4.48	3.45	
33	Carrier APC noise bandwidth ($2B_{LO}$ = 12 cps)	10.8 db	0.5	0.0	

Notes: 1. Sources for spacecraft parameters are specifications given in section 3.0 of this volume

2. Only maximum polarization loss was computed

3. Sources for the DSIF parameters are indicated in tables 3.6 and 3.7.

Table 49. Telecommunications Design Control (Cont)

Channel: Turnaround Ranging with 234-3/8 bps Telemetry
Mode: Spacecraft High-Gain Antenna, 210-Foot DSIF Antenna 100 kw Diplexed

No.	Parameter	Value	Tolerance +	Tolerance -	Source
Carrier Performance					
34	Threshold SNR in 2 B_{LO}	6.0 db	0.5	1.0	
35	Threshold carrier power	-166.96 dbm	1.82	1.72	
36	Performance margin	28.59 db	6.30	5.17	
Ranging Performance					
37	Modulation loss $\theta_R' = 0.70$; $\sigma_n^2 = 0.246$ $\theta_D = 0.74$, $\theta_S = 0.25$	-7.79 db	2.17	3.67	
38	Received ranging power	-139.83 dbm	5.72	5.87	
39	Ranging APC noise bandwidth ($2B_{LO} = 0.8$ cps)	-0.97 db cps	0.5	0.0	
40	Threshold SNR in $2B_{LO}$	22.0 db	1.0	1.0	
41	Threshold ranging power	-162.73 dbm	2.32	1.72	
42	Performance margin	22.90 db	8.04	7.59	
Data Channel					
43	Modulation loss $\theta_D = 0.74$, $\theta_S = 0.25$ $\theta_R' = 0.70$, $\sigma_n^2 = 0.246$	-7.09 db	0.99	1.29	
44	Received data subcarrier power	-139.13 dbm	4.54	3.49	
45	Bit rate (1/T) (234-3/8 bits/sec)	23.7 db	0.0	0.0	
46	Required ST/N/B ($P_e^b = 5 \times 10^{-3}$)	7.7 db	1.0	1.0	
47	Threshold subcarrier power	-152.36 dbm	1.82	1.72	
48	Performance margin	13.23 db	6.36	5.21	
Sync Channel					
49	Modulation loss $\theta_D = 0.74$, $\theta_S = 0.25$ $\theta_R' = 0.70$, $\sigma_n^2 = 0.246$	-18.19 db	1.29	1.67	
50	Received SYNC subcarrier power	-150.23 dbm	4.84	3.87	
51	Sync APC noise bandwidth (2 B_{LO} + 0.5 cps)	-3.01 db cps	0.4	0.4	
52	Threshold SNR in 2 B_{LO}	18.4 db	1.0	1.0	
53	Threshold subcarrier power	-168.37 dbm	2.22	2.12	
54	Performance margin	18.14	7.06	5.99	

Tables 50 and 51 show the performance of turnaround ranging, with multiplexed telemetry using an 85-foot DSIF station. Under strong signal conditions at the spacecraft, the range code modulation index was chosen to be 1.1 radians in order that the maximum carrier modulation loss was less than 10 db when 10 per cent of the total power was allocated to the telemetry. The composite telemetry signal received from telemetry subsystem is attenuated by 4.2 db in the phase modulators during the multiplexed ranging/telemetry mode. Table 50 shows that multiplexed ranging and telemetry via the spacecraft low-gain antenna, using a diplexed 10-kw, 85-foot DSIF station, is possible to at least 10 days (2.6×10^6 km) after launch. Table 51

Table 50. Telecommunications Design Control

Channel: Turnaround Ranging with Telemetry

Mode: 85 Foot DSIF Antenna, 10 kw Diplexed, Spacecraft Low-Gain Antenna

No.	Parameter	Value	Tolerance		Source
Uplink			+	-	
1	Total transmitter power	70.0 dbm	0.5	0.0	Note 3
2	Transmitting circuit loss	-0.4 db	0.1	0.1	
3	Transmitting antenna gain	51.0 db	1.0	0.5	
4	Transmitting antenna pointing loss	0.0 db	0.0	0.0	
5	Space loss (launch +10 days)				
	2115 Mc $R = 2.0 \times 10^6$ km	-225.00 db	0.0	0.0	
6	Polarization loss				
	ϵ(DISF) = 1.5 db ϵ(S/C) = 6 db	-0.74 db	-	-	Note 2
7	Receiving antenna gain	5.0 db	0.5	0.5	Note 1
8	Receiving antenna pointing loss	-4.6 db	-	-	
9	Receiving circuit loss	-1.26 db	0.2	0.2	
10	Net circuit loss	-173.52 db	1.8	1.3	
11	Total received power	-103.52 dbm	2.3	1.3	
12	Receiver noise spectral density (N/B)				
	T system = 2610°K	-164.44 dbm/cps	1.12	2.18	
13	Receiver noise bandwidth	65.18 db cps	0.45	0.42	
14	Total received SNR (in 3.3 Mc) = $1/\beta$	-6.74 db	3.87	3.9	
15	Limiter signal suppression factor				
	$\alpha_s^2 = 1/(1 + \frac{4}{\pi}\beta)$	0.143	0.15	0.08	
16	Limiter noise suppression factor				
	$\alpha_n^2 = \frac{\beta}{\beta + 2}$	0.7	0.21	0.15	
Downlink					
17	Total transmitter power	47.0 dbm	1.0	0.0	
18	Transmitting circuit loss	-2.65 db	0.8	0.8	
19	Transmitting antenna gain	4.2 db	0.5	0.5	
20	Transmitting antenna pointing loss	-4.6 db	-	-	
21	Space loss				
	2295 Mc $R = 2.0 \times 10^6$ km	-225.70 db	0.0	0.0	
22	Polarization loss				
	ϵ(DSIF) = 1.5 db ϵ(S/C) = 4 db	-0.34 db	-	-	
23	Receiving antenna gain	53.0 db	1.0	0.5	
24	Receiving antenna pointing loss	0.0 db	0.0	0.0	
25	Receiving circuit loss	-0.2 db	0.1	0.1	
26	Net circuit loss	-176.29 db	2.4	1.9	
27	Total received power	-129.99	3.4	1.9	
28	Receiver noise spectral density (N/B)				
	T system = $55 \pm 10^\circ$K	-181.2 dbm/cps	0.9	0.7	
29	Uplink contribution to noise spectral density	0.07 db	0.02	0.01	
30	Total effective noise spectral density	-181.13 dbm/cps	0.92	0.71	
31	Carrier modulation loss				
	$\theta_R' = 0.416$; $\sigma_n^2 = 0.473$ $\theta_D = 0.74$, $\theta_S = 0.25$	-5.74 db	0.34	0.64	
32	Received carrier power	-135.73 dbm	3.74	2.54	
33	Carrier APC noise bandwidth ($2B_{LO}$ = 12 cps)	10.8 db cps	0.5	0.0	

Notes: 1. Sources for spacecraft parameters are specifications given in section 3.0 of this volume
2. Only maximum polarization loss was computed
3. Sources for the DSIF parameters are indicated in tables 3.6 and 3.7.

Table 50. Telecommunications Design Control (Cont)

Channel: Turnaround Ranging with Telemetry
Mode: 85 Foot DSIF Antenna, 10 kw Diplexed, Spacecraft Low-Gain Antenna

No.	Parameter	Value	Tolerance +	Tolerance -	Source
Carrier Performance					
34	Threshold SNR in 2 B_{LO}	6.0 db	0.5	1.0	
35	Threshold carrier power	-164.33 dbm	1.92	1.71	
36	Performance margin	28.63 db	5.66	4.25	
Ranging Performance					
37	Modulation loss $\theta_D = 0.74$, $\theta_S = 0.25$ $\theta_R' = 0.416$; $\sigma_n^2 = 0.473$	-12.82 db	4.7	4.18	
38	Received ranging power	-142.81 dbm	8.10	6.08	
39	Ranging APC noise bandwidth (2 B_{LO} = 0.8 cps)	-0.97 db cps	0.5	0.0	
40	Threshold SNR in 2 B_{LO}	22.0 db	1.0	1.0	
41	Performance margin	-160.10 dbm	2.42	1.71	
42	Performance margin	17.29 db	10.52	7.79	
Data Channel					
43	Modulation loss	-6.50 db	0.40	0.68	
44	Received data subcarrier power	-136.49 dbm	3.83	2.58	
45	Bit rate (1/T) (234-3/8 bps)	23.7 db	0.0	0.0	
46	Required ST/N/B ($P_e^b = 5 \times 10^{-3}$)	7.7 db	1.0	1.0	
47	Threshold subcarrier power	-149.84 dbm	1.92	1.71	
48	Performance margin	13.35 db	5.75	4.29	
Sync Channel					
49	Modulation loss	-17.60 db	0.70	1.06	
50	Received sync subcarrier power	-147.59 dbm	4.10	2.96	
51	Sync APC noise bandwidth (2 B_{LO} = 0.5 cps)	-3.01 db cps	0.4	0.4	
52	Threshold SNR in 2 B_{LO}	18.4 db	1.0	1.0	
53	Threshold subcarrier power	-165.74 dbm	2.32	2.41	
54	Performance margin	18.15 db	6.42	5.07	

Table 51. Telecommunications Design Control

Channel: Turnaround Ranging with Telemetry
Mode: 85 Foot DSIF Antenna, 10 kw Diplexed, Spacecraft High-Gain Antenna

No.	Parameter	Value	Tolerance +	Tolerance -	Source
Uplink					
1	Total transmitter power	70.0 dbm	0.5	0.0	Note 3
2	Transmitting circuit loss	-0.4 db	0.1	0.1	
3	Transmitting antenna gain	51.0 db	1.0	0.5	
4	Transmitting antenna pointing loss	0.0 db	0.0	0.0	
5	Space loss L + 150 days				
	2115 Mc R - 84.0 x 10^6 km	-257.36 db	0.0	0.0	
6	Polarization loss				
	ϵ(DSIF) = 1.5 db ϵ(S/C) = 5 db	-0.73 db	-	-	Note 2
7	Receiving antenna gain	33.5 db	0.25	0.5	Note 1
8	Receiving antenna pointing loss	-0.4 db	0.4	0.3	
9	Receiving circuit loss	-1.36 db	0.2	0.2	
10	Net circuit loss	-175.75 db	1.95	1.6	
11	Total received power	-105.75 dbm	2.45	1.6	
12	Receiver noise spectral density (N/B)				
	T system = 2610°K	-164.44 dbm/cps	1.12	2.18	
13	Receiver noise bandwidth	65.18 db cps	0.45	0.42	
14	Total received SNR (in 3.3 Mc) = $1/\beta$	-6.49 db	4.02	4.20	
15	Limiter signal suppression factor				
	$\alpha_s^2 = 1/(1+\frac{4}{\pi}\beta)$	0.150	0.158	0.087	
16	Limiter noise suppression factor				
	$\alpha_n^2 = \frac{\beta}{\beta+2}$	0.69	0.22	0.165	
Downlink					
17	Total transmitter power	47.0 dbm	1.0	0.0	Note 1
18	Transmitting circuit loss	-2.05 db	0.8	0.8	
19	Transmitting antenna gain	34.0 db	0.25	0.5	
20	Transmitting antenna pointing loss	-0.4 db	0.4	0.3	
21	Space loss				
	2115 Mc R = 84.0 x 10^6 km	-258.16 db	0.0	0.0	
22	Polarization loss				
	ϵ(DSIF) = 1.5 db ϵ(S/C) = 2.0 db	-0.13 db	-	-	Note 2
23	Receiving antenna gain	53.0 db	1.0	0.5	Note 3
24	Receiving antenna pointing loss	0.0 db	0.0	0.0	
25	Receiving circuit loss	-0.2 db	0.1	0.1	
26	Net circuit loss	-173.94 db	2.55	2.2	
27	Total received power	-126.94 dbm	3.55	2.2	
28	Receiver noise spectral density (N/B)				
	T system = 55°K ±10°K	-181.2 dbm/cps	0.9	0.7	
29	Uplink contribution to noise spectral density	0.14 db	0.04	0.01	
30	Total effective noise spectral density	-181.06	0.94	0.71	
31	Carrier modulation loss				
	$\theta_D = 0.74$; $\sigma_n^2 = 0.464$ $\theta_R' = 0.427$ $\theta_S = 0.25$	-5.76 db	0.32	0.59	
32	Received carrier power	-132.70 db	3.87	2.79	
33	Carrier APC noise bandwidth ($2B_{LO}$ = 12 cps)	10.8 db	0.5	0.0	

Notes:
1. Sources for spacecraft parameters are specifications given in section 3.0 of this volume
2. Only maximum polarization loss was computed
3. Sources for the DSIF parameters are indicated in tables 3.6 and 3.7.

Table 51. Telecommunications Design Control (Cont)

Channel: Turnaround Ranging with Telemetry
Mode: 84 Foot DSIF Antenna, 10 kw Diplexed, Spacecraft High-Gain Antenna

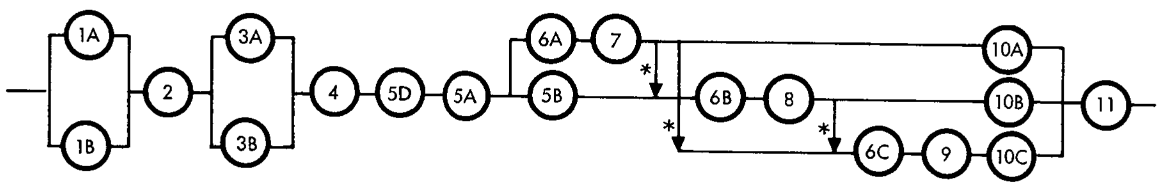
No.	Parameter	Value	Tolerance +	Tolerance -	Source
Carrier Performance					
34	Threshold SNR in 2 B_{LO}	6.0 db	0.5	1.0	
35	Threshold carrier power	-164.26 dbm	1.94	1.71	
36	Performance margin	31.56 db	5.81	4.50	
Ranging Performance					
37	Modulation Loss $\theta_R' = 0.427$, $\theta_S = 0.25$ $\theta_D = 0.74$; $\sigma_n^2 = .464$	-12.58 db	3.8	4.67	
38	Received ranging power	-139.52 dbm	7.35	6.87	
39	Ranging APC noise bandwidth (2 B_{LO} = 0.8 cps)	-0.97 db	0.5	0.0	
40	Threshold SNR in 2 B_{LO}	22.0 db	1.0	1.0	
41	Threshold ranging power	-160.03 dbm/cps	2.44	1.71	
42	Performance margin	20.51 db	9.79	8.58	
Data Channel					
43	Modulation loss	-6.52 db	0.38	0.63	
44	Received data subcarrier power	-133.46 dbm	3.93	2.83	
45	Bit rate (1/T) (234-3/8 bits/sec)	23.7 db	0.0	0.0	
46	Required ST/N/B (P_e^b = 5 x 10^{-3})	7.7 db	1.0	1.0	
47	Threshold subcarrier power	-149.66 dbm	1.94	1.71	
48	Performance margin	16.20 db	5.87	4.54	
Sync Channel					
49	Modulation loss	-17.62 db	0.68	1.01	
50	Received sync subcarrier power	-144.56 dbm	4.23	3.21	
51	Sync APC noise bandwidth (2 B_{LO} = 0.5 cps)	-3.01 db cps	0.4	0.4	
52	Threshold SNR in 2 B_{LO}	18.4 db	1.0	1.0	
53	Threshold subcarrier power	-165.67 dbm	2.34	2.11	
54	Performance margin	21.11 db	6.57	5.32	

indicates that by substituting the high-gain spacecraft for the low-gain antenna, the 10-kw, 85-foot stations provide satisfactory performance to 84 x 10^6 km.

3.5.5 Reliability

The S-band radio subsystem is essentially the same as that analyzed in Task A, except that the command detectors are now part of the command subsystem. Although the antenna drive electronics and mechanisms are part of the guidance and control subsystem, the effects of the failure rates of these elements are included for the sake of completeness. Although the Task A study determined the reliability of the transmission and reception functions separately, this analysis considers the entire radio subsystem as one function. Since the high-power transmitter on the low-gain antenna and the low-power transmitter on any antenna do not provide satisfactory system performance (science data rates) at Mars ranges, these paths are not included in the reliability analysis. Under these assumptions, the estimated reliability for the combined transmitting and receiving functions is 0.95 from (and including) launch to the end of one month in Mars orbit.

Figure 81 shows the S-band radio subsystem reliability block diagram. Table 52 summarizes the unit failure rates and deployment reliabilities. The reliability model is conservative to the extent that it does not decrease the failure rate of the nonoperating TWT and it assumes that every part is required for successful operation of a unit.



$$R = \left[1 - (1-R_1)^2\right] R_2 \left[1 - (1-R_3)^2\right] R_4 R_5^2 R_6 R_{10} \Big\{ R_7 + (1-R_6 R_7) R_5 R_8 \left[1 + R_6 R_9 (1 - R_{10})\right] + R_6 R_7 (1-R_{10}) \left[R_8 + (1-R_6 R_8 R_{10}) R_9\right] \Big\} R_{11}$$

R = 0.9468 FOR TCTAL MISSION INCLUDING ONE MONTH IN MARS ORBIT (t = 5350 HOURS)

$R_i = e^{-\lambda_i t} R_{Di}$ WHERE $i = 1, 2, \ldots 11$

* THE ARROWS INDICATE SUCCESS PATHS

Figure 81. S-Band Radio Subsystem Reliability Block Diagram

Table 52. Radio Subsystem Failure Rate Summary

Item	Notes	Item Description	Failure Rate (Failures/10^9 hrs)	Deployment Reliability R_{Di}
1	1, 2	Modulator-exciter and power monitor	5,036	1
2		Four part hybrid ring	250	1
3	1, 2	50-watt TWT, power supply, and power monitor	15,313	1
4		Transmitter selector	2,342	1
5	1, 2 3, 4	Circulator switch	250	1
6	1, 2 3	Diplexer	250	1
7		High-gain antenna	42,410	0.9998
8		Medium-gain antenna	21,220	0.9998
9		Low-gain antenna	60	0.9998
10	1, 2 3	S-Band receiver	17,228	1
11		Receiver selector	1,219	1

Notes:
1. Associated with high-gain antenna
2. Associated with medium-gain antenna
3. Associated with low-gain antenna
4. Circulator switch No. 1

3.6 Design Alternatives

This section discusses the rationale which led to changes in the radio subsystem since the completion of the Phase IA, Task A study.

3.6.1 Bit Rate Selection

Range capability at the bit rates selected under the Task B study is shown in Table 53 for four transmission modes.

Table 53. Range Capability

	Bit Rates					
	7.3	234	1875	3750	7500	15000
	$\times 10^6$ km	$\times 10^6$ km	$\times 10^6$ km	$\times 10^6$ km	$\times 10^6$ km	$\times 10^6$ km
1) 50-watt, high gain antenna	390[1]	390[1]	390[2]	390[2]	390[2]	270[2]
2) 50-watt, medium-gain antenna	390[1]	310[1]	390[2]	270[2]	190[2]	135[2]
3) 50-watt, low-gain antenna	180[2]	45[2]				
4) 1-watt, high-gain antenna	390[2]	260[2]	97[2]	68[2]	48[2]	34[2]
	Emergency Mode	Boost-Cruise	Orbital Operations			

Note: Encounter Range 80 to 180 x 10^6 km

+ 2 months 150 to 265 x 10^6 km

+ 6 months 310 to 390 x 10^6 km

[1] 85-foot DSIF antenna

[2] 210-foot DSIF antenna

The increase in the orbital operation bit rates from those in the Task A study (4096, 2048, 1024) is obtained by an increase in transmitted power and antenna gain (discussed in Paragraphs 3.6.2 through 3.6.4) made possible by the Task B program redirection and revised spacecraft parameters.

The 15,000 and 3750 bit rates were selected to obtain coverage to encounter plus 2 months for transmission modes 1 and 2. The 7500 and 1875 bit rates provide coverage to worst-case encounter plus 6 months for these modes. The 7500 bit rate is also required after encounter if both spacecraft are transmitting in order to limit the maximum data rate total to 15,000 bits/sec, as specified by JPL.

The 234.4 bit rate was selected to provide adequate cruise and maneuver telemetry rate, and at the same time provide sufficient signal margin for early post launch communications, particularly for immediate post injection tracking utilizing the DSN acquisition aid receiving system.

The 7.4 bit rate provides diagnostic telemetry in the event of abnormal spacecraft operation. In transmission mode 3, it provides coverage to worst-case encounter range; while in mode 4 it provides coverage to encounter plus 6 months.

3.6.2 Fifty-watt TWT Amplifier

JPL Task B design requirements specify the use of a 50-watt power amplifier. In view of the additional development time made available by elimination of the 1969 Voyager mission, and the increased solar array power of the Task B spacecraft configuration, a 50-watt amplifier is quite feasible, particularly in view of development activities in this area now being initiated by JPL.

The TWT and ESK klystron are competitive in performance at the 50-watt power level. The ESK klystron has the advantage of a much less critical power supply interface. In addition, it can be made nonmagnetic. The TWT, on the other hand, has a much greater usage and reliability history (highly favorable), and a conservative choice favors selection of the TWT at this time.

3.6.3 High-Gain Antenna

The high-gain antenna was previously limited in size by fairing and stowage constraints to be a 5.5 by 6.5-foot elliptical paraboloid providing 30-db gain. The Task B spacecraft configuration allows room for a 9.5-foot diameter paraboloid providing 34-db gain at the transmitting frequency. The use of an 18-db edge illumination taper broadens the main lobe slightly, thus maintaining pointing loss to about the same value (≈ 1 db) as obtained with the Task A high-gain antenna.

3.6.4 Medium-Gain Antenna

The Task A concept of utilizing a single-gimballed, medium-gain antenna, so as to provide alternate mechanization redundancy for the high-gain antenna, has been retained. However, with the new spacecraft configuration, it becomes possible to enlarge considerably the antenna dimensions. The selected 84 by 36-inch elliptical aperture has a 10-degree beamwidth in the fixed plane providing the same pointing loss as in the Task A design. The single gimbal provides updating in the narrow (4-degree) beam dimension.

3.6.5 Low-Gain Antenna

The basic low-gain antenna is identical with that proposed for the Task A design. However, in order to provide coverage to a cone angle maximum of about 120 degrees for any clock angle, the antenna is mounted on a 14-foot deployable boom. This is of importance during the first 2 to 3 weeks of flight when the cone angle is greater than 90 degrees to insure uplink and downlink communications capability for any roll (clock) angle. During boost, the boom is stowed and the antenna radiates through an aperture coupler on the shroud, thus eliminating the need for the secondary low-gain antenna utilized in the Task A design.

4. CAPSULE RADIO LINK

4.1 General Description

This section describes those portions of the capsule radio link which form an integrated part of the spacecraft equipment and are retained on the spacecraft after capsule separation. The equipment which comprises the complete capsule radio link (with the possible exception of the receiving antenna) will be supplied GFE to the spacecraft contractor for integration into the spacecraft. The purpose of the ensuing paragraphs is to define and establish spacecraft requirements necessary to support the capsule radio link receiver(s), demodulator(s), tape recorder, and antenna.

The subsystem elements of the capsule radio link which mate directly with the spacecraft are:

- UHF receivers (2)
- Demodulators (2)
- Preamphifier
- Tape recorder

Figure 82 is a functional block diagram of these elements showing interfaces with the spacecraft.

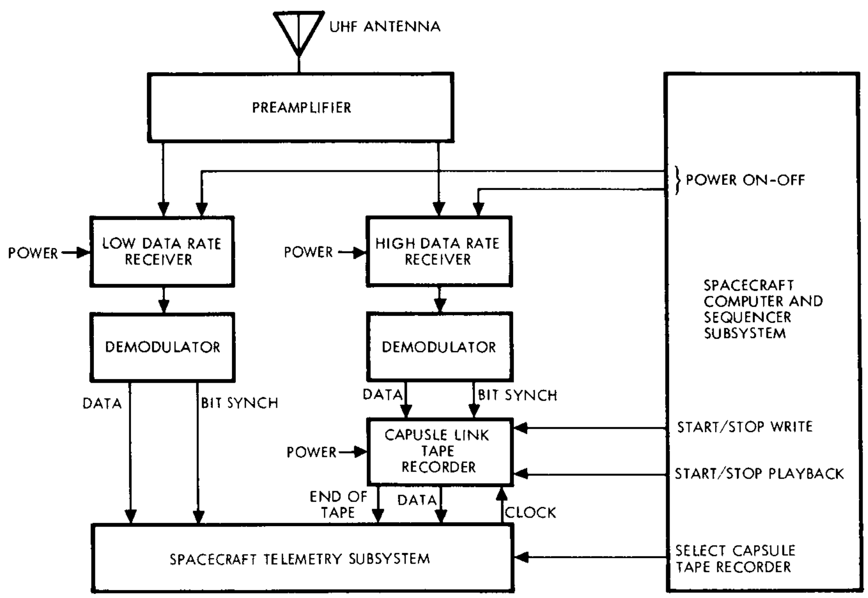


Figure 82. Functional Block Diagram of Spacecraft Equipment

4.2 Requirements and Constraints

4.2.1 Mission Constraints

The capsule relay link will operate with two separate data channels with separate capsule transmitters. A low data rate of 100 bits per second will comprise one data channel and operate from spacecraft-capsule separation to Mars impact. A high data rate channel of 50,000 to 200,000 bits per second will comprise the other data channel and operate from atmospheric entry (approximately 800,000 feet altitude) to Mars impace (5 to 15 minutes as determined by the entry flight time).

The orbital descent time (from capsule motor ignition to atmospheric entry) must be greater than 50 minutes and less than 12 hours.

The design of the capsule radio link should not be limited to the nominal orbit of 2000 to 20,000 kilometers defined in Volume 1. Sufficient capability will be required to accommodate other than the nominal orbit based on prelaunch selection and adjustment and the accuracy of achieving a specified orbit.

The maximum allowable communications range is determined by G(db) = 10 log (antenna gain factor) -20 log (range in thousands of km), where G must be greater than -5 db.

With a nominal antenna gain of 10 db, the maximum allowable range is approximately 5600 km.

A maximum range of approximately 3000 km is reached during the capsule terminal descent and entry phase (for a nominal orbit of the flight spacecraft) for various types of capsule descent orbits. The first class of descent orbits requires separation at a true anomaly of approximately 240 to 270 degrees of the spacecraft reference orbit and yields descent times not exceeding 2 hours. The second class of descent orbits, which meets additional constraints imposed on the capsule descent and landing phase, (see also Appendix B of Volume 1) requires separation at a true anomaly of approximately 120 to 140 degrees of the spacecraft reference orbit, and yields descent times up to, but not exceeding, 12 hours. During the early portion of the descent the separation range between flight spacecraft and capsule increases to approximately 5000 km, but decreases to less than 3000 km at the time of capsule atmospheric entry.

For the 1971 mission the lifetime of the lander (from capsule separation to capsule impact) will be greater than 50 minutes and less than 12 hours. The capsule is assumed to be destroyed on impact; all data transmission and reception over the capsule radio link will take place during the descent time period.

All sequential events for capsule radio link operation will be automatically programmed through the spacecraft C and S prior to separation. Those portions of the capsule link retained on the spacecraft subsequent to capsule separation will derive controls for programmed sequences from the spacecraft C and S.

4.2.2 Subsystem Requirements

The following requirements are placed on the capsule radio link:

- Two frequencies in the 400 ±1 Mc band will be required, one for the low data rate link and one for the high data rate link.
- One frequency will carry telemetry at 100 tps continuously from separation to end of mission.
- The other frequency will carry telemetry at 50 to 200 kilobits/sec during the terminal descent to end of mission.
- The low data rate information will be relayed to earth in real time by the spacecraft telemetry. The high data rate information is recorded and played back later to earth.
- During the cruise phase engineering measurements of capsule functions are transmitted to the spacecraft through the capsule umbilical. This data will be transmitted over the capsule radio link low data rate transmitter during the orbital descent phases.

4.3 Functional Interfaces

4.3.1 Electrical Input Signals

a. Received Capsule Signal

Both the low and high data rate carriers will be received by a single antenna. The signals will have a power level between -4 and -120 dbm.

b. C and S Subsystem

A power on-off command and a separation signal will be provided to control the operation of the capsule radio equipment and to switch the operation from hard line to radio link. Start-stop commands will be supplied by the C and S to the capsule link tape recorder for both write and playback modes.

c. Command Subsystem

Back-up commands for the on-off and separation function will be provided by the command subsystem.

d. Primary Power

The primary power supplied to the capsule radio link receivers on the spacecraft is 4.1 kc ±1%, 50 volt ±2% peak to peak square wave at 5 watts. The primary power supplied to the capsule link tape recorder is 10 watts maximum.

e. Telemetry Subsystem

Capsule data mode select commands will be supplied to the spacecraft telemetry by the spacecraft C and S. Clock signals will be provided by the spacecraft telemetry subsystem to the tape recorder.

4.3.2 Electrical Output Signals

a. Telemetry Subsystem

The capsule data bit stream and bit sync will be provided for both the low and high data rate links. Selected telemetry monitor points will be provided for signal strength, regulated voltages, and in-lock indication.

4.3.3 Mechanical Interfaces

The UHF antenna occupies a volume of 27 x 27 x 10 inches and weighs 14 pounds. It is mounted on the aft side of the solar array at a clock angle of 230°. The beam is pointed in the direction of the capsule descent trajectory at a nominal cone angle of 140° and a nominal clock angle of 255°. The antenna is designed to withstand a temperature range of -250°F to $+300^{\circ}$F which is adequate for any spacecraft-sun angle.

4.4 Design Description

The capsule link antenna shown in Figure 83 is a quad-spiral array which provides a right-hand circularly polarized single lobe radiation pattern, symmetrical about the axis of the array. The array has a gain of 10 dbi and a half-power beamwidth of 50 degrees. Each element in the array is a cavity-backed, two-arm Archimedian spiral fed by a balm transformer incorporated into the feed transmission line. Three integrated into the feed transmission line. Three integrated hybrid rings divide the power equally and maintain equal phase. Output connectors of

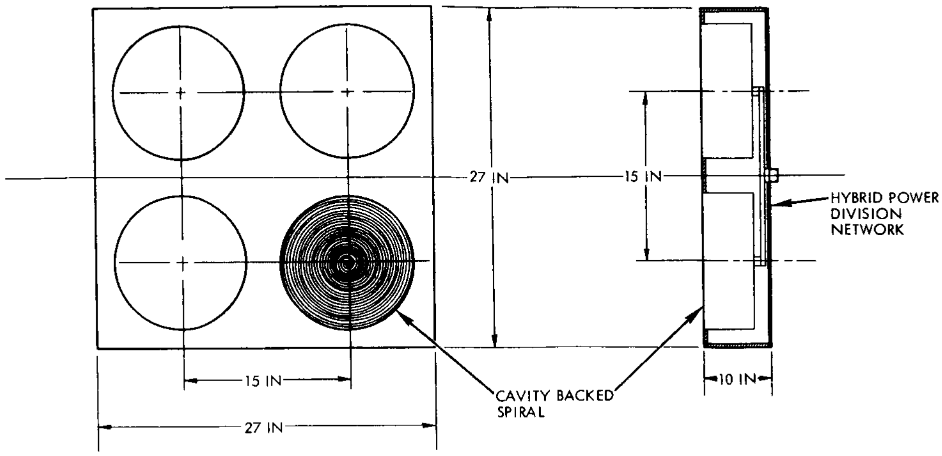


Figure 83. UHF Capsule-to-Vehicle Antenna

the power divider are connected directly to the balun and transmission line of the element. The element design is selected for midband operation at the link frequency for maximum efficiency. The optimum spacing between elements of the array is near one-half-wavelength center-to-center. The overall size of the array is 27 x 27 x 10 inches. The form factor of the antenna allows it to be placed near the edge of the spacecraft, where a minimum of pattern interference from surrounding structure is experienced.

The UHF preamplifier, receiver, and demodulator, packaged as a single unit, has the following characteristics:

size:	260 cubic inches
weight:	7 pounds
power:	2.5 watts (low data rate only)
	5 watts (low and high data rate)
packaging:	standard 7" x 7" Voyager spacecraft modules.

The capsule link tape recorder has the following characteristics:

Data rate input (write): 50,000 to 200,000 bits/sec.
Data rate output (playback): 15, 7.5, 3.75, 1.875 kbps

size:	720 cubic inches
weight:	12 pounds
power:	10 watts write
	5 watts playback

5. TELEMETRY SUBSYSTEM

5.1 General

The flight spacecraft telemetry subsystem performs the following functions:

a) Accepts analog, discrete, and binary data from the flight spacecraft subsystems and transducers

b) Time-multiplexes data inputs

c) Conditions and encodes analog data into a seven-bit word

d) Generates synchronization words, frame number words, and data mode words

e) Formats special words and data words according to the data mode selected

f) Provides bit rate selection

g) Combines the coded data with a pseudonoise code sequence to enable ground station bit synchronization

h) Provides data and synchronization subcarrier inputs to the S-band radio transmitter, enabling data transmission to the ground receiving station.

The system consists of two redundant pulse code modulation telemetry encoders, each of which has the following major subassemblies:

- Analog multiplexer
- Analog-to-digital converter
- Capsule data buffer
- PN generator
- Digital multiplexer
- Modulator-mixer.

5.2 Requirements and Design Constraints

5.2.1 Mission Constraints

The flight spacecraft will transmit flight capsule telemetry data during launch through capsule-spacecraft separation and relay capsule telemetry data to earth subsequent to capsule-spacecraft separation. The flight spacecraft will measure and transmit engineering performance data to earth, and transmit science data to earth.

Science payload data requirements are summarized in Table 54.

Table 54. Science Payload Data Rate Requirements

Mission Phase	Minimum	Design Goal
Cruise	2.5×10^6 bits/day = 29 bits/sec	5×10^6 bits/day = 58 bits/sec
Encounter + 6 months	5×10^7 bits/day = 580 bits/sec	10^8 bits/day = 1160 bits/sec

The flight spacecraft telemetry rate will not exceed 15,000 bits per second from two planetary vehicles during simultaneous operation, or 15,000 bits per second from either vehicle.

The flight capsule-flight spacecraft link requires three bit rates: very low ($\approx$10 bits/sec) during launch and cruise, low (100 bits/sec) from capsule separation to atmospheric entry, and high (50 to 200 thousand bits/sec) during capsule terminal descent. Flight capsule-flight spacecraft interfaces will be via a cable, RF link, and tape recorder.

5.2.2 Subsystem Requirements

The telemetry subsystem will accept and process six types of data for input to the radio transmitter as a serial PCM signal: engineering data, real-time science data, real-time science engineering data, recorded science data, flight capsule data, and maneuver data.

Spacecraft engineering data will consist of:

1) Voltage analogs standardized to a three-volt full scale level.

2) Event pulses, counted by the telemetry, the result read out in digital form.

3) Elapsed time between event pulses, counted and read out in digital form.

4) Command detector clock phase counter, read out in digital form.

5) Status indicators, on-off functions that are scanned, and then read out in digital form.

6) Digital words, representing shaft angles and computing and sequencing subsystem words, read out in digital form.

Approximately 400 engineering telemetry channels will be divided into four sampling rate groups. All analog quantities are to be converted to seven-bit words and formatted with other digital words in data frames according to the required sampling rate. Digital words longer than seven bits will be transmitted as two or three telemetry words, as required.

Real-time science data will be gathered, processed, formatted, and presented to the telemetry subsystem in serial digital form by the science subsystem.

Science data will be stored on five tape recorders in the data storage subsystem. Total storage capacity is 2.21×10^8 bits.

Real-time capsule data will be presented to the telemetry in serial digital form, asynchronous to the telemetry data rate. The telemetry subsystem will accept, buffer, and synchronize the data at the telemetry bit rate. The telemetry will also be capable of accepting and synchronizing stored capsule data.

During maneuvers when contact with earth is lost, engineering telemetry data will be stored. These data are routed to the data storage subsystem for playback through the telemetry subsequent to the maneuver.

5.2.3 Design Requirements

a. Bit Rates

Six-bit rates are required:

a) 15,000 bits/sec (orbital operations)

b) 7500 bits/sec (orbital operations)

c) 3750 bits/sec (orbital operations)

d) 1875 bits/sec (orbital operations)

e) 234.4 bits/sec (launch and injection, cruise, maneuver)

f) 7.3 bits/sec (emergency)

b. Data Modes

To match telemetry data transmission to the mission requirements, five data transmission modes are required:

1) Mode 1, engineering data only, includes low rate capsule engineering data. This mode is used during launch, maneuvers, and when required for failure analysis.

2) Mode 2 is a combination of spacecraft and capsule engineering data and real-time science data, such as field and particle measurements. This mode is used during cruise.

3) Mode 3 is the playback of non-real-time data such as stored science instrument data, stored capsule data, and data recorded during maneuvers.

4) Mode 4 is selected engineering data plus C and S memory readout. This mode is used prior to maneuvers for verification of maneuver commands and for checkout of guidance and control equipments prior to use.

c. Engineering Data Format

The basic telemetry engineering data format will be as shown in Figure 84, indicating the prime commutator and the number and location of subcommutators and sub-subcommutators. Four data sampling rates are provided, with the following channels available for data, after fixed word requirements are met:

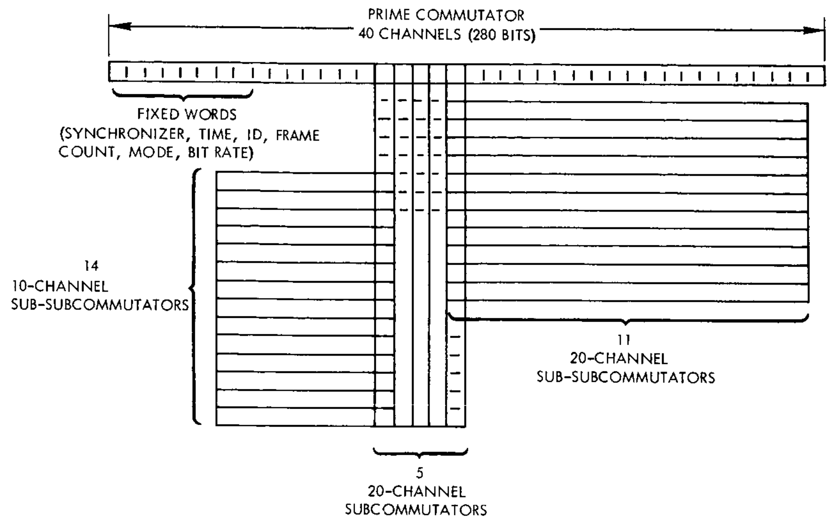


Figure 84. Telemetry Data Format

1) 27 channels sampled at the frame rate

2) 70 channels sampled at 1/20 of the frame rate

3) 126 channels sampled at 1/200 of the frame rate

4) 209 channels sampled at 1/400 of the frame rate

A total of 432 seven-bit channels are devoted to engineering measurements. The manner in which the commutators are implemented is to be flexible enough to permit adjustments between sampling requirements and commutator capacities.

d. Synchronization

The telemetry system is a synchronous two-channel PCM/PSK/PM system. Synchronization is accomplished by a pseudonoise code of 63 bits, cycling at the telemetry word rate and providing bit and word synchronization. Data and the PN code are used to modulate two subcarriers, the outputs of which are linearly mixed to produce a signal for input to the radio transmitter.

The engineering data format includes a 21-bit code at the beginning of each frame enabling rapid frame synchronization in the decommutator. A word of the data frame contains a frame count, enabling identification of subframe and sub-subframe channels, and a data mode word indicating the telemetry mode and data rate.

5.3 Functional Interfaces

5.3.1 Spacecraft Electrical Interfaces

a. Subsystem Engineering Telemetry

Engineering data is received from all subsystems (including telemetry) in one of the following forms:

1) Voltage analogs standardized in each subsystem to 0 to 3 volts full scale.

2) Bilevel pulse signals.

3) Digital words in serial form, at the telemetry bit rate, during a gated time interval. (The gated time interval is a telemetry output, as is the telemetry clock.)

b. Data Storage Subsystem

1) Recorded data is accepted in serial form, at the telemetry bit rate, from the storage subsystem on any of six lines, one for each tape recorder. The line is selected on the basis of a C and S command to the telemetry.

2) Clock pulses are provided to the data storage subsystem, to synchronize the playback data with the telemetry bit rate.

3) A data gap signal is provided to the telemetry for each recorder to indicate the absence of data during playback of a recorder. The telemetry switches to the transmission of real-time science and engineering data during a data gap.

4) An end-of-tape signal is provided to the telemetry to indicate that each recorder is empty. The telemetry reverts to the transmission of real-time science and engineering data.

5) Maneuver data—during maneuvers, spacecraft telemetry data is recorded on the maneuver recorder, for playback after the maneuver.

c. Computer and Sequencer Subsystem

1) The C and S provides the commands listed in Table 55.

2) The C and S provides a digital time word to the telemetry for data identification.

3) The C and S provides event pulses to the telemetry.

Table 55. Telemetry Subsystem Command Requirements

No.	Command	Remarks
1	Mode 1	Engineering data only
2	Mode 2	Engineering and science data
3	Mode 3	Playback data
4	Mode 4	Pre-maneuver data
5	Bit rate 1	15 k-bits/sec
6	Bit rate 2	7.5 k-bits/sec
7	Bit rate 3	3.75 k-bits/sec
8	Bit rate 4	1.875 k-bits/sec
9	Bit rate 5	234 bits/sec
10	Store maneuver data	
11	Select TR 1	Television data
12	Select TR 2	Television data
13	Select TR 3	Spectrometer data
14	Select TR 4	IR scanner data
15	Select TR 5	Fields and particles data
16	Select TR 6	Maneuver data
17	Select capsule data recorder	Playback capsule data

4) The C and S provides a data line for telemetering the C and S memory contents.

5) The telemetry provides clock and "word end" pulses to the C and S to synchronize memory readout.

d. Command Subsystem

1) The command subsystem provides a digital word indicating the number of events which have been transmitted since the last interrogation by telemetry.

2) The command clock is provided to the telemetry for a synchronization indication.

3) The command subsystem provides a command to enable telemetry operation at the emergency bit rate of 7.3 bits/sec.

4) The command subsystem provides backup commands for the C and S commands.

e. Power Subsystem

1) The primary clock frequency of 270 kcps is received from the power subsystem.

2) AC power, 4 kcps is required at 50 volts and 5 watts for each of the PCM encoders.

f. Radio Subsystem

The composite, two-channel, telemetry subcarrier is supplied to the radio subsystem.

g. Science Subsystem

1) Bit rate, word rate, and frame rate are supplied by the telemetry.

2) Data mode indication is supplied by the telemetry.

3) Real-time science data, formatted, and at the telemetry bit rate is supplied by the science subsystem.

5.3.2 Flight Capsule Interfaces

The telemetry-flight capsule interface permits the flight spacecraft to transmit flight capsule data during launch through capsule-spacecraft separation, and to relay capsule data to earth subsequent to separation. The interface has three forms:

a. Hard Wire

Capsule data and clock, prior to separation, is received by the telemetry at a low rate ($\approx$10 bits/sec) via a hard line to the capsule. The capsule data is in serial digital form, asynchronous with the telemetry clock. The telemetry buffers the data for inclusion in the telemetry format.

b. RF Link

Capsule data and clock, after separation, is received by the telemetry at a rate of 100 bits/sec via an RF link. Demodulated data is supplied to the telemetry asynchronous with the telemetry bit rate by the capsule data demodulator. The data is accepted by the telemetry capsule buffer and formatted into the telemetry data stream until the capsule switches to the high data rate used during entry. These data are recorded by the capsule data recorder.

c. Recorder

Recorded capsule data is received from the capsule data recorder during the telemetry playback mode. The high rate capsule data is played back at a slower rate, compatible with the telemetry bit rate. The telemetry provides a clock to synchronize the data playback, and the recorder provides an end-of-tape signal signifying that the recorder is empty.

5.4 Design Description

5.4.1 Subsystem Organization

The telemetry subsystem consists of two redundant PCM encoders, one of which is shown in the block diagram of Figure 85.

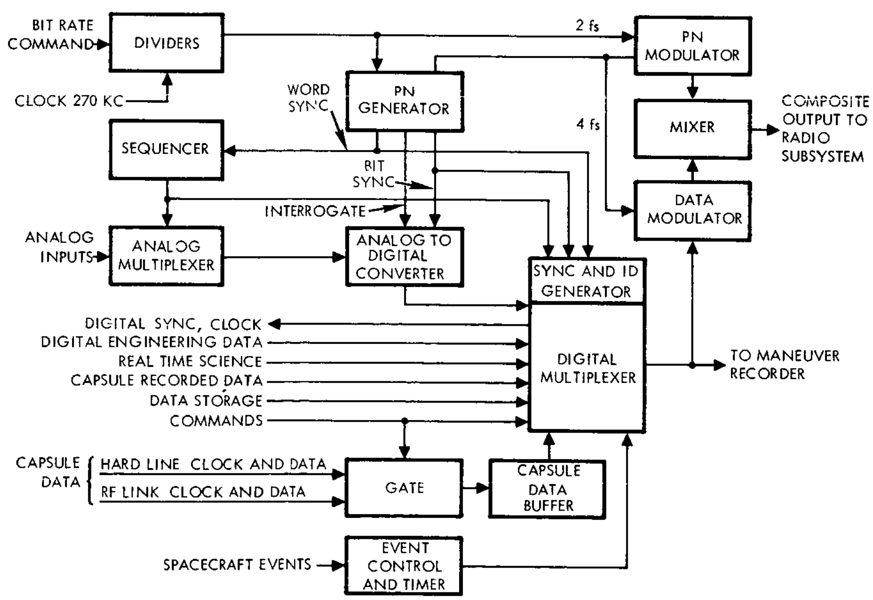


Figure 85. Pulse Code Modulator Encoder

The multiplexer sequentially samples the analog data inputs, presenting them to the analog-to-digital converter for translation into a seven-bit digital word. Digital outputs of the analog-to-digital converter are applied to the digital multiplexer, as are all digital data, capsule data, the real-time science data, and the outputs of the data storage subsystem. The digital multiplexer, under C and S control, formats the telemetry data. The PN generator provides a 63-bit binary sequence for ground-station bit synchronization, which is combined with the serial data stream in the modulator-mixer. The resultant waveform is applied to the radio subsystem for transmission to earth.

a. Analog Multiplexer and Sequencer

The multiplexer is arranged to sample the various input data at four different sampling rates via subcommutation and sub-subcommutation. It is arranged to provide the telemetry data format of Figure 84 and contains 32 switch decks, each deck consisting of 10 switches and a series isolation switch. Four of the decks can be considered as a single 40-position multiplexer running at one revolution per unit time. The subcommutators advance one position for each complete revolution of the prime commutator. Similarly, the sub-subconnutators advance one step for each revolution of the subcommutators. The decks, therefore, provide sampling as follows:

1) 40 channels at 1 channel per unit time
2) 100 channels at 1 channel per 20 units time
3) 140 channels at 1 channel per 200 units time
4) 220 channels at 1 channel per 400 units time

Since each sample of data is encoded into a seven-bit binary number, the actual sampling rate depends upon the bit rate. Further, the sampling rate depends upon the data mode, being highest in Mode 1 with only engineering data transmitted, and somewhat lower in Mode 2, when science data is mixed with the engineering data.

The series isolating switch associated with each deck serves to isolate error-causing leakage current within each deck and also isolates faults to the deck in which they occur.

The multiplexer is controlled by the sequencer, which sequences the data switches through the data format. The sequencer is essentially a decoded counter which is stepped at the telemetry word rate, obtained from the PN generator.

b. Analog-to-Digital Converter

The data sampled by the multiplexer is applied to the analog-to-digital converter (ADC), where the data voltage amplitude is converted to an equivalent seven-bit binary code.

Conversion takes place at 150 k-bits/sec, upon receipt of an interrogate pulse from the PN generator. When the conversion is completed, the binary word is placed in the ADC output register, to be shifted out into the PCM bit stream at the data clock rate.

The conversion technique is successive approximation or "half-split" encoding, in which the input voltage is approximated by a precision reference through successive trials, one at each bit time. A seven-bit conversion can be completed in seven clock intervals. The rapidity of conversion obviates the need for a sample and hold circuit at the encoder input.

The converter, shown in Figure 86, consists of a comparator, a digital-to-analog converter, and two registers, one for timing, the other for control of the digital-to-analog converter ladder switches.

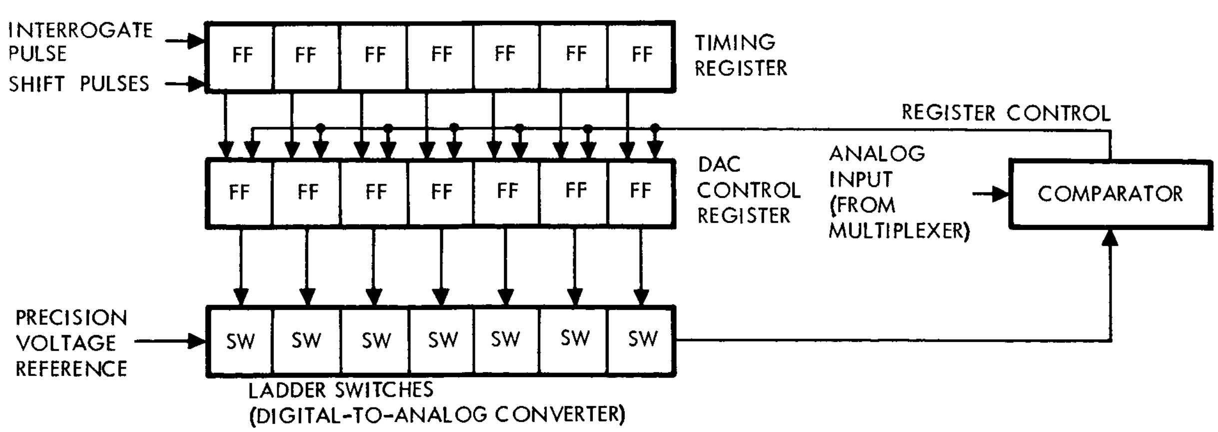


Figure 86. Analog-to-Digital Converter

c. PN Generator

The telemetry data rate is derived from a timing reference clock of 270 kcps supplied by the power subsystem. This clock is applied to the telemetry frequency dividers, and is divided down to produce the PN generator rate and two subcarrier rates used in the PN and data modulators.

The pseudonoise generator provides:

1) A 63-bit pseudorandom code

2) Bit sync pulses every ninth bit of the PN code, and a word sync pulse once every 63-bit cycle.

3) An interrogate pulse, preceding word sync, to initiate an analog-to-digital conversion cycle.

The PN generator is a six-stage shift register counter, with the outputs of the first and sixth stages half-added and applied to the input of the first stage. The generator is driven by clock pulses nine times the frequency of the data bit rate. The generator output is a cyclic 63-bit code, with every nine PN bits corresponding, in time, to one telemetry data bit as follows:

100000111	111010101	100110111	011010010	011100010	11100101	000110000
data bit	data bit	data bit	data bit	data bit	data bit	data bit

A gating matrix, interrogating the counter, produces the data clock stream. In addition, one gate produces a word pulse every seven data bits, and another gate produces an interrogate pulse once per counter cycle. The multiplexer sequencer steps on the word pulse, while each interrogate pulse initiates an analog-to-digital conversion cycle.

d. Digital Multiplexer

The digital multiplexer, illustrated in Figure 87, performs two major functions. First, it formats analog and digital engineering data. Second, it selects the telemetry data source. Other functions include special word generation, frame count and frame sync generation, and the generation of control signals to interface with external sources

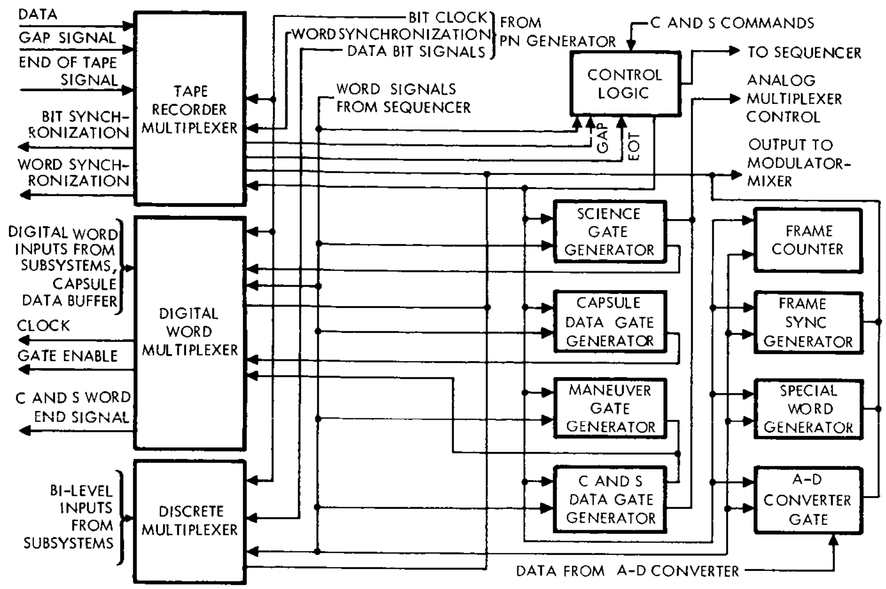


Figure 87. Digital Multiplexer Block Diagram

of digital data. The multiplexer receives a clock at the telemetry data rate, a clock at the word rate, and data bit signals (sever per word) from the PN generator. The analog multiplexer sequencer supplies word gate signals for selected words in the prime frame and subframes.

The tape recorder multiplexer selects the tape recorded output from the data storage subsystem when the telemetry is in Mode 3, data playback.

The digital word multiplexer gates data from digital engineering sources, the serial real-time science data from the science subsystem, serial data from the capsule data buffer and memory words from the C and S subsystem.

Control for the digital word multiplexer is derived from C and S commands, and synchronizing signals are obtained from the sequencer and the various data gate generators: Science data gate, capsule data gate, maneuver data gate, and C and S data gate.

The discrete multiplexer accepts one-bit bilevel inputs from the spacecraft subsystems. These are sampled and combined, utilizing wordgate data bit signals.

Circuits are included to generate a frame count, frame sync word and special words at appropriate places in the frame, and are timed by word signals from the sequencer.

e. Modulator-Mixer

The PN generator waveform and the PN clock (2fs) are applied to a biphase modulator, in which the PN waveform phase shift keys the 2fs subcarrier square wave. In a like manner, the data wave train from the digital multiplexer phase shift keys a 4fs subcarrier square wave. The outputs of these two modulators are linearly mixed to produce the two-channel PCM output.

The biphase modulators are modulo-two adders or "exclusive-or-gates" in which the subcarrier output has the same phase as the subcarrier input when the modulating signal is a one, and is of the opposite phase when the modulating input is a zero. The mixing is an operational summing amplifier, with mixing ratios depending on the telemetry bit rate. The output amplitude is 2 to 3 volts, sufficient to modulate the radio transmitter.

f. Capsule Data Buffer

The data buffer accepts asynchronous capsule data and clock, accumulates the serial data bits into words, and supplies the words to the digital multiplexer upon receipt of gated time interval signals. The buffer accepts data from 5 to 100 bits/sec. A command-controlled gate at the input of the buffer selects either the hard wire or capsule demodulator data source.

The data buffer will be interrogated at a rate higher than the buffer input rate. The buffer will accumulate input data until it is assembled into 6-bit words, then add a seventh bit as a tag indicating "good data" before presenting it to the digital multiplexer upon receipt of a gate signal. If the multiplexer gate signal occurs before the buffer has accumulated 6 new bits, the buffer will present a word containing all zeros and no tag to the multiplexer.

5.4.2 Data Transmission Modes

a. Mode 1, Engineering Data

When operating in Mode 1, the telemetry sequences through the 40 channels of the prime commutator, and then repeats. The organization of the commutator is such that only engineering measurements will be sampled (including flight capsule engineering data) with all data points having been sampled at least once after 400 cycles of the prime commutator. Figure 84, the telemetry data format, illustrates the manner in which data points are sampled.

During the portions of the maneuver sequences in which contact with earth is lost, the telemetry data is recorded. At these times the telemetry sequences through the 40-word prime frame twice, and then puts out no data for a time period equal to one frame. The data gap thereby provided enables the transmission of real-time engineering data during playback of the stored maneuver data.

Sampling rates of the four groups of available data channels are given in Table 56 for the emergency low data rate of 7.3 bits/sec, and the cruise data rate of 234.4 bits/sec. In addition, the sampling rates for maneuver data stored on tape at 234.4 bits/sec are given.

Four channels of the engineering prime commutator are assigned to capsule data. Since each channel contains 6 capsule data bits plus a one-bit tag, 24 bits of capsule data are transmitted during each engineering data frame. At a data rate of 234.4 bits/sec, the telemetry has the capability of transmitting 20 bits/sec of capsule data.

Table 56. Mode 1 - Engineering Data Sampling Rates

Channels	Sampling Rate for Real-Time Data 234.4 bits/sec	Sampling Rate for Real-Time Data 7.3 bits/sec	Sampling Rate for Stored Data 234.4 bits/ec	Remarks
27	1.2 sec	38.4 sec	1.8 sec	Prime commutator
70	24 sec	12.8 min	36 sec	Subcommutator
126	4 min	2 hrs	6 min	Sub-sub
209	8 min	4 hrs	12 min	Sub-sub

b. Mode 2, Engineering and Low Rate Science

In Mode 2, the telemetry sequences through the 40 words of the prime commutator and then accepts 560 bits (equivalent to 80 words) of real-time science data from the science subsystem making a frame 840 bits in length (Figure 88).

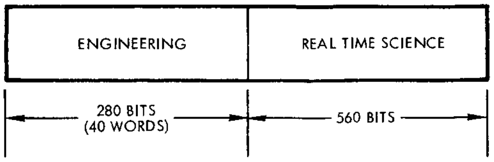


Figure 88. Mode 2—Telemetry Format

In Mode 2, engineering data is sampled one-third as often in Mode 1. Hence, the sampling times in Table 56 multiplied by three provide the Mode 2 sampling rates.

Capsule data, occupying 4 channels of the engineering prime commutator, can be transmitted at a maximum rate of 6-2/3 bits/sec at the data rate of 234.4 bits/sec used during cruise. During orbit, when the capsule is separated, capsule data can be transmitted at a rate of 108 bits/sec at a telemetry data rate of 3.75 K bits/sec. This is the minimum data rate at encounter if the medium gain antenna is used.

c. Mode 3, Stored Data Read-Out

One telemetry data mode is provided for the playback of recorded data, whether it be science, maneuver, or capsule data. A C and S command to the telemetry will select the tape recorder output to be telemetered. During playback, the telemetry system monitors the data gap signal from the recorder, and when no data is present Mode 2 engineering data and real-time science data are sampled and inserted in the telemetry bit stream. Gaps in recorded science data are controlled by the science subsystem, and should occur, in picture transmission, after every line of the picture. The gaps must be long enough and often enough to allow sampling of enough real-time engineering and science data to provide the minimum acceptable data rate. Temporary storage (between gaps) of real-time science data must be provided in the science subsystem. Science recording, other than pictures, requires similar data gaps, while maneuver data will be provided with gaps by the telemetry. Capsule data playback, since it occurs relatively infrequently, will not require gaps.

Details of the data gap for one picture and for one line of one picture, are illustrated in Figure 89 and are shown typically for television recording. A picture contains 6.2×10^3 bits. It is proposed that 140 bits, or one-sixth of a Mode 2 frame, be transmitted between each line of a picture. When other gaps appear as a result of the science instrument, such as the video erase gap, additional science and engineering data is telemetered.

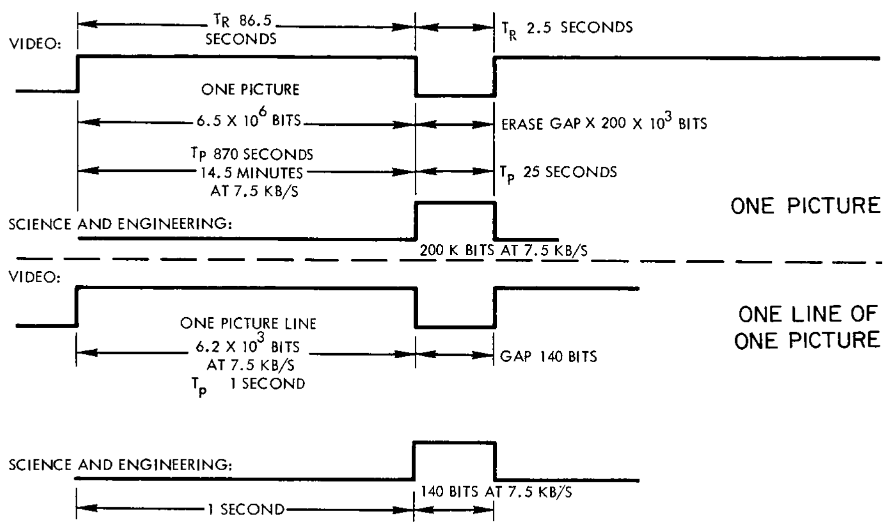


Figure 89. Data Gap

Television and other science data may be reproduced at one of the telemetry data rates used during orbital operations (15, 7.5, 3.75, or 1.875 k-bits/sec). Figure 89 illustrates the approximate times for the nominal rate of 7.5 k-bits/sec. Engineering data sampling rates at the orbital data rates are given in Table 57 for the gap length discussed.

Table 57. Mode 3 - Engineering Data Sampling Rates

Number of Channels	Date Rate 15 k-bits/sec	7.5 k-bits/sec	3.75 k-bits/sec	1.875 k-bits/sec	Maneuver Playback 234.4 k-bits/sec	Remarks
27	2.5 sec	5 sec	10 sec	20 sec	10.8 sec	Prime Commutator
70	50 sec	100 sec	3.3 min.	6.6 min.	3.6 min.	Subcommutator
126	8 min.	16 min.	32 min.	1 hr	36.0 min.	Sub-sub
209	16 min.	33 min.	1 hr	2 hrs	72.0 min.	Sub-sub

Also given in Table 57 are the sampling rate of real-time engineering data during the playback of stored maneuver data. The real time science main frame (560 bits) are sampled as often as the engineering prime commutator.

Figure 90 shows the timing relations that might exist when two tape recorders are played back. A C and S command to the telemetry selects the outputs of tape recorder No. 1, and the telemetry transmits science and engineering data until the data gap signal from the selected recorder indicates the presence of stored data. Then the telemetry transmits stored data, interleaving real-time data in the gaps, until the recorder is emptied. A signal from the C and S may then select tape recorder No. 2.

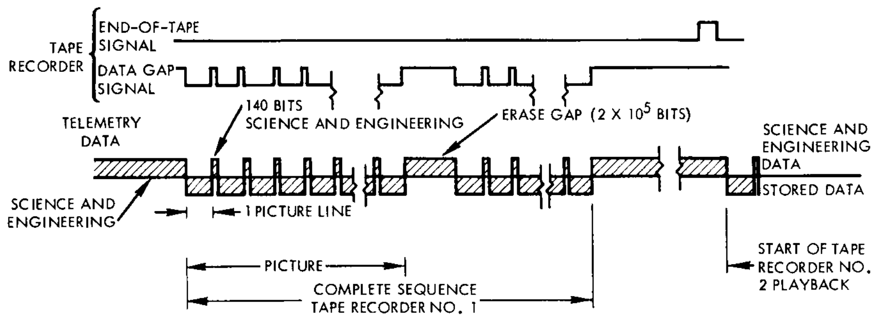


Figure 90. Mode 3 Operations

When a tape recorder is selected and the stored data is initially transmitted, the frame size changes and consequently frame synchronization must be reaquired on the ground. The gap signal is used to control the real-time telemetry frame, and since the gap signal is exactly one-sixth of the real-time frame, synchronization can be maintained during the playback of stored data while interleaving real time data.

d. Mode 4, Premaneuver Checkout

Before maneuvers, the entire computing and sequencing subsystem memory of 256 20-bit words is telemetered to earth to verify the proper receipt of maneuver sequence data. In addition, data points in the guidance and control subsystem, undergoing the thrust vector control test, are sampled at twice the normal rate to enable selection of the primary or backup control components for use during propulsion operation.

5.4.3 Redundancy Mode

The subsystem contains two PCM encoders, each capable of performing the mission. In the event of a failure in an encoder, power is switched by ground command to the other encoder. Power switching is in the power distribution unit of the power subsystem, responding to commands received from the command subsystem.

5.5 Parameter and Performance Summary

5.5.1 PCM Encoder Performance

Table 58 indicates the number of channels available and their sampling rates for each of the telemetry bit rates and modes. As discussed in Section 5.4.2, four transmission modes are available.

Bit and word synchronization are provided by a 63-bit pseudonoise code. The PN code biphase modulates a subcarrier, as does the data. These two modulated subcarriers are added to form the composite telemetry output. Table 59 shows the subcarrier frequencies used for each bit rate.

The data subcarrier frequency is 18 times the data bit rate, and the PN subcarrier frequency is nine times the data bit rate. Frame synchronization is provided by a 21-bit code, occupying the first 21 bits of each frame of engineering data. Sub and sub-subcommutator synchronization is provided by word 4, the subcommutator ID word.

Table 58. Channels and Sampling Rates

Characteristic	Number of Channels	Launch Maneuver Mode 1 234.4 bits/sec	Cruise Mode 2 234.4 bits/sec	Low Bit Rate Mode 1 7.3 bits/sec	Orbit Mode 3 15 k-bits/sec	Orbit Mode 3 7.5 k-bits/sec	Orbit Mode 3 3.75 k-bits/sec	Orbit Mode 3 1.875 k-bits/sec
Prime commutator	27	1.2 sec	3.6 sec	38.4 sec	2.5 sec	5 sec	10 sec	20 sec
Subcommutator	70	24 sec	1.2 min.	12.8 min.	50 sec	100 sec	3.3 min.	6.6 min.
Sub-subcommutator	126	4 min.	12 min.	2 hrs	8 min.	16 min.	32 min.	1 hr
Sub-subcommutator	209	8 min.	24 min.	4 hrs	16 min.	33 min.	1 hr	2 hr

Note: Table gives time between successive samples of a data point.

Table 59. Bit Rate Versus Subcarrier Frequencies

Data Clock Frequency	PN Code Clock Frequency	Data Subcarrier Frequency	PN Code Subcarrier Frequency
15 kcps	135 kcps	270 kcps	135 kcps
7.5 kcps	67.5 kcps	135 kcps	67.5 kcps
3.75 kcps	33.75 kcps	67.5 kcps	33.75 kcps
1.875 kcps	16.875 kcps	33.75 kcps	16.875 kcps
234.4 cps	2110 cps	4219 cps	2110 cps
7.3 cps	65.88 cps	131.7 cps	65.88 cps

Each PCM encoder of the telemetry subsystem will weigh less than 4 pounds and occupy a volume of 113.4 cubic inches, with outside dimensions of 7 x 6 x 2.7 inches. The subsystem requires 5 watts of power, maximum, supplied as a 50-volt p-p, 4 kcps square wave, and requires a clock frequency of 270 kcps ± 0.01 per cent, supplied by the power subsystem.

Analog inputs are encoded with a linearity of ± 0.5 per cent of full scale; analog measurement accuracy is ± 1 quantization level. To maintain the modulation index constraint, the amplitude of the telemetry subsystem output is stable to better than ± 3 per cent, with ± 1 per cent the design goal.

5.5.2 Reliability Assessment

The telemetry subsystem contains two redundant PCM encoders, either of which can accomplish the mission objectives. Selection of the encoder in use at any time is accomplished through the command subsystem. Based on a parts population analysis, the total estimated failure rate for one PCM encoder is:

$$\lambda_{PCM} = 18,725 + 3,855 = 22,580 \text{ failures}/10^9 \text{ hrs.}$$

The realiability from launch to the end of one month in Mars orbit is:

$$r = 0.886.$$

For two encoders in parallel redundancy, the success probability is:

$$R = 1 - (1-r)^2 = 0.987.$$

6. DATA STORAGE SUBSYSTEM

6.1 Subsystem Description

The Voyager spacecraft data storage subsystem performs the following functions:

- Records planetary science data
- Records fields and particles data
- Records spacecraft maneuver data
- Plays recorded data back synchronously through the telemetry subsystem.

The subsystem contains six tape recorders, each assigned a separate function. The recorders and their interfaces are shown in Figure 91 Each recorder is independent of the others, with separate power, control, and signal lines.

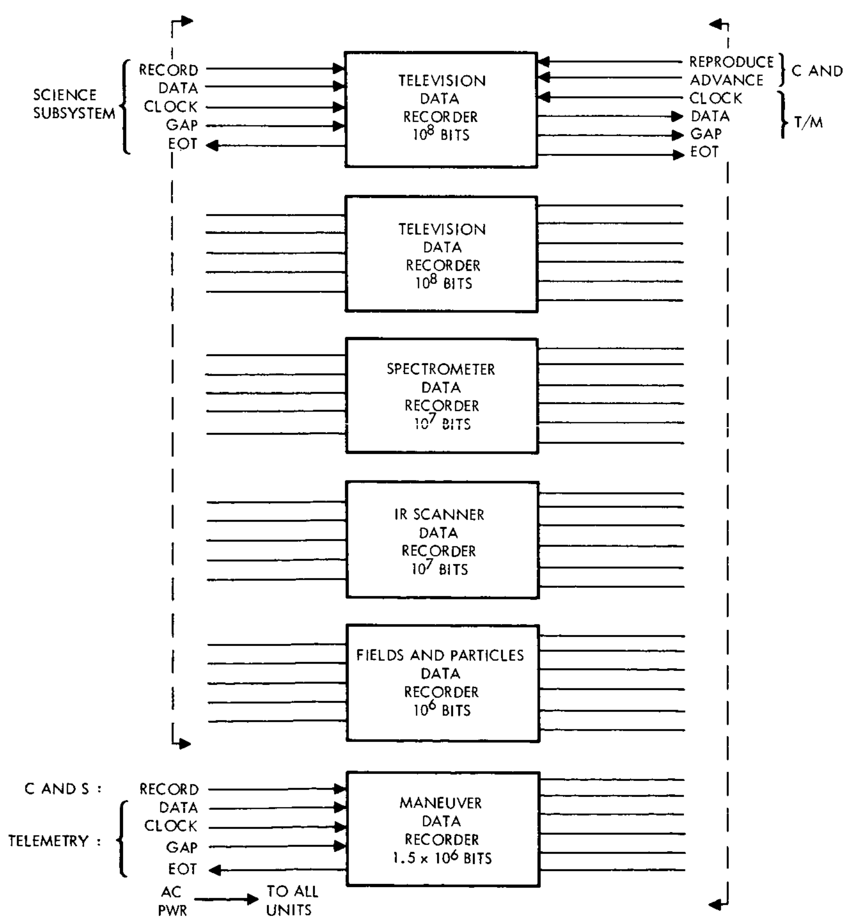


Figure 91. Data Storage Subsystem Block Diagram

6.2 Requirements and Design Constraints

6.2.1 Subsystem Performance Requirements

The flight spacecraft is required to support the science mission, telemetering the science data to earth. To support this task, the data storage subsystem will have a total science storage capacity of 2.21×10^8 bits. This requirement is further delineated as follows:

- Television data recorder: 2 recorders, each with a capacity of 10^8 bits
- Spectrometer data recorder: 1 recorder with a capacity of 10^7 bits
- IR scanner data recorder: 1 recorder with a capacity of 10^7 bits

The tape recorders shall accept data at the following rates:

- Television data: 90 k-bits/sec
- Spectrometer data: 2 k-bits/sec
- Scanner data: 5 k-bits/sec

Each recorder can be played back synchronously at rates of 15, 7.5, 3.75, or 1.875 k-bits/sec.

A recorder with a capacity of 1.5×10^6 bits will be provided to store data accumulated during maneuver. Recording and playback of the stored data will be at 234.4 bits/sec.

Fields and particles data are provided by the science subsystem at 700 bits/sec.* The stored data are reproduced and telemetered at rates of 15, 7.5, 3.75, or 1.875 k-bits/sec. A rate of 10^6 bits of storage will be provided for these data.

6.2.2 Design Requirements

The tape recorders will be capable of recording data, clock, and a data gap signal. The science subsystem will supply these signals for science data and format the data. Data will be in serial digital form. Recording and playback will be initiated and halted by pulse-type commands.

*Low data rate fields and particle data at 70 bits/sec are transmitted in real time by the telemetry subsystem. High data rate associated with solar flare activity (700 bits/sec assumed) is recorded.

The recorders will embody a servo-speed control system for controlling the playback tape speed and synchronizing the recorded clock and data with the telemetry data rate. The recorders will supply the telemetry subsystem with a data gap signal, indicating the absence of data on the recorder output. To permit the reproduction of stored data more than once, the recorders will incorporate a tape advance mode, initiated and halted by pulse-type commands.

All required input buffering and data formatting for the science data will be performed by the data automation equipment (DAE). The science subsystem will furnish signals to control the start and stop of the science data tape recorders in the record mode. The C and S subsystem will furnish signals to control the start and stop operation of the tape recorders in the playback mode. A signal from each science tape recorder will be furnished to the science subsystem, indicating that the tape is loaded.

6.3 Functional Interfaces

6.3.1 Interfaces with Telemetry Subsystem

The tape recorders provide the following signals to the telemetry subsystem:

- Data: A NRZ flow of stored data from each recorder at the prevailing telemetry bit rate
- Data gap signal: A NRZ signal from each recorder indicating the absence of data on the recorder output
- Engineering data: Three types of engineering data on the status of each recorder: recorder pressure; recorder temperature; and status discretes (read mode, write mode end-of-tape, tape motion).

The telemetry subsystem provides the following signals to the tape recorders:

- Bit sync: A signal at the prevailing telemetry bit rate to clock data out of the recorder
- Maneuver data: Data to the maneuver data recorder is presented as a 234.4-bit/sec NRZ train
- Telemetry clock: A 234.4-bit/sec clock signal is stored in the maneuver recorder during the recording of maneuver data

- Data gaps: A NRZ signal whenever gaps occur in the maneuver data is stored on the maneuver recorder during maneuvers.

6.3.2 Interfaces with Science Subsystem

One signal is provided by the recorders to the DAE: a signal from each indicating that it is full. Four signals are provided by the science subsystem to the recorders as follows:

- Science data: Serial digital data are supplied by the DAE at the rate of the associated science instrument.
- Bit sync: A clock signal is supplied by the DAE at the corresponding data rate and stored with the data.
- Start-stop signal: A pulse-type command is sent to each recorder to start it in the record mode and another to stop it.

6.4 Design Description

6.4.1 Subsystem Organization

As shown in Figure 91, the subsystem is composed of six independent recorders, with separate interfaces and containers. Where a single line interfaces with several recorders, such as the AC power or the telemetry clock, isolating elements insure that a failure in any recorder will not affect the operation of the other recorders. Each recorder has an independent input line and the subsystem presents six output lines to the telemetry. The recorder playback selection is performed by C and S command with backup via the command subsystem.

6.4.2 Mission Operation

a. Prelaunch, Launch

All recorders and corresponding instruments are functionally exercised to determine their operational status during prelaunch tests. During launch, the tape recorders are not operating.

b. Cruise

In the event of solar flare activity, the fields and particles data recorder is actuated by the DAE. Playback is initiated by ground command through the C and S subsystem. Other science recorders are dormant during cruise.

c. Maneuver

Prior to a maneuver, the telemetry subsystem is switched to Mode 1, the transmission of engineering data, at a bit rate of 234.4 bits/sec. In the maneuver, contact with earth may be lost for a total worst-case time of approximately 100 minutes. In addition to being transmitted, telemetry data is stored on the maneuver tape recorder for playback subsequent to the maneuver at a data rate of 234.4 bits/sec. Control of the maneuver tape recorder in the recording and playback sequence is by command of the C and S.

To permit the telemetering of real-time data during the playback of maneuver data, the maneuver data is recorded with a 33-per cent data gap provided by the telemetry. The telemetry, during playback, senses the gaps and inserts real-time science and engineering data.

d. Orbital Operation

A possible orbital science storage-telemetry sequence for a typical orbit is shown in Figure 92. Data is recorded during specific portions of the orbit and played back through the telemetry during subsequent portions of the orbit. The record sequence of each tape recorder

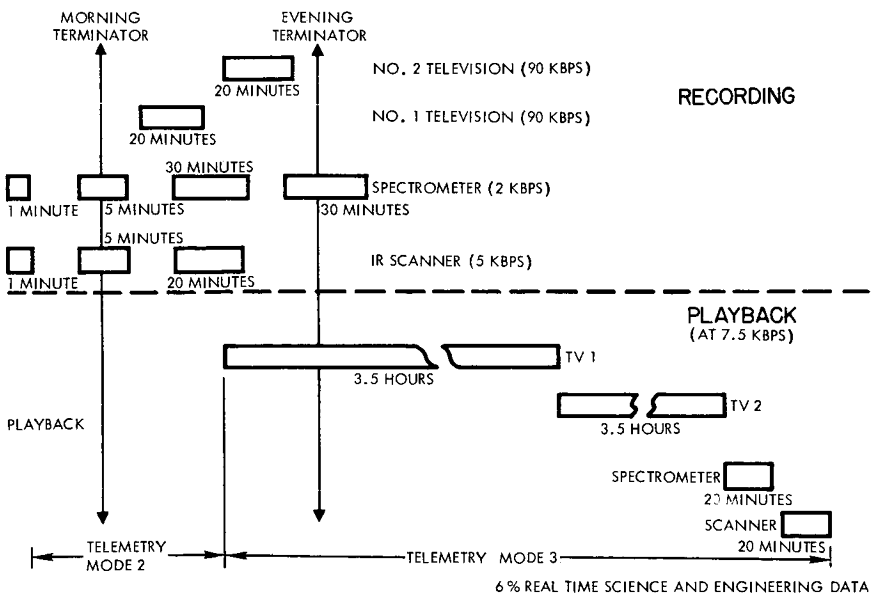


Figure 92. Representative Planetary Storage - Telemetry Sequence

will be determined by the science instrument requirements, and the start-stop commands will be provided by the science subsystem. The playback sequence, occupying the remainder of the orbit, takes longer than the record sequence and is controlled by C and S commands.

Science data is recorded with data gaps formatted by the science subsystem. The telemetry tape recorder interface is essentially through the data gap signal. The telemetry detects the data gap during playback and transmits real time engineering and science data during the data gap period. Control of recording and playback of the fields and particles data is the same as for planetary science.

6.4.3 Alternate Modes of Operation

In the event of a recorder failure, the recording and playback sequences may be altered, omitting the failed recorder, and data storage operation continues in a degraded mode, without the function of the failed recorder. The TV tape recorders are redundant, i.e., the DAE can switch either camera to either recorder.

6.4.4 Tape Recorder Design

a. Tape Transport

The peripheral belt drive for use in all recorders is shown in Figure 93. The tape, stored on two flangeless reels, is driven by a belt which goes around the periphery of the two reels. The belt is, in turn, driven by a differential capstan system, with the capstan at the bottom of the figure going faster than the capstan at the top of the figure. This differential produces increased tension in the portion of the belt around the takeup reel; hence an increased velocity in the belt and a speed differential between the tape takeup and supply reels. This differential develops the tape tension across the heads and the tape reel winding tension.

The differential capstans are driven by one of two motors, one for recording, the other for playback. The motors are coupled to one of the two capstans through a clutch so that only one motor is in the drive system at a time. The differential speeds of the two capstans are obtained by using a slightly different pulley dimension on one capstan.

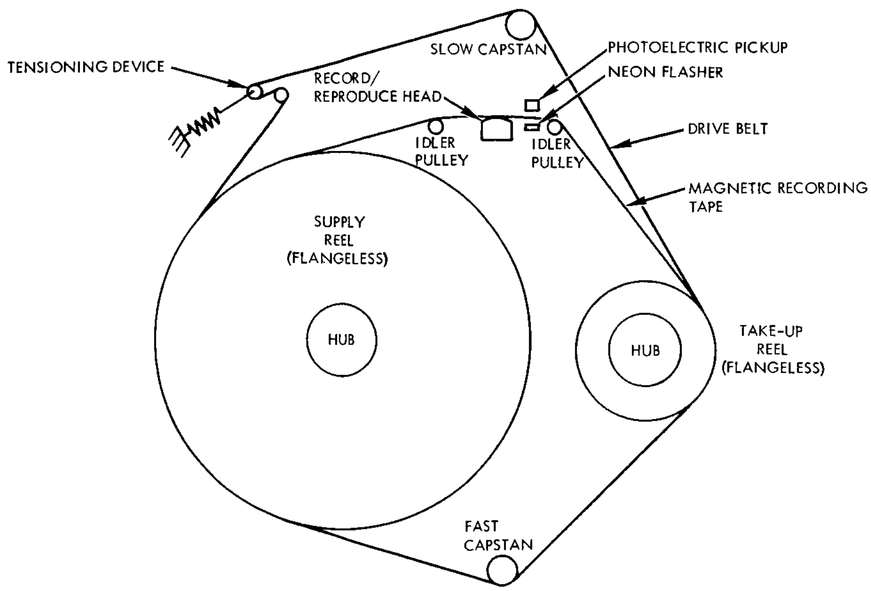


Figure 93. Peripheral Drive System

Also shown in Figure 93 is the end-of-tape detector: a flashing neon bulb and photoelectric detector. The magnetic oxide is stripped from a portion of the tape near the end of the tape pack. The end-of-tape signal is indicated when this section passes between the neon flasher and the photoelectric detector.

b. Electronics

A block diagram of a tape recorder is shown in Figure 94 Recording is in parallel; hence the serial input to the recorder is entered into a shift register, then gated to the head drivers. The recording format is biphase saturation, eliminating the requirement for an erase head.

During playback, the data signals are amplified and held in the skew register and then transferred to the output register (effectively removing any timing errors between tracks). The serial output is formed by shifting the contents of the output register with the telemetry bit rate clock.

Recording speed is established by a synchronous record motor driven by 400 cps from the power subsystem. In order to provide readout synchronous with the telemetry bit rate, playback speed is controlled by a servo system comparing the phase of the recorded clock signal with the phase of the telemetry clock.

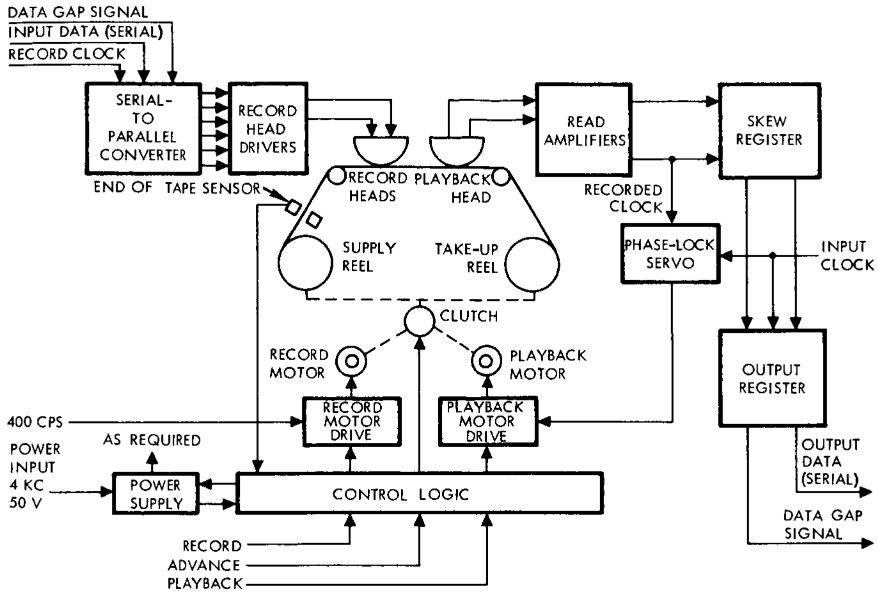


Figure 94. Tape Recorder Block Diagram

c. Power and Command

Each recorder contains a transformer-rectifier, converting the 4-kc AC power input to the required DC voltages. Short circuit protection will limit the current in the event of a short circuit within the recorder.

Each recorder requires the following pulse commands:

- Record: Start and stop
- Reproduce: Start and stop
- Advance: Start and stop

In addition to controlling tape motion, these signals will be used to turn DC power on and off as required by the recorder.

6.5 Parameters and Performance Summary

6.5.1 Specification Summary

Table 60 presents abbreviated specifications for all six tape recorders. The specifications summarize data rates, capacities, and physical characteristics of each unit.

6.5.2 Reliability Assessment

The failure rate of a single tape recorder from operation to encounter plus one month is estimated to be 12,200 failures per 10^9 hours. This

Table 60. Preliminary Specifications (Data Storage Subsystem Tape Recorder)

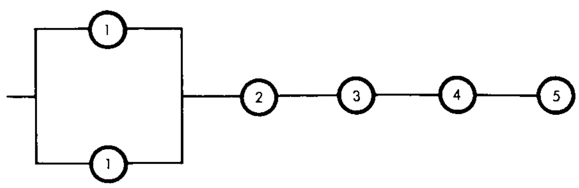
Function	Television	Spectrometer	IR Scanner	Fields and Particles	Maneuver
PERFORMANCE CHARACTERISTICS					
Capacity (bits)	10^8	10^7	10^7	10^6	1.5×10^6
Input rate (k-bits/sec)	90	2	5	0.7	0.234
Output rate (k-bits/sec)	15, 7.5, 3.75, 1.875	15, 7.5, 3.75 1.875	15, 7.5, 3.75, 1.875	15, 7.5, 3.75, 1.875	0.234
Tape width (in.)	0.5	0.5	0.5	0.25	0.25
Tape length (ft)	1200	120	120	84	126
Tracks	9*	9*	9*	3*	3*
Bit density (lbs/in.)	1000	1000	1000	1000	1000
Record method	Biphase	Biphase	Biphase	Biphase	Biphase
Record speed (in./sec)	13	0.286	0.715	0.7	0.234
Playback speed (in./sec)	2.14, 1.07, 0.53, 0.26	2.14, 1.07, 0.53, 0.26	2.14, 1.07, 0.53, 0.26	15, 7.5, 3.75 1.875	0.234
Bit dropout rate (bits)	1 in 10^5	1 in 10^5	1 in 10^5	1 in 10^5	1 in 10^5
Power: record (watts)	12	10	10	8	8
playback (watts)	5	4	4	4	4
Input voltages	4 k-cps, 400 cps	4 k-cps, 400 cps	4 k-cps, 400 cps	4 k-cps, 400 cps	4 k-cps, 400 cps
PHYSICAL CHARACTERISTICS*					
Size (in.)	10 x 12 x 5	10 x 12 x 5	10 x 12 x 5	10 x 12 x 5	10 x 12 x 5
Weight (lb)	12	12	12	12	12

*One track clock, one track data gap—remainder are data tracks.

Requirements:
1. Playback shall be synchronous with telemetry data rate
2. End-of-tape shall be provided for recording and playback
3. Commands shall be: record, playback, and advance
4. Electromechanical elements shall be in a pressurized container
5. The angular momentum change shall not exceed ± 0.1 ft-lb-seconds between zero and full speed
6. Telemetry monitoring points: temperature, pressure, status

estimate is based on a parts count using partial circuit redundancy. The reliability potential corresponding to encounter plus one month is 0.936 for a single recorder.

The reliability block diagram of the data storage subsystem is shown below.



The two television data recorders are shown as redundant, assuming the DAE has the capability of using either one or both recorders. Mission success probability for the subsystem is:

$$R_1 = \left[1 - (1-r)^2\right] r^4 = 0.764.$$

If mission success is dependent only on the two television data recorders, the success probability is:

$$R_2 = 1 - (1-r)^2 = 0.996.$$

Further consideration of the possibility of multiple use of other recorders is required, depending on the implementation of the science subsystem.

7. COMMAND SUBSYSTEM

The Voyager security guidelines indicate that the command word structure and subcarrier frequencies carry the security classification of confidential. To avoid classifying Volume 2, the command subsystem description has been extracted and is submitted under separate cover.

8. COMPUTING AND SEQUENCING SUBSYSTEM

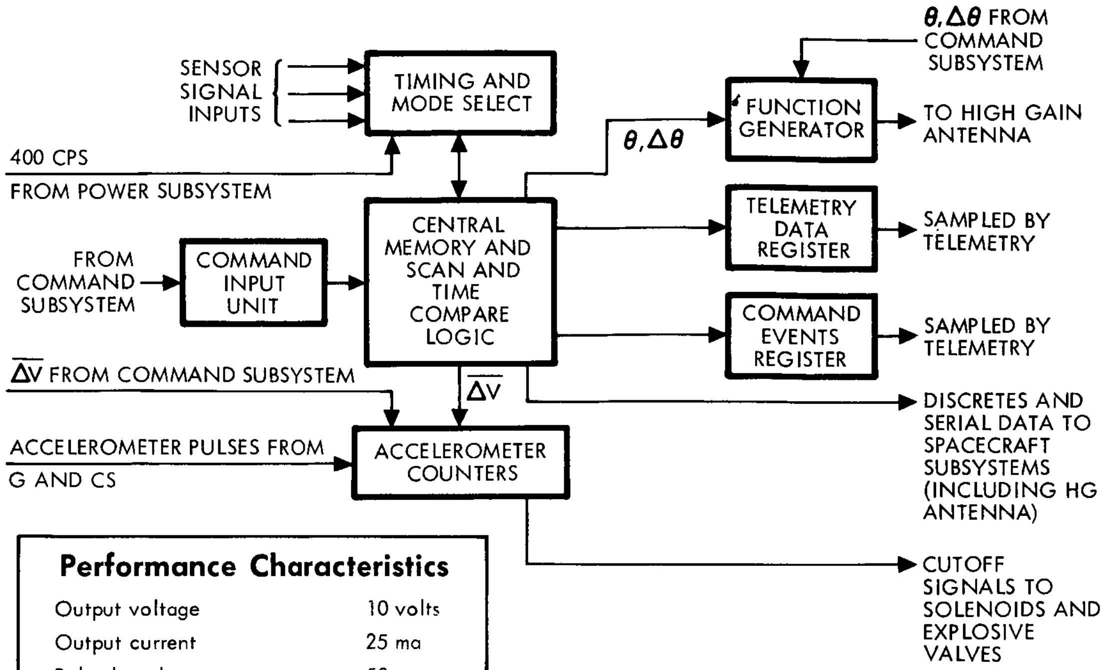
8.1 General Description

The computing and sequencing subsystem includes a special purpose sequencer as a primary unit and a backup sequencer as a secondary unit for those functions for which the command subsystem cannot provide an effective backup. The primary sequencer consists of a system of clocks, a central memory, a command input unit, frequency divide logic, a function generator, accelerometer pulse counters, signal input logic, a memory data telemetry register, and a command events telemetry register. The functional relationships of these units are shown in Figure 95. The backup system consists of a less extensive system of clocks, a central memory, a command input unit, frequency divide logic, a memory data telemetry register, and simplified signal input logic.

Under normal operating conditions the input unit of the primary sequencer accepts quantitative commands routed to it by the command subsystem, performs a format check, and stores the data in the appropriate cells in memory. The memory is a nonvolatile, ferrite core unit with a capacity of 256 20-bit words. Scan and time-compare logic cycles through the memory cells to find commands to be issued. The stored quantity in each memory location consists of a sequence identification (mode) tag and the time relative to that sequence at which the command is to be issued. The function to be performed is indicated by the memory address of the stored quantity. When a command mode and time coincidence is detected, an output is issued to the appropriate subsystem via a hard line connection. The outputs are in the form of command discretes and serial data.

Commands out of the C and S are organized into distinct timed sequences controlled by the clock references. The sequences are initiated individually at times determined either by ground command, stored commands, or specific mission events, such as launch, separation, or terminator crossing. The stored commands in each sequence are time-referenced to the start of the sequence and are issued at the indicated elapsed time from the point of initiation of the associated counter controlling the sequence.

PRELIMINARY SPECIFICATION



Performance Characteristics

Output voltage	10 volts
Output current	25 ma
Pulse length	50 ms

Physical Characteristics

	Primary	Backup
Weight	16 pounds	14 pounds
Power	23 watts	18 watts
Volume	400 inches^3	360 inches^3
Reliability	0.94	
Memory size	256 bits	256 bits

Sequencing Capability

Planetary vehicle and spacecraft maneuvers

Orbit science sequencing

Interplanetary cruise operations

Post separation sequencing

Antenna alignment confirmation

Canopus acquisition and magnetometer calibration

Special Features

- Accelerometer counter can be loaded independently by quantity from command subsystem if necessary and continue to operate even if memory fails
- Backup maneuver sequencing off of alternate timer is available
- Engine cutoff can be controlled separately by timed command out of central memory for each engine burn mode
- Independent and isolated accelerometer counters and gating systems are available
- Backup memory and clock system provides additional maneuver capability and antenna pointing
- Function generator can continue to operate even if memory fails. Register can be loaded either from ground or C and S memory
- Science timers to control instruments can be initiated by C and S. Backup control of some instruments is also provided
- Additional roll turn is provided in case antenna gimbal degradation or limitation occurs
- Stored sequences can be synchronized to a variety of physical events and can be enabled or inhibited in a group
- Individual commands can be enabled or inhibited

Figure 95. Preliminary Specification—Computing and Sequencing Subsystem

Three sequence counters provide for all mission phases. The range and granularity of each counter are determined by the requirements of the sequences using the clock. Associated with each counter is a mode register to identify the particular sequence to be timed out by the counter. Included in each mode register is an enable bit which can be used to inhibit or enable the sequence identified by the mode tag, or which can be used to synchronize the operation of the sequence to specific events, such as terminator or limb crossings.

Long term events, such as Canopus sensor update, are timed from launch, Some sequences of events, such as the planetary vehicle Mars orbit insertion sequence, are initiated at a time referenced from launch and involve the long term sequence as well as the fine-grained maneuver sequence. The initiate times are stored before launch, but can be updated to correct for trajectory dispersion. Other maneuver sequences such as orbit trim are initiated by radio command as the need arises. Orbit sequences are timed from Mars arrival, from the completion of a previous sequence, from a terminator or limb crossing signal, from the occurrence of a solar flare, or from a ground command. All sequences or events can be updated or inhibited by ground command as desired.

8.2 Requirements and Constraints

8.2.1 Mission Requirements

The C and S must provide timed outputs for control of flight spacecraft operations during the following mission phases: prelaunch, launch, automatic acquisition, interplanetary trajectory correction and orbit trim maneuvers, interplanetary cruise operations, orbital insertion maneuver, and flight spacecraft-capsule separation. It also must perform sequencing on a mission independent basis using sensor outputs for control and/or generation of sequences; provide control of planetary observation sequences except sequencing internal to the science subsystem; and generate antenna pointing commands.

Long term orbital events to ensure proper operation of the data acquisition and playback functions of the spacecraft and science subsystems include the following: inhibit the orbital sequence during at least the first orbit after orbit insertion to facilitate battery recharge, update the

Canopus sensor angles, command antenna positions and data rates, extend the orbital sequence if required, turn on gyros, and control planetary sensor platform modes of operation.

Short term C and S events during the orbital sequence include: start orbit sequences, change telemetry data mode, start science sequences, command PSP gimbal angles, change data mode for playback of stored data, and initiate the science instrument calibration sequences.

The commands to perform these operations are stored or received or modified in flight by commands from ground via the command subsystem.

Timed outputs which are dependent on mission, trajectory, or dispersions will be capable of being updated by RF command. In particular, commands to the capsule may require updating. Those outputs which are independent of mission, trajectory, or dispersion will be fixed relative to specific mission events, such as planetary vehicle separation.

Ground command capability will be provided to establish the interplanetary trajectory correction and orbit trim maneuver parameters, to alter or adjust flight spacecraft-flight capsule separation parameters, and to modify orbital sequence parameters if such adjustments are desired. Capability will be provided to verify and correct critical spacecraft maneuver commands prior to their execution.

The subsystem will provide capability for inhibiting the maneuver sequence in the event of an error in command data or failure in a maneuver event. In particular, the C and S will include the capability to inhibit engine burn if the maneuver attitude is incorrect. This will be accomplished through an enable bit in the timer word which can be set by ground command. If an enable signal is not received after spacecraft reorientation for trajectory or orbit correction, the engine fire signal will not be sent. The C and S will instead disarm pyrotechnics and command sun acquisition. At the discretion of the MOS, the inhibit function can be omitted by pre-enabling the maneuver commands.

8.2.2 Performance Requirements

The C and S provides redundant counters for integration of the output from a pulse rebalanced accelerometer, and provides time-referenced backup for thrust termination.

During subsystem, system, and prelaunch tests the C and S can be operated through the entire mission in less than four hours. The capability for inhibiting all events will be included. The sequencer is also capable of being automatically or manually controlled by its OSE through a real-time or speedup mission while all input and output interfaces are verified. Capability will be provided for individual initiation of all inputs and outputs in a non-sequential mode.

All modes can be commanded from three sources: onboard logic, ground command via RF link, or OSE via direct access.

The C and S operates independently of the command subsystem and does not use the command subsystem decoder. The C and S does not have direct control of the science instruments. These functions are normally handled by the DAE, based on initiate signals and over-all timing commands from the C and S.

The sequencer provides up to 246 timed outputs for operation of flight spacecraft and science subsystems and 10 outputs for the flight capsule. The C and S also provides outputs based on sensor data inputs originating from external events such as morning and evening terminator signal crossings, limb crossing, solar flare occurrences, booster-planetary vehicle separation, and Canopus acquisition.

Under normal operations the C and S does not store or decode science commands, but a backup capability can be provided. Initiation of individual science timers is performed by the C and S as a normal function. The C and S will be designed to operate continuously for seven months in transit and at least six months in Mars orbit.

Redundancy will be employed only when comparable design reliability cannot be achieved by simplification or alternate means for the intended function and when the reliability improvement offered by the redundancy can be shown to be real in spite of the increased complexity. Redundancy for most C and S functions can be provided by the command subsystem. The C and S includes redundancies at the subassembly level and a simplified back-up sequencer to provide for maneuvers and propulsion command functions.

8.2.3 Sequencing Requirements

The C and S provides for the following operations: interplanetary cruise, maneuver preparations, maneuver orientation, mid-course burn, deboost burn, trim burn, spacecraft-capsule separation, post burn or separation operations, roll maneuver for Canopus acquisition, science preparations, and photoimaging, record and playback. These sequences are summarized in Table 61 including the signals which initiate each sequence.

In general, sequences (or programs) will be initiated by timed commands. The commands load a timer with a word containing a mode tag to identify the sequence and a zero time for the counter. The counter will be inhibited from counting if the enable bit in the code is not set.

Sequences which depend on terminator crossing signals are initiated by timed commands and are enabled by the crossing signal to synchronize the execution of commands in the sequence with the terminators. Sequences which depend on the limb crossing signals are initiated by command in the current orbit science sequence prior to limb crossing and are enabled by the limb crossing signal. Sequences whose execution does not depend on synchronizing or enabling signals may be initiated and enabled simultaneously, i.e., the enable bit may be preset.

8.3 Functional Interfaces

8.3.1 Command Subsystem Interfaces

From the input decoder in the command subsystem the C and S receives all of the command time words or associated data that are to be stored in the central memory. The C and S also receives data from the command subsystem router to be stored in special registers in the function generator and accelerometer counters. This operation is used as a backup in the event of a failure in the memory system where the information is normally available.

The C and S also receives quantitative commands and time tags which directly load the mode registers and associated counters in the timing and mode select subassembly. For sequence initiation the numerical portion of two of the commands contains a 14 bit time for the counter, an enable bit, a 5 bit mode tag and the class and address of the designated mode register. The mission times are loaded with a 32 bit word, which

Table 61. List of Sequences

Sequence	Operation
Post-Separation Sequence (Maneuver Counter)	
1. Enable G and C electronics (immediately) and switch G and C to inertial hold mode (10 sec. after separation) - Reaction Control vats on high thrust.	Initiated by booster planetary vehicle separation.
2. Arm pyrotechnics (backup) 30 seconds after separation signal	Uses maneuver timer.
3. Deploy magnetometer boom, ionsphere antenna, and PSP	Self-terminating sequence
4. Uncage PSP gimbal	
5. Remove aperture covers and release medium and high gain antenna	
6. Deploy low gain antenna	
7. Switch transmitter to high power	
8. Enable G and CS sun acquisition (during initial eclipse)	
9. Transmit high and medium gain antenna angles.	
10. Enter TM data Mode II (234 bps)	
11. Turn on science power and initiate cruise science sequencing	
12. Turn on Canopus sensor and earth detector	
13. Begin 0.22 deg/sec roll-spin	Enabled by sun acquisition gate. Occurs only at first sun acquisition.
14. Begin 0.1 deg/sec roll-search	Ground command enabled.
15. Turn off earth detector	Self-terminating sequence.
16. Turn off gyros, reaction control jets to low thrust.	Enabled by Canopus acquisition
17. Neutralize maneuver timer	
Interplanetary Cruise Operations (Mission Counter)	
1. Turn on 50-watt transmitter	Initiated and enabled just before liftoff.
2. Switch to high gain antenna	
3. Generate antenna pointing commands (using function generator and stored commands)	First event occurs after appendage deployment.
4. Set TM to Mode II (Engineering and science data)	
5. Update Canopus (as required)	Operates off of the mission timer.
6. Initiate sequence to inject into Mars orbit	Sequence terminated only by ground command. Timer may be reset at Mars arrival if desired.
7. Initiate capsule separation sequence	
8. Update science sequencing (automatic mode)	
Maneuver Preparations (Maneuver Timer)	
1. Switch TM to maneuver preparations (Mode IV) mode	Use of orbit timer optional
2. Turn on gyros	Initiated by ground command
3. Test TVC system (on and off commands) gain on TVC high or low as appropriate	
4. Initiate capsule checkout sequence (pre-capsule maneuver only)	May be initiated by ground command.
5. Switch to medium gain antenna (stay on low gain for early midcourse)	Switching command to orientation phase must be ground enabled.
6. Point high gain antenna to maneuver orientation	
7. Initiate maneuver orientation sequence (if enabled by ground command)	
Maneuver Orientation (Maneuver Timer)	
1. Switch G and C to 1/4 deg limit cycle	(Use of orbit timer optional)
2. Switch TM to Mode I.	Initiated by ground command
3. Switch G and C to maneuver mode (go off celestial reference)	Enabled by ground command after verification.
4. Turn on maneuver tape recorder.	
5. Start roll turn	
6. Stop roll turn	
7. Start pitch turn	
8. Stop pitch turn	
9. Transfer to additional roll sequence (if this command enabled by ground command)	
10. Switch to high gain antenna	
11. Initiate engine operations sequence (if this command enabled by ground command) (midcourse burn, deboost, trim or capsule separation)	
Midcourse and Orbit Trim Burns	
1. Turn on TVC power	Initiated by ground command.
2. Turn on accelerometer	Enabled by ground command after verification.
3. Switch G and C to TVC mode.	
4. Arm pyrotechnic buses	Self-terminating sequence
5. Set pintle to low thrust position and select high TVC gain.	
6. Open start-tank solenoid valve (if enable not received, switch G and C to acquisition).	
7. Open normally closed explosive valves (or solenoids)	
8. Close starting valves (or solenoids)	
9. Close normally open explosive valves (accelerometer counter output)	
10. Close normally open explosive valves (timed backup)	
11. Pyrotechnics to "safe"	
12. Turn off TVC power	
13. Switch G and C to acquisition mode	
14. Neutralize maneuver timer	

Table 61. List of Sequences (Continued)

Sequence (Continued)	Operation (Continued)
Deboost Burn	
1. Turn on TVC power and select low gain	Initiated by stored command timer from mission timer for automatic mode (with manual update).
2. Turn on accelerometer.	
3. Switch G and C to TVC mode.	
4. Arm pyrotechnic busses	
5. Open helium isolation explosive valve	
6. Open start tank solenoid valves	
7. Open normally closed explosive valves	Enabled by ground command.
8. Open quad solenoid valves (engine start)	
9. Close start tank valves (6 seconds after No. 7)	
10. Open main tank large normally closed explosive valves	Uses maneuver timer with backup on orbit timer
11. Move pintle to full thrust level (and select low TVC gain)	
12. Close main tank quad valves (3 seconds after No. 9)	Self-terminating sequence
13. Close main tank large normally open explosive valves (from accelerometer counter and from central memory.)	
14. Turn on limb-terminator crossing sensor	
15. Reset busses	
16. Turn off TVC power and electronics	
17. Switch G and C to acquisition mode	
18. Neutralize maneuver timer	
Capsule Separation	
1. Enable capsule separation	Initiated by ground command or stored command off mission timer
2. Turn on capsule relay link receiver	
3. Switch TM to Mode I.	Enabled by ground command.
4. Start capsule sequences.	Uses maneuver timer, with backup on orbit timer.
5. Neutralize maneuver timer	Self-terminating sequence
Post Maneuver Sequence (Maneuver Timer)	
1. Switch G and C to low thrust	Initiated and enabled by Canopus acquired signal
2. Stop maneuver tape recorder	
3. Switch S-band radio to low gain (only for first midcourse maneuver)	Uses maneuver timer, with backup on orbit timer.
4. Enable antenna function generator	
5. Switch to TM Mode III (Readout stored data)	Self-terminating sequence
6. Switch to TM Mode II (cruise telemetry)	
7. Neutralize timer	
8. Trim off gyro	
9. Neutralize maneuver timer	
Orbit Science Preparations	
1. Turn on PSP	Initiated by photo sequence command.
2. Select PSP drive mode	
3. Set acquisition angles and begin tracking Mars	Enabled at evening (light-dark) terminator.
4. Turn on planetary instrument	
5. Calibrate planetary experiment	
6. Set color filter for photoimaging	Operates from orbit timer
7. Set shutter speeds	
8. Set IMC inputs to PSP (or photoimaging experiment.	
9. Switch G and C to $1/4^{\circ}$ limit cycle for PSP Mode I operation	
10. Load photo sequence code into orbit timer	
Photoimaging	
1. Actuate IMC	Initiated by orbit science preparation sequence
2. Actuate shutters	
3. Start tape recorder	
4. Read camera 1	Enabled at dawn limit crossing.
5. Read camera 2	
6. Stop tape recorder	
7-12 Repeat 1-6	Operates from orbit timer
13. Set G and C to $1/2^{\circ}$ limit cycle	
14. Load orbit science preparations code into orbit timer	
IR and UR Record, and Science Playback	
1. Start IR spectrometer recording	Initiated by stored or ground command.
2. Start UR spectrometer recording	Re-initiated by last command in the sequence
3. Start IR radiometer recording	
4. Stop IR spectrometer recording	Enabled at dawn terminator. Operates from maneuver timer.
5. Stop UV spectrometer recording	
6. Stop IR radiometer recording	
7. Set tape recorders for telemetry rate	
8. Start spectrometer playback	
9. Stop spectrometer playback	
10. Start radiometer playback	
11. Stop radiometer playback	
12. Start photoimaging playback	
13. Stop photoimaging playback	
14. Load record and playback code into maneuver timer	

contains a 26 bit numerical portion, an enable bit and a mode tag.

A special command routed through the command subsystem is available for use by ground control to enable stored commands in the C and S or to enable the mode registers. In this command the class and address portion identify the word in memory to be enabled. One of the available mode tags is used to identify this type of command.

8.3.2 Guidance and Control Interfaces

From the guidance and control subsystem a number of sensor signals will be received by the C and S subsystem. Three of these, the dawn and evening terminator and the dawn limb crossing signals, will be input to the timing and mode select subassembly to synchronize orbit science sequencing. The dawn terminator crossing signal will set an enable bit in the maneuver mode register and permit the counter to start its count from that point in the orbit. (For the non-science modes this bit will be preset with the mode select to allow the count to proceed from the time of command.) A Canopus-acquired signal will initiate the post maneuver sequence.

A serial pulse stream is received from the accelerometer and applied to the accelerometer counters at a rate determined by the vehicle acceleration. Each pulse corresponds to a velocity increment of 0.015 fps.

The C and S computes and issues antenna gimbal angle commands to the medium and high gain antennas. The commands are in the form of 12 bit serial bit streams loaded into registers in the G and CS from the central memory or from the function generator. Data for the medium gain antenna hinge angle and for the high gain antenna hinge and shaft angles are provided by timed command out of the central memory. Hinge angle data for the high gain antenna is provided by the function generator. A clear pulse to reset the command register precedes each command.

8.3.3 Science Subsystem Interfaces

Solar flare signals are received by the timing and mode select subassembly of the C and S to set the maneuver and alternate mode registers to special flare environment modes. These modes inhibit the normal science operations and permit the execution of special sequencing. This signal will select mode and time data from specified memory locations and load them into the mode registers. If orbit synchronizing is required, the enable bits are set by the dawn terminator or limb crossing signals.

Under normal operations the C and S issues mode commands to the DAE sequencer to initiate sequencing of individual science experiments. Each command starts an experiment timer in the DAE which may turn on power for that experiment, calibrate the instrument, start and stop data recording, turn off power and generally perform any functions related to the experiment. The C and S can organize these timer sequences into orbit sequences synchronized to terminator and limb crossing points and incorporating playback of recorded data. The C and S/science interfaces are illustrated in Figure 96.

The C and S provides a clock for the DAE which time tags the science data during recording. The "record" command in the experiment sequence automatically samples the lower order bits of the mission timer in the C and S and stores this as part of the tape heading for the data. The data start point can be resolved to a second. The time interval ambiguity can be removed by periodically transmitting the higher order bits via telemetry.

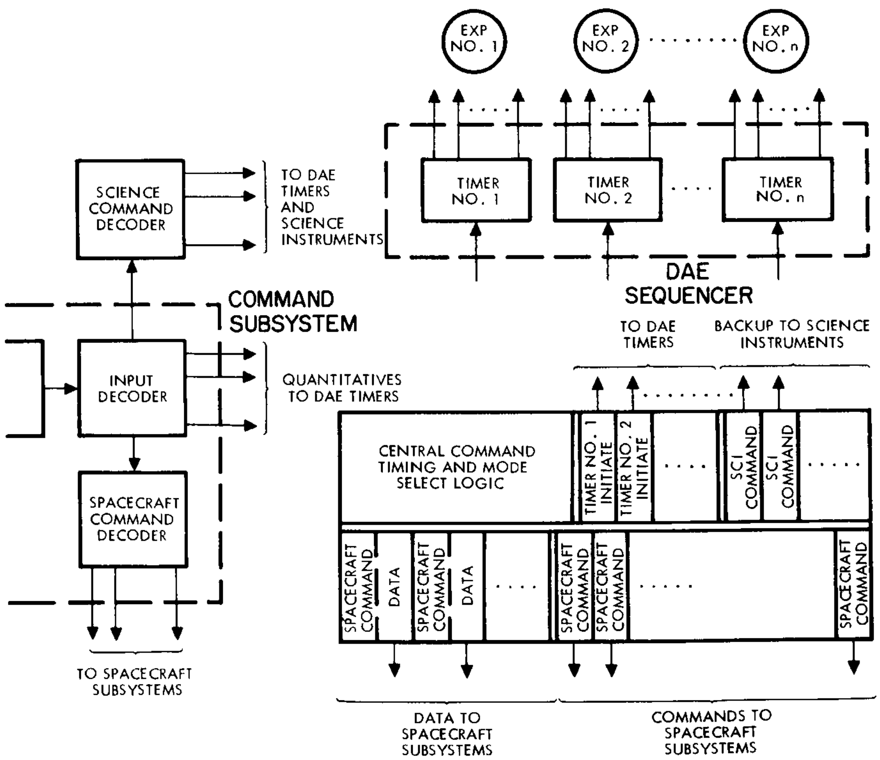


Figure 96. Illustrative DAE, C and S, and Command Subsystem Interfaces

In the normal operating mode no C and S outputs will directly control science instruments, but as a backup capability provision can be made to issue approximately thirty commands. These can include commands to point the planetary scan platform and to control the scan instruments in a simplified sequence. Some additional instrument control capability is also provided.

8.3.4 Launch Vehicle Interface

At the time of separation of the planetary vehicles from the Saturn, a signal is received by the timing and mode select subassembly to set a mode in the register to begin post-separation sequencing. This signal will load a sequence code, but shall only operate on one register.

8.3.5 Power Subsystem Interface

Electrical power at 50 volt 4 kc is required by the C and S subsystem for conversion to appropriate voltage levels by an internal power supply. The C and S also receives a 540 kc signal and a 400-cycle signal for timing and control of internal sequences. Further division to 1 and 0.25 bps pulses will be performed by the C and S.

8.3.6 Telemetry and Data Storage Subsystem Interfaces

Outputs to telemetry include data mode, data rate, and recorder playback commands and C and S engineering data. The commands are summarized in Table .

For telemetry encoding, C and S memory words are issued one at a time to a 21 bit register sampled by telemetry at a rate determined by the data mode. When the sampling is completed the next word in memory is loaded into the register. Both a low and a high rate telemetry data mode are provided. The high rate is commanded as a premaneuver mode to permit verification of maneuver data.

Gates are included in the telemetry and C and S subsystems to enable asynchronous operation of the two subsystems. Every three 7-bit samples, telemetry issues a pulse which enables a C and S gate to permit loading the register. During the loading operation telemetry sampling is inhibited. (In consequence, a memory word may be delayed for one full sampling cycle.) To maintain ground sync, one of the memory words is identified by a unique pattern and is used to locate the data words in memory. All data words begin with a "1" bit to differentiate between actual zero words and no-sample words.

A count of the number of C and S command outputs is provided in a 7 bit register to be sampled by telemetry. This register is reset to zero by the telemetry subsystem after the sample.

8.3.7 Propulsion Subsystem Interface

Outputs to the propulsion subsystem consists of command discretes to open normally closed valves or to close normally open valves. Double commands are issued simultaneously to open solenoid pairs in the fuel and oxidizer systems for engine burn. Commands relating to actuation of squib valves are applied to the pyrotechnics subsystem (see Table 62).

8.3.8 Capsule Interface

Provision has been made in the C and S to issue 10 command discretes to the capsule.

8.3.9 Pyrotechnics Subsystem

Twenty-four commands are issued from the C and S to pyrotechnic devices. These commands include safe/arm, capsule separation, capsule umbilical disconnect, and PSP gimbal uncaging, as well as those associated with G and CS and propulsion. A summary of the interface signals is given in Table 62.

8.4 Design Description

8.4.1 Organization and Operation

A functional block diagram of the primary sequencer is shown in Figure 97. The command input unit accepts messages from the command subsystem, processes the data, and loads it into memory. The data is issued to the input unit in the form of a pulse train having 20 information bits to be stored, 2 data class bits, 8 address bits to identify the storage location, and 2 parity bits. The messages to be stored include command times and mode designation, clock times and mode code, and numerical data. A special format is used to identify the "command enable" word which sets the enable bit in a memory word. The formats for these messages are shown in Figure 98.

Table 62. Summary of Input and Output Requirements

Subsystems	Function	No. of Commands	Remarks
COMMAND OUTPUTS			
Guidance and Control	Pitch start and stop time	2	Start and stop commands required one time per maneuver
	Roll start and stop times	4	Start and stop commands required up to two times per maneuver
	Turn polarity	2	Polarity selected two times
	Canopus sensor cone angle update	6	6 different angles
	TVC test signal	2	On and off
	TVC power	2	On and off (Switch by power distributor)
	TVC electronics on/off	2	On and off commands required for trajectory corrections
	TVC gain high/low	2	For propulsion low andhigh thrust
	Gyros on/off	2	On and off commands required for maneuvers
	Dead-zone select	2	On and off commands required for reorientation maneuvers and PSP imaging
	Acquisition mode	1	Initiate automatic search process
	GCS on	1	Backup to separation switch
	Canopus sensor on	1	Starts roll search mode
	Gyros rate/position	2	Function of modes
	Sun sensor inhibit	2	Initial attitude hold mode
	Reaction control thrusters, high/low	2	Function of mode
	High gain antenna hinge position	2	
	High gain antenna shaft position		
	Medium gain antenna hinge position		
	Earth detector on/off	2	
	Accelerometer on/off	2	Used during propulsion operation
	Roll rate- magnetometer calibration	2	Magnetometer calibration canopus search
	Limb-terminator crossing sensor on/off	2	
	Release high and medium antenna	2	
	Deploy low gain	1	
C and S	Maneuver sequence select	12	Maneuvers divided into four sub-maneuvers in C and S
	Science sequence select	6	Provides alternate sequencing
	Load accelerometer counter	2	One for each counter
	Load function generator	1	
Propulsion Subsystem	Selection of solenoid valves explosive valves, injection pintle, and gas pressurization actuators	18	Operate pyro devices for arming, starting and stopping midcourse propulsion corrections, arming, starting, and stopping orbit insertion propulsion, and arming and firing orbit trim propulsion enable.
S-band Radio Subsystem	Transmitter/antenna	3	Two onboard backup commands for selection
	Receiver mode select	2	Includes special mode select in event of loss attitude control
Capsule Radio Link	Capsule data	2	
	Receiver on/off	4	Two receivers
	Write	2	Recoder in CRL.
	Playback	2	
Telemetry and Data Storage	Data modes	4	
	Store maneuver data	1	
	Data rates	5	
	Tape recorder select	7	
	Tape recorder playback	14	Includes start/stop on playback. Data record function controlled by DAE.

Table 62. Summary if Input and Output Requirements (Continued)

Subsystems	Function	No. of Commands	Remarks
COMMAND OUTPUTS (Continued)			
Telemetry and Data Storage (Cont.)	Tape advance select	12	Includes start/stop in tape advance mode
Capsule	Mode select	10	
Science Subsystem	DAE sequencing control	12	One for each instrument to initiate timer
	Science backup in C and S	30	Includes PSP mode control, gimbal angles, and shutter actuation, plus other science sequencing relating to limb or terminator crossing and solar flare signals
	Deploy Ionosphere antenna	1	
	Deploy Magnetometer Boom	1	
Pyrotechnics Subsystem	Arm (main)	1	Timed from separation
	Arm (individual)	8	
	Deploy PSP gimbal arm	1	
	PSP gimbal uncaging	1	
	Cap emergency separation	1	
	Capsule umbilical disconnect	1	
	Reset	1	
	Jettison science subsystem cover	1	

INPUTS

Type	Source
Dawn terminator crossing signal	G and CS
Evening terminator crossing signal	G and CS
Limb crossing signal	G and CS
Solar flare signal	Science
Booster/planetary vehicle separation signal	Adapter
Fine sun-sensor signal	G and CS
Outputs of quantitative commands	Command Subsystem
Canopus acquired signal	G and CS
Accelerometer pulses	

FREQUENCIES

Frequency	Source	Remarks
540 kc	Power	Internal C and S clock
400 cps	Power	Used to provide 1 pps + 1/4 pps
4 kc	Power	Main power input frequency
1 pps	Command Subsystem	Data input rate

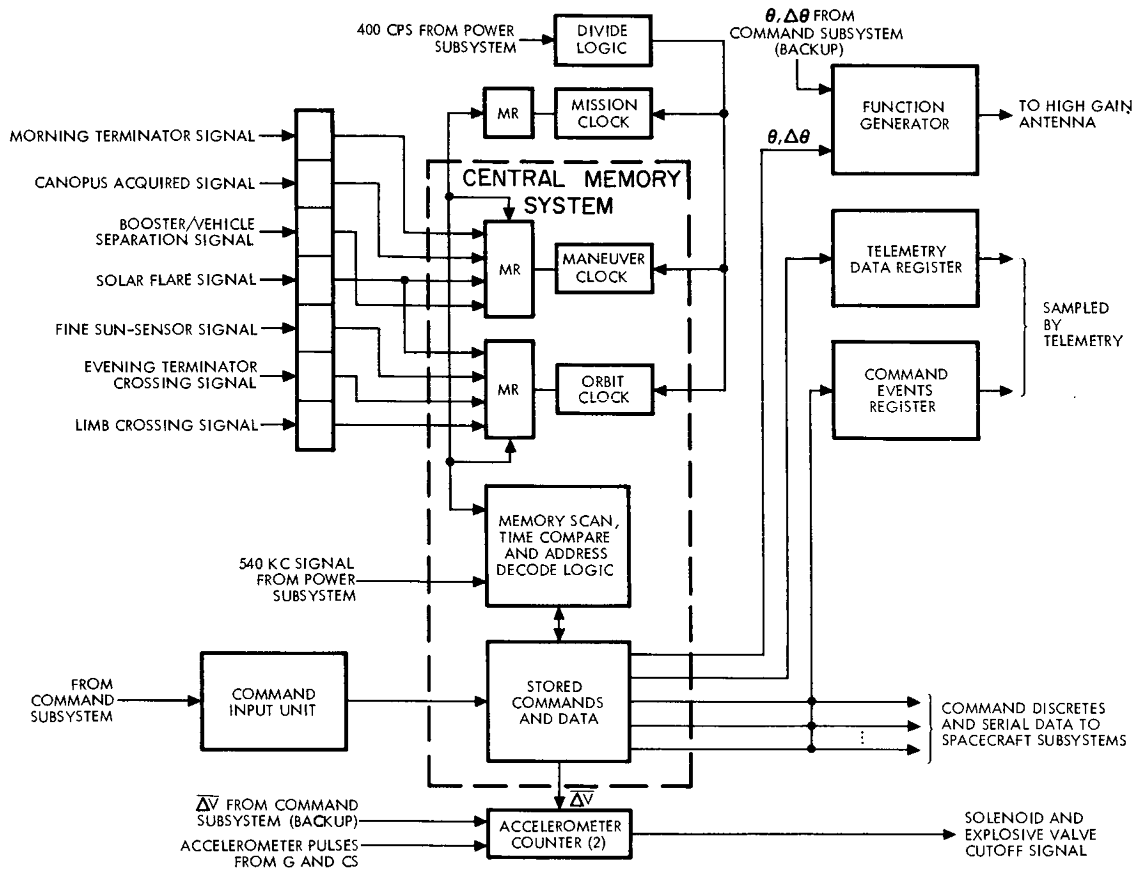
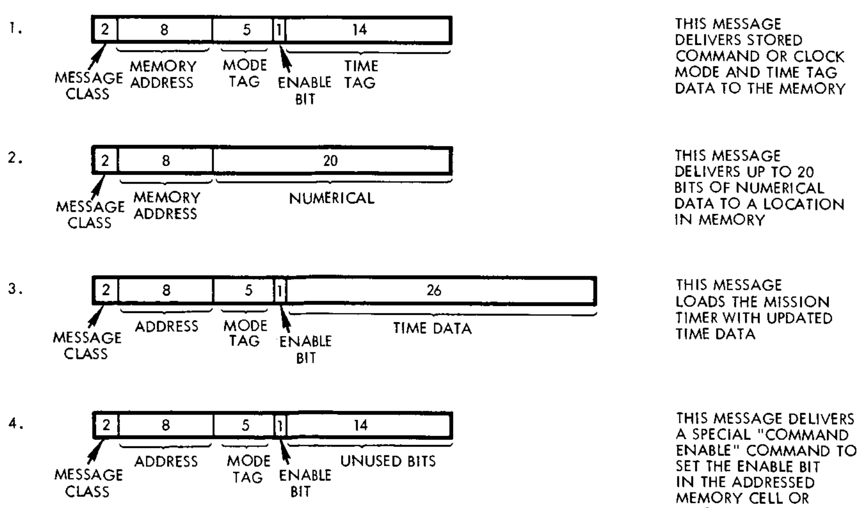


Figure 97. Primary C and S Functional Diagram



NOTE: TWO PARITY BITS WHICH ACCOMPANY THE MESSAGE AND ARE STRIPPED BY THE INPUT UNIT ARE NOT SHOWN.

Figure 98. Message Formats

The C and S has three counters, one associated with each mode register. Memory scan and time-compare logic cycles through the memory locations once every second to compare stored words with timer words. If both time and mode coincidence occurs and if the command word enable bit is set, then, depending on the function of that memory location, either a command discrete or a serial bit stream is issued to a particular subsystem.

A sequence can be synchronized to a particular event by setting the enable bit at the time of the event. This is accomplished by a signal from the terminator, limb crossing or fine sun-sensors, by a separation signal, by a Canopus acquisition signal, or by a ground command. Setting the enable bit releases a counter inhibit and permits the counter to increment.

Three counters can be used to sequence commands concurrently. The mission counter has a granularity of one second and a range of 24 months and is used for sequencing long term events, beginning other sequences, and time tagging science data. Memory words are compared with only the 14 most significant bits of this counter, thereby providing a 1.08-hour granularity. The second counter has a four-second granularity and 17-hour range and is used for all orbit science sequences which require the terminator crossing signal. The third counter, with one-second granularity and a 4.5-hour range, is used for both maneuver and science sequences. Special sequences initiated by a solar flare signal use either the second or the third counter. The mode register and counter pairs (timers) can be loaded either by command data in memory or by quantitative commands.

The times or order of occurrence of commands within any sequence can be modified or updated at any time by ground commands. The C and S can repeat an orbit sequence as desired in a periodic fashion measured from the morning terminator. Each orbit sequence (data acquisition and playback) normally lasts for one orbit but may last for more than one. The C and S can perform a series of orbit sequences initiated by the long term orbital time reference such that the initiation of each sequence in the string coincides with terminator crossing. This permits the sequencing of events in one orbit sequence to prepare for events to occur in the next orbit sequence.

The mission timer is used to sequence long term events both from the central memory and from the function generator. The function generator is used to generate antenna pointing commands. No mode designation is required for the function generator and the data output time is established by sampling the mission counter. Computations are performed from starting data obtained either from the central memory or from ground command.

Engine cutoff commands are issued from the central memory by mode control and time compare, but this is a backup mode to the accelerometer counter operation. Normally, the accelerometer counters accumulate accelerometer pulses and issue a cutoff signal when the count integrates to the required ΔV correction. The cutoff point is established by preloading the counter with the ΔV complement and letting the overflow pulse cut off the engine. The overflow pulse is sent to the correct cutoff valve by first enabling the appropriate one of five gates, one for each of five sets of valves.

Finally, two telemetry registers complete the C and S subassemblies. One register is used to telemeter stored data. Words are loaded into the telemetry register and are sampled by telemetry sequentially. Gating logic prevents interference of the loading and sampling operations. The other register is used to count the number of command outputs since the last interrogation and readout by telemetry.

8.4.2 Timing and Mode Select Unit

a. Mission Timer

The mission counter is driven by a one second clock which is derived from a 400-cps source from the power subsystem (Figure 99) A modulo 40 and a modulo 10 counter are used to divide down the 400 cycle input. The contents of the mission counter can be altered or interrogated under the following conditions:

- The mission clock is filled directly from the command subsystem.
- It is filled from the C and S memory.
- It is read out to the telemetry register of the C and S.

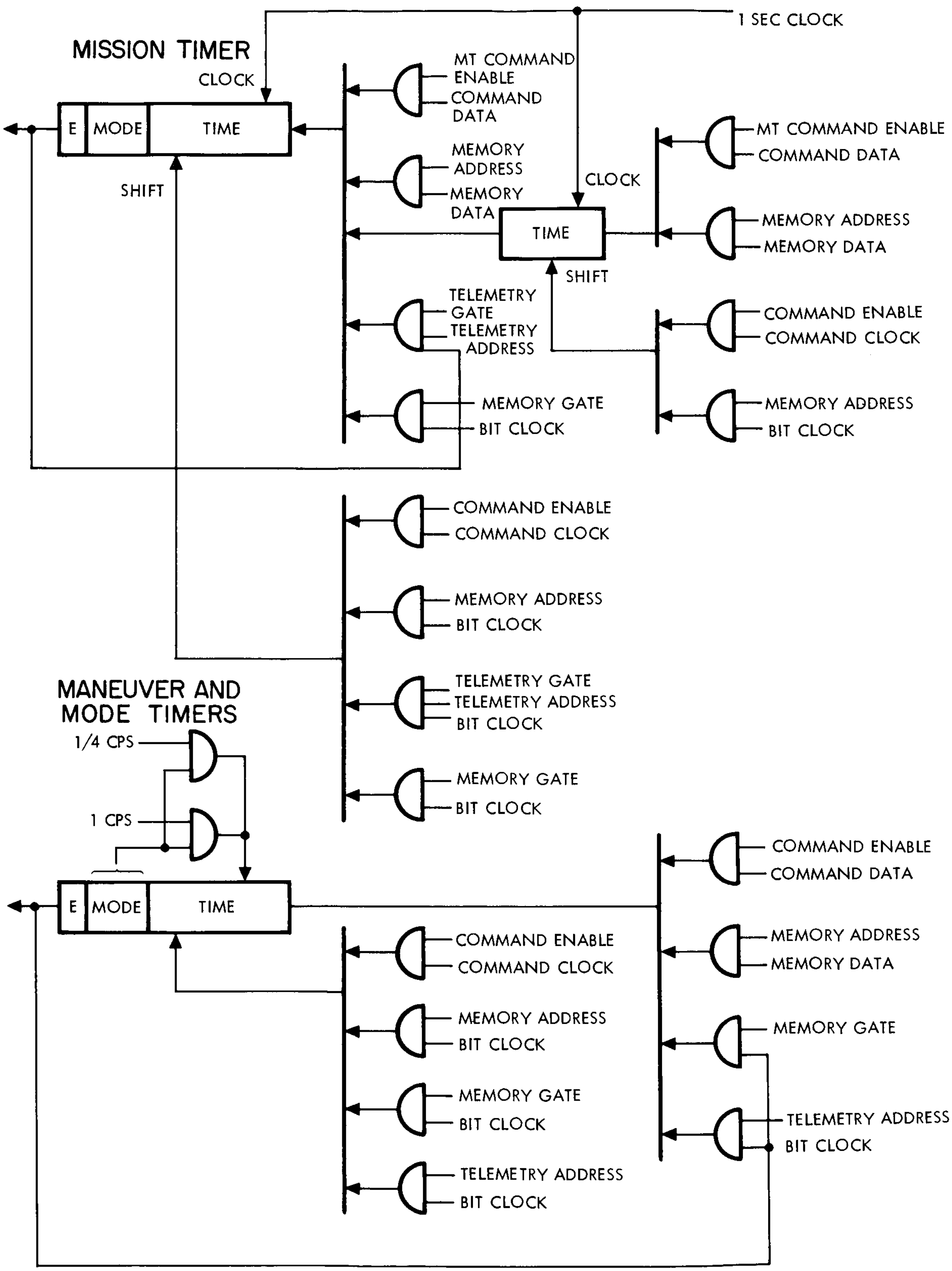


Figure 99. Timing and Mode Select Logic

- It is compared with the mode/time word currently being read from memory.

b. Maneuver Timer

The maneuver counter(which is also used for orbital operations) is driven by a one-second clock. However when used for some orbital operations it is driven at a 4 second rate. The count rate for the maneuver timer is determined by its associated mode register contents. Its contents can be altered or interrogated under the following conditions:

- At time of solar flare, the enable bit is reset, the mode is set to the solar flare code, and the time count is reset.
- At dawn terminator crossing, the enable bit is set, letting the one-second clock drive the counter.
- The timer is filled directly from the command subsystem;
- The timer is filled from the C and S memory.
- The timer is read out to the telemetry register of the C and S.
- The timer is compared with the mode time word currently being read from memory.

c. Orbital Timer

The orbital counter is driven only by a 4-second clock derived from the one-second basic clock by a divide by 4 counter. Its contents can be altered or interrogated under the following conditions:

- At time of solar flare, the enable bit is reset, the mode is set to the solar flare code, and the time count is reset.
- At limb crossing, the enable bit is set, letting the 4-second clock drive the time counter.
- The timer is filled directly from the command subsystem.
- It is filled from the C and S memory.
- It is read out to the telemetry register of the C and S.
- It is compared with the mode/time word currently being read from memory.

8.4.3 Command Input Unit

The function of the command input unit (Figure 100) is to accept a command word from the command subsystem, store it until the command subsystem has verified the parity, strip the parity bits from the word, and then transmit this information to the memory unit of the C and S subsystem.

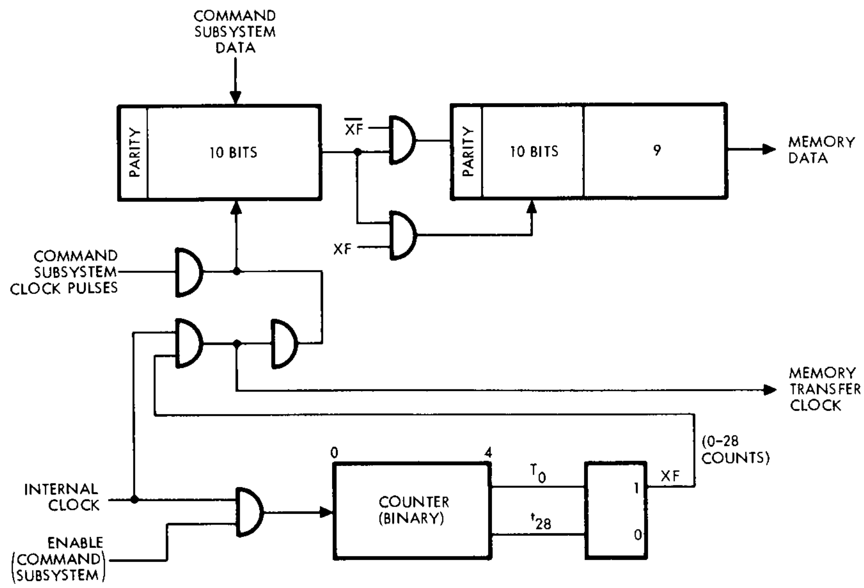


Figure 100. Command Input Unit

The command input unit accepts the input data and clock pulses from the command subsystem. Using the 1-bps clock pulses, the bits are shifted into a register which is capable of holding the entire message, including the middle and end parity bits. Following the storage of the entire data word, the command subsystem issues an enable signal signifying a valid parity check. This enable signal gates the C and S internal clock into a four bit binary counter. The counter is decoded with two counts, one equivalent to the beginning of the message and the second equivalent to the end of the message. During the interval the contents of the register are transferred to the memory unit. The transfer is accomplished with the internal high speed clock. In order to strip the parity bits from the message a path is wired around the middle parity bit. This path is activated only during the transfer time from the command input unit to the memory unit. The end parity bit is ignored during the transfer to memory by limiting the bit count to one less than the message length.

8.4.4 Memory System

The fundamental task of the C and S memory is to output mode/time data to the mode/time comparators and to output the command pulse or associated data to the proper subsystem when a time coincidence occurs.

The system consists of a 256 word by 20 bit core memory, an 8 stage word address counter/decoder and drive/switch circuits for word selection, a 5 stage bit counter/decoder and drive/switch circuits for bit selection, two mode/time counters (i. e., the maneuver and orbital timers), mode/time comparator circuits, output select and drive matrix, and timing control logic. The organization of these subassemblies is shown in Figure 101.

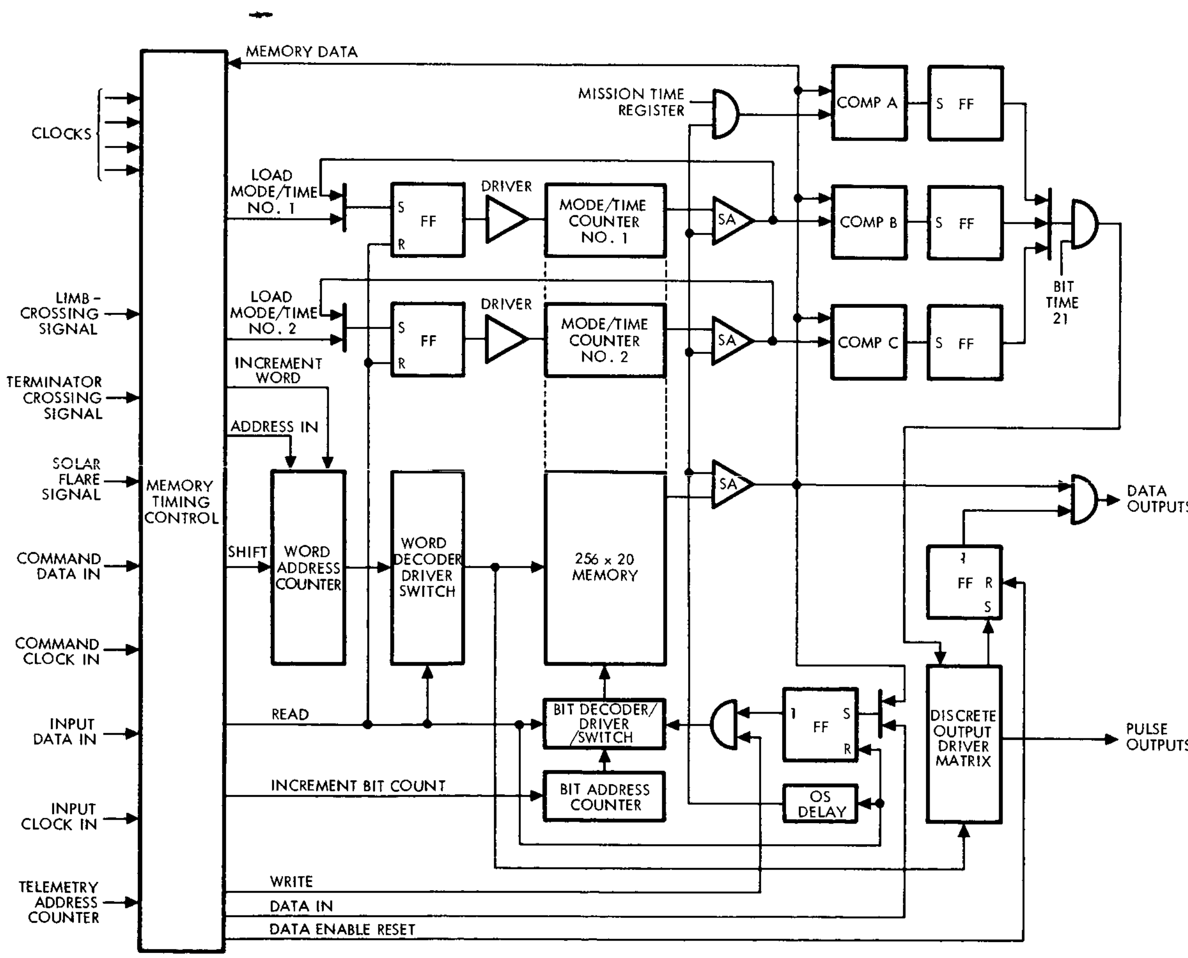


Figure 101. Memory Subsystem Block Diagram

Each of the command memory words is read serially from core memory into the mode/time comparator circuits to be compared with the contents of the maneuver and orbital timers and with the 20 most significant bits of the mission clock. The scan logic cycles through memory once every second. When a coincidence occurs between a memory word and one of the three registers, the output corresponding to that memory word is activated by the output driver matrix. This matrix sets an enable flip-flop if serial data is to be issued.

The maneuver and orbital timer registers are circulating registers contained within the core array, but they have independent electronics except for the bit selection windings. The logical conditions on these counters are discussed in Paragraph 8.4.2.

The memory timing control logic sequences the memory subsystem. Timing clock pulses from the frequency divide logic (Paragraph 4.2) are used to form the required memory clocks. Limb crossing, terminator crossing, solar flare, booster/planetary vehicle separation, Canopus acquisition, and fine sun-sensor signals are used to modify the contents of the mode/time counters. Data from the command subsystem and the command input subassembly of the C and S are accepted and entered into the appropriate mode/time registers or memory cell.

8.4.5 Telemetry Data Register

The telemetry register of the C and S (Figure 102) is used to sequentially store memory words, mode/time register data, function generator data, and ΔV data. This data is input to the register if the following conditions exist: the memory cycle is in the "TM gating" state and telemetry has shifted out all 21 bits of the previous word stored in the register. An interlock is provided to prevent the loading of the register until telemetry has completed its interrogation, and to prevent an interrogation during the loading phase.

8.4.6 Command Events Register

The function of the command events register is to count the number of stored commands issued to the various spacecraft subsystems. A comparison can then be made between the number of commands issued and the number executed.

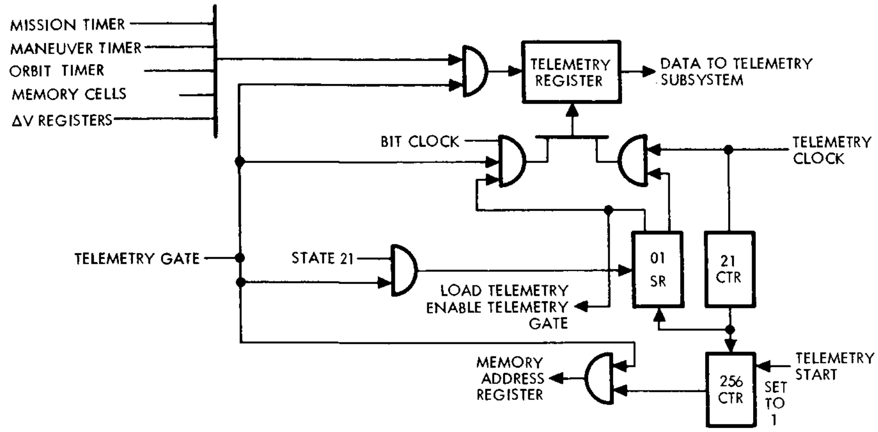


Figure 102. Telemetry Data Register

The command events register is a combination binary counter and shift register. As a counter the device is pulsed by any of the stored command output lines from the memory unit. All output lines are OR'ed together to form one pulse input for the counter. The counter is capable of counting up to 128 command discretes between telemetry samplings. The OR combination of input signals is accomplished through gates which prevent any feedback from one command line to the next.

When telemetry requests a command count, the command events register operates as a shift register, and the contents are shifted out serially. The clock pulses for the shifting process are derived from a central clock and gated to the register by the C and S enable. The clock pulses are also made available to Telemetry for gating the information. The signal from telemetry requesting the register count is an enable which is active for the duration of the data shift. Figure 103 illustrates the logical operation of this device.

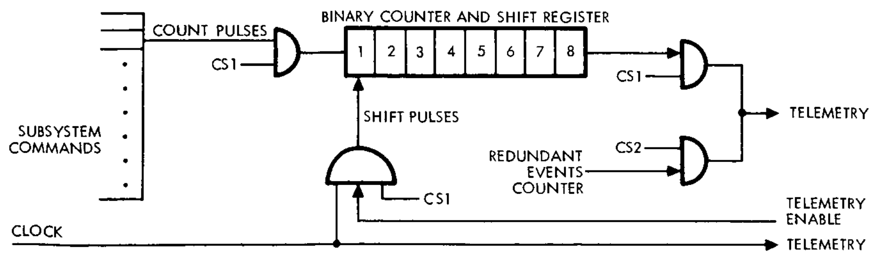
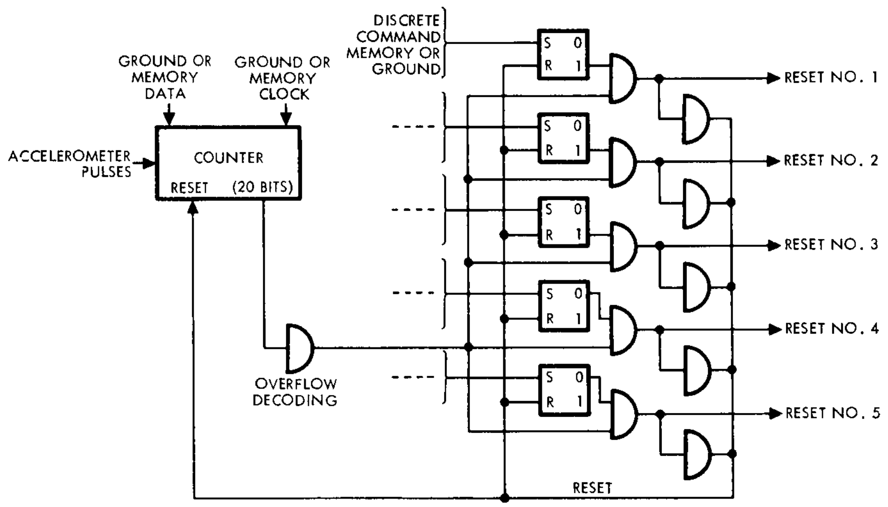


Figure 103. Command Events Register

8.4.7 Accelerometer Pulse Counter

The velocity correction requirement for the vehicle ranges from 3.28 to 6250 ft/sec with a defined maximum acceleration of 32 ft/sec^2. The quantization is .015 ft/sec-pulse so the pulse count can range from 219 to 417,000. The maximum pulse rate is 2,150 pulses/sec. For the given granularity and maximum ΔV requirement, a 19-bit counter is adequate to accumulate the velocity increment without changing scale factor in the accelerometer. Since a 20-bit data word is permitted in memory, this size is specified for the ΔV accumulator register. (See Figure 104).



NOTE: AN IDENTICAL UNIT IS LOGICALLY OR'ed WITH THIS ONE TO FORM A REDUNDANT SYSTEM.

Figure 104. Accelerometer Counter System

The accelerometer counter controls the length of engine burn time by issuing a signal to an appropriate value after a predetermined number of accelerometer pulses have been counted. The correct count needed for the signal (or reset pulse) is obtained by setting the counter initially to the complement of the number of required pulses determined by the ΔV requirement. When an overflow occurs the reset pulse is issued.

Two commands are necessary to activate the counter. One command is required to load the counter initially with a predetermined number.

Another is needed to set an enable which selects one of five reset gates. The enable pulse is stored in a flip flop for the duration of the counting process. Following the counter overflow decoding, the reset pulse is issued via the enabled gate and this pulse also resets the enable flip flop and the entire counter.

During input of quantitative data from either the memory or ground command, the counter acts as a shift register being shifted by either the memory transfer clock or the ground command transfer clock. During periods of acceleration the counter acts as a binary counter from which an overflow bit can be sensed. Two separate counters and gate systems are used, and each is separated addressable from memory and from ground command.

8.4.8 Function Generator

The function generator uses approximating linear functions to point the high gain antenna on a continuous basis over the full mission. The unit accepts angular start data from either stored information or outputs of quantitative commands and issues command data to the hinge angle gimbal register of the antenna.

The pointing angle function, given in spacecraft body coordinates, is approximated by n linear segments, $\theta_i(t)$, where $n < 10$.

Let $\theta_i(t_i^o) = \theta_i^o$ be the starting angle for the i^{th} segment and let $\Delta\theta_i$ be the fixed angle change in the i^{th} segment for the fixed increment of time Δt. Assume the start times, t_i^o, for the segments are given in terms T_i of the mission clock.

Cruise Mode	Time T_i to Initiate i^{th} Segment	
		"start computation" command
$\Delta\theta_i$	θ_i^o	start data

At the initiate time T_1 for the first linear segment, shift θ_1^o and $\Delta\theta_1$ into the function generator register, using the computation "start" command, and issue the angle θ, to the high gain antenna cone gimbal angle. At $T_1 + \Delta t$ form $\theta_1^o + \Delta\theta_1$, store this quantity in the θ_1^o register and issue the quantity to the gimbal angle. Repeat for $\theta_1^o + 2\,\Delta\theta_1$, etc. At T_2, shift θ_2 and $\Delta\theta_2$ into the registers and continue the process.

The length for the time tag, T_i, is 14 bits. This time is referenced to the high order 14 bits of the mission clock. The time increment Δt is obtained from a low order bit of the high order 14 bits of the mission clock. This is sampled by the function generatory on a continuous basis to establish the times to issue gimbal angle data. The bit length for θ_i is 14 and the bit length for $\Delta\theta$ is 6.

Figure 105 shows a block diagram of the high-gain antenna hinge axis angle function generator. Two main registers, "angle" and "increment" contain the present pointing angle θ and the current $\Delta\theta$ increment. At each computation time obtained from the mission clock the full adder adds $\Delta\theta$ to θ, sends a control signal to the antenna and issues a 14 bit stream of pulses to the hinge axis gimbal. This process continues throughout the mission and occurs on a regular basis. It is interrupted only when new start data is available. On a control signal from the memory (or from ground) indicating the presence of the new data, a 20-bit serial stream loads the two registers, and the additions continue. The memory transfer time is used to reset the carry and carry-delay flip-flops. Memory transfers, angle transfers, and additions are performed under an internal two-phase clock shown in Figure 105 as C1 and C2. An inhibit signal generated by the mode register decoding prevents any output to the antenna during maneuver conditions.

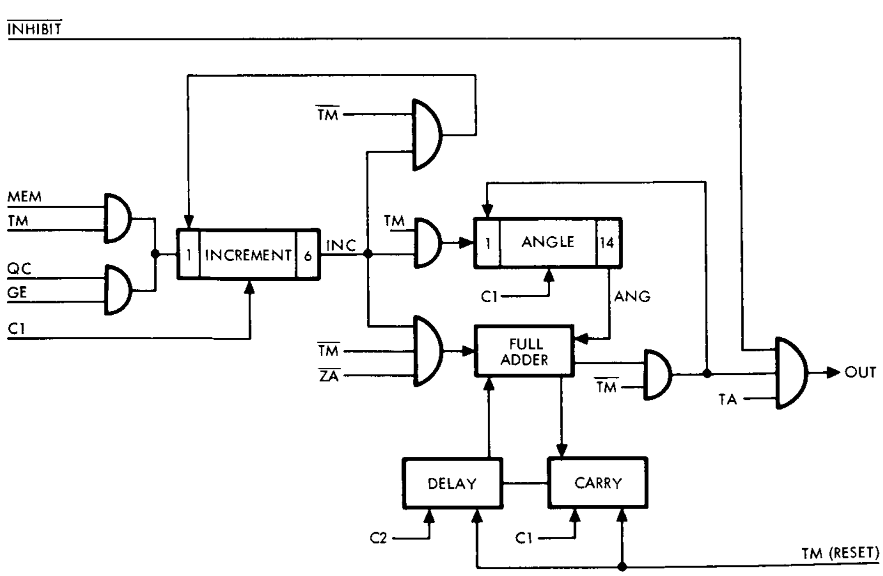


Figure 105. Function Generator Block Diagram

8.5 Reliability Considerations

8.5.1 Hardware Selection

The basic approach in the C and S implementation has been to incorporate as many functions as practicable into a centralized core memory system consisting of an array of 30-mil, wide temperature, ferrite cores and an x-y selection technique which is driven by external logic to access (DRO) one bit at a time. Core registers with compatible word size and accessing requirements (timers) have also been included. Other registers (telemetry) have been implemented as independent core shift registers similar to those of the Mariner C CC and S. For purpose of study and planning, the remainder of the Voyager C and S is considered to be constructed of DTL type integrated circuits.

These components and techniques have been employed by TRW in developing a general purpose programmer for near future Air Force requirements. Although this programmer differs in several particulars from the Voyager C and S it accomplishes the same basic functions. The components and techniques have also been applied in the design of the LEM abort guidance computer, for which a complete set of high-reliabilty circuit approaches has been developed.

8.5.2 Subassembly Redundancy

Cooperative multichannel redundancy is used in the timing and mode select unit to control the C and S sequencing. This is accomplished by permitting command maneuver sequences to be controlled from two registers. The maneuver and orbital mode registers and counters are needed to control concurrent orbital operations synchronized separately to terminator and limb crossing points, but they are also capable of providing some backup capability for each other. If one of them fails, the other can be loaded from memory or from ground to initiate sequences and to time events. However, to load the register from memory an additional storage location must be provided since each location is identified with a specific function.

Functional redundancy is also provided because of the ground backup loading capability. Each timer is separately addressed from the command subsystem and can be loaded by quantitative command. Individual sequences can thus be initiated from the ground and controlled by the backup timer. In the same way ground command backup is provided for sequence synchronization.

Similar redundancy is provided for the ΔV counters and the function generator. Normally these units are fed from the central memory, which contains command quantities or start data. However, as a backup they can also be loaded from ground. Each unit contains an input register which accepts the data and permits execution of the function, so a quantitative command can be used to control the operation in the event that the central memory fails.

If either of the ΔV counters fails, the engine can still be cut off using the alternate counter or the backup time count by command from memory. Stored start and stop commands are provided for each of the engine operating modes. The stop commands are issued over output lines to the same valves controlled from the ΔV counter and can activate those valves if the counters fail.

Should the sensor signal input unit fail to operate, sequencing start times can be controlled by properly timed quantitative commands. Orbit science sequencing can thus be synchronized to the Mars lighting conditions even if the sensors themselves fail to operate.

8.5.3 Subsystem-Level Redundancy

In addition to cooperative multichannel or functional redundancy, direct commands via the command subsystem provide backup or alternate means of accomplishing many C and S functions. It is not appropriate, however, to use direct command backup for all functions. Because of the danger in using direct commands to control the attitude maneuvers and propulsion start-stop, an alternate means for on-board control needs to be provided. TRW has briefly considered two methods of providing backup capability for these functions. One method involves the use of a scaled-down version of the primary sequencer, omitting components that have other provisions. The other involves the use of a special-purpose sequencer that can provide only the desired back-up in a limited capability sense. Early in Phase IB these alternates will be evaluated and one chosen. A brief description of the components involved in both of these alternatives follows.

a. Scaled-down Primary Sequencer

A block diagram of a scaled-down version of the primary sequencer is shown in Figure 106. Command capability from the central memory of this assembly duplicates the maneuver operations performed by the primary system: maneuver preparations, orientation, engine operations, and postmaneuver operations.

Antenna pointing commands are also available in a reduced capability. These commands are stored in memory and can be updated by ground command as desired. These commands point both the medium and high gain antenna hinge angles, and the high gain antenna shaft angle.

Only two clocks are provided, a long-term mission clock which has only 14 bits of time data compared with 26 in the primary system and another the same as the maneuver timer of the primary system.

A sequence tag in the mode register of the maneuver timer identifies the sequence to be executed. An enable bit in the code permits synchronization with the dawn terminator. This allows some automatic orbit science sequencing.

b. Special Function Back-up Sequencer

Typical logic for a backup C and S is given in Figure 107. This unit is a reduced and highly specialized backup system having individual timers, with associated storage registers, for each of the maneuver preparations, maneuver orientation, engine operations, spacecraft separation, and post maneuver sequences.

Each sequence is enabled separately after ground verification of stored data, and commands or data are issued sequentially on a Δt countdown for each item. After each command in the sequence the timer is loaded with the time delay to the next command, and the appropriate output gate is enabled to issue a signal at the underflow pulse of the countdown.

Another counter keeps track of the successive Δt registers and associated output gates to shift the correct Δt into the timer and issue the correct output signal. A low frequency asynchronous multivibrator is used for the basic frequency sources, the sequence is initiated by a discrete from the command subsystem. The storage registers are loaded

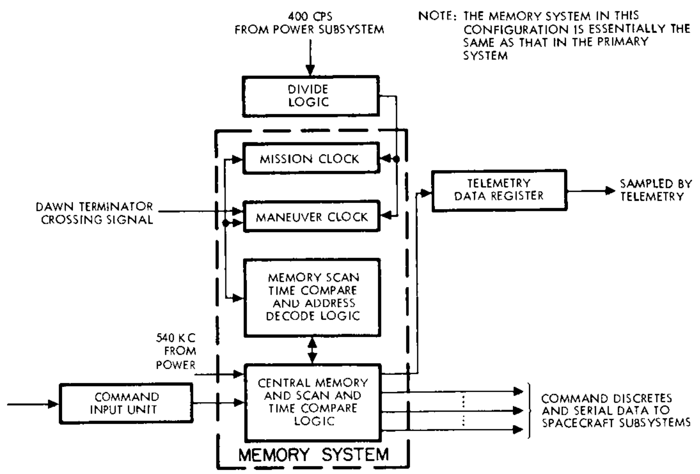


Figure 106. Backup C and S Functional Diagram

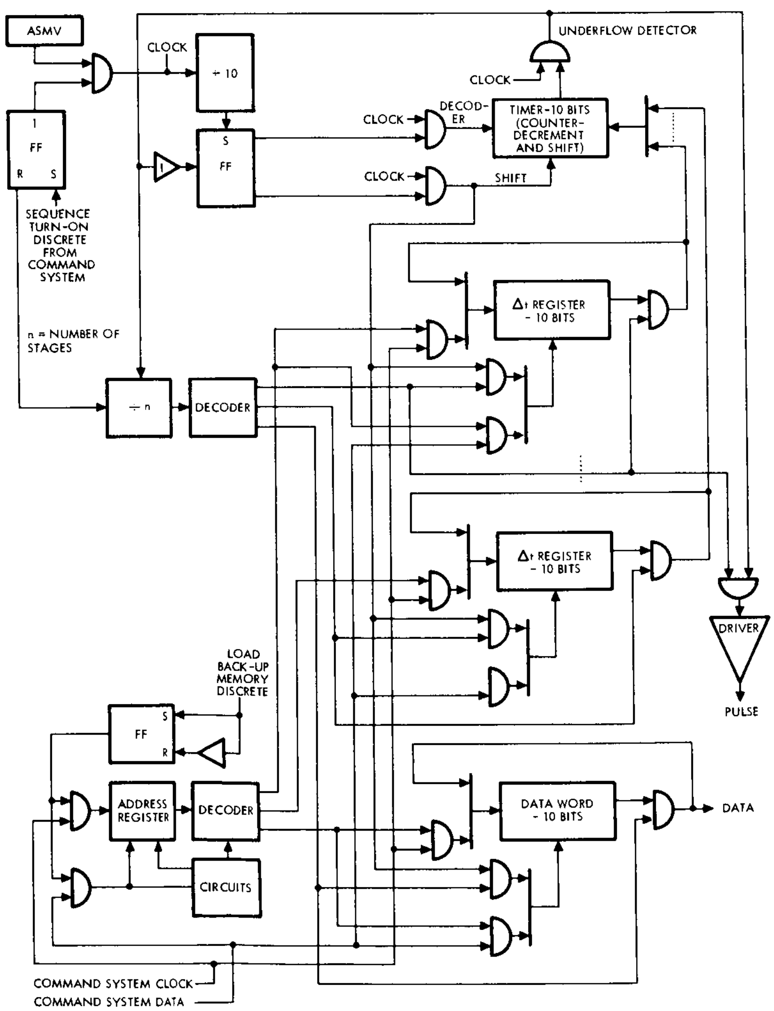


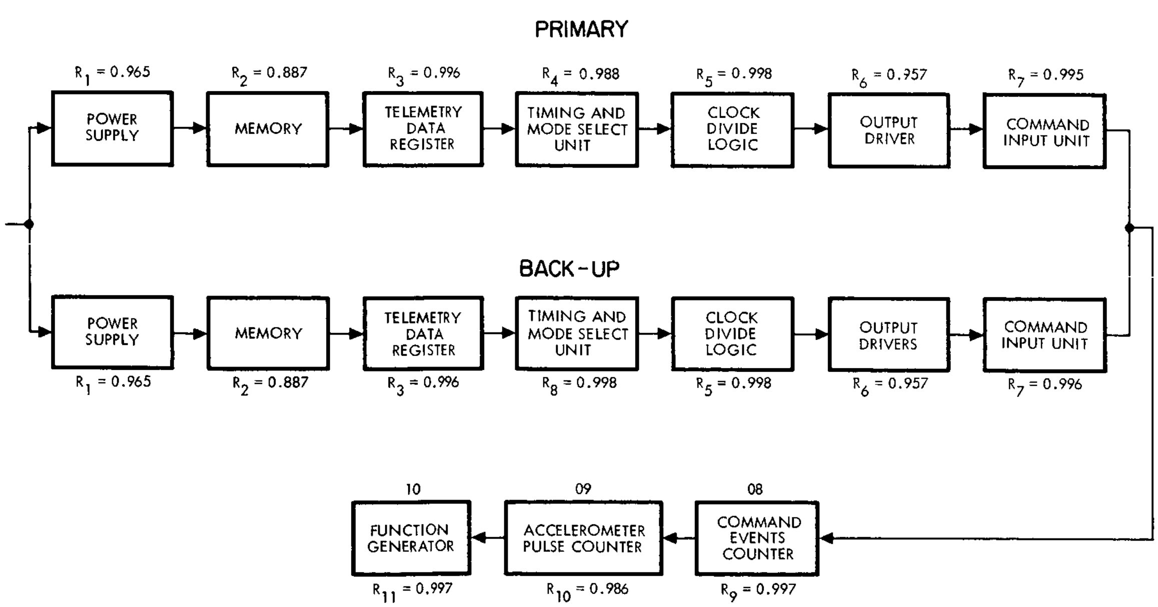
Figure 107. Typical Sequence Backup

through an input unit which accepts and decodes binary coded data and clock signals from the command subsystem.

8.5.4 Reliability Assessment

The reliability estimate for each component in the C and S subsystem is indicated in Figure 108. This assessment is based on the use of a back-up sequencer that is identical with the primary sequencer except for the omission of the function generator and the accelerometer pulse counters.

Two subsystem parts population reliability calculations were performed, with and without the accelerometer counter and function generator, which are not required for mission success, since the basic memory can perform both of these functions. For a mission from liftoff to the end of one month in Mars orbit, the total subsystem is estimated to have a reliability of 0.942. Considering the subsystem without the accelerometer pulse counter and function generator, the reliability potential is estimated to be 0.959. These estimates are conservative since any component failure is assumed to be catastrophic, whereas in many instances the result would be degraded but continued operative.



TOTAL SUBSYSTEM RELIABILITY = 0.942

SUBSYSTEM RELIABILITY (WITHOUT ACCELEROMETER PULSE COUNTER AND FUNCTION GENERATOR) = 0.959

Figure 108. Reliability Block Diagram

9. STRUCTURAL SUBSYSTEM

9.1 General Description

The structural subsystem is the framework of the flight spacecraft and provides the platform for support and spatial alignment of all subsystems and the flight capsule. It is designed to be compatible with all natural and induced environments which will be experienced from initial assembly through mission end. The subsystem features the utilization of the Lunar Excursion Module (LEM) descent stage for the basic element of the bus structure which provides structural continuity and integrates the thermal control and micrometeorite protection functions.

An exploded view of the complete spacecraft structure is shown in Figure 109 and Figure 110 is a more conventional structural configuration drawing. The structure is composed of the following major assemblies:

- Flight Capsule interstage structure
- Main equipment compartment module
- Outrigger assemblies
- Equipment mounting panels
- Aft equipment module

The mechanical section of the subsystem consists of the mechanisms required to accomplish the separation and appendage release and deployment functions. These functions include: the separation of the planetary vehicle from the nose fairing; the emergency separation of the flight capsule from the flight spacecraft; the retention and release of the high- and medium-gain antennas; the retention, release and deployment of the low-gain antenna, the planetary scan platform (PSP) and magnetometer. All functions are initiated by completely redundant pyrotechnics. The mechanical elements within this functional subsystem include the following:

- Pyrotechnic separation nuts
- Debris catchers
- Separation springs
- Pyrotechnic pin pullers

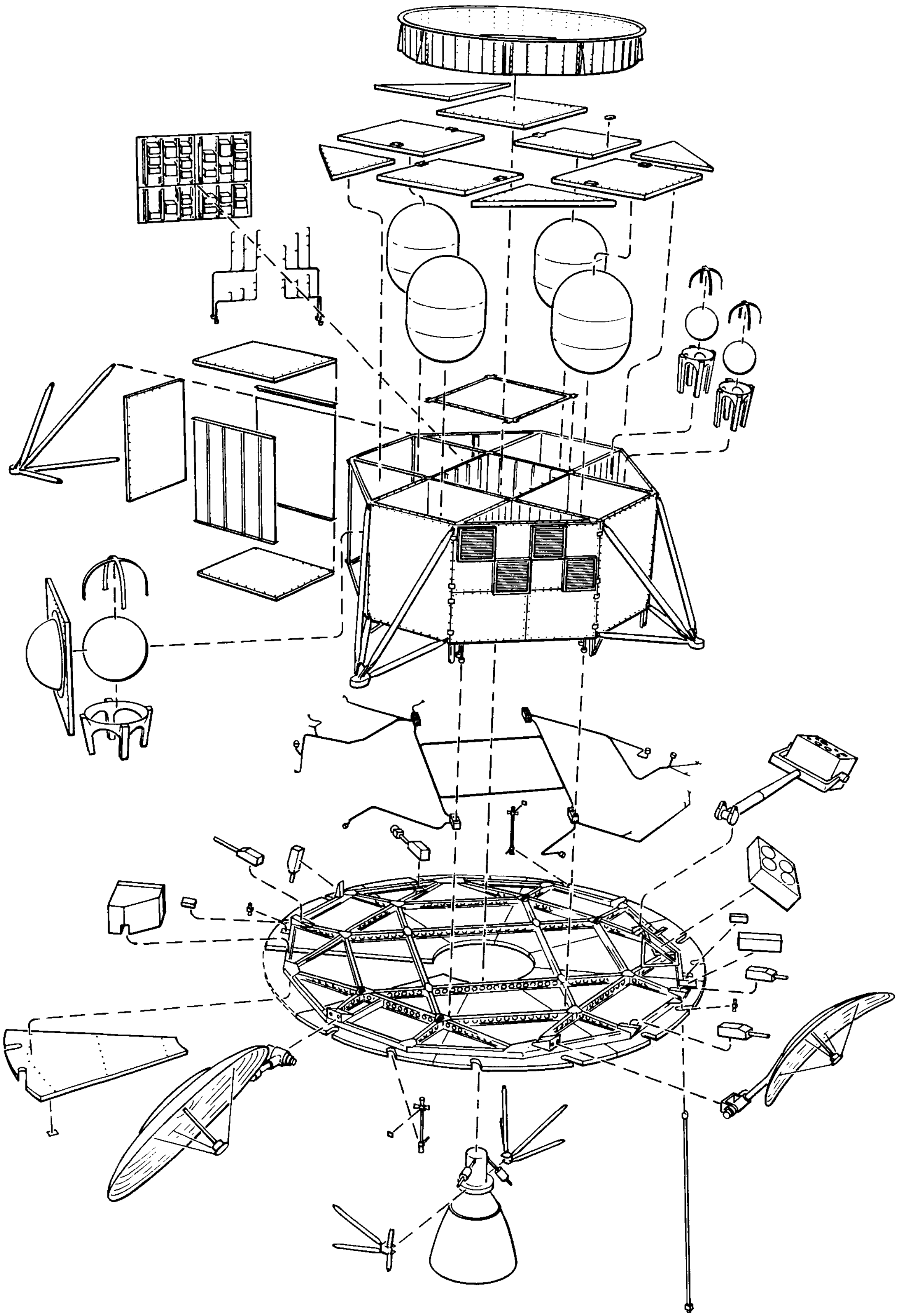
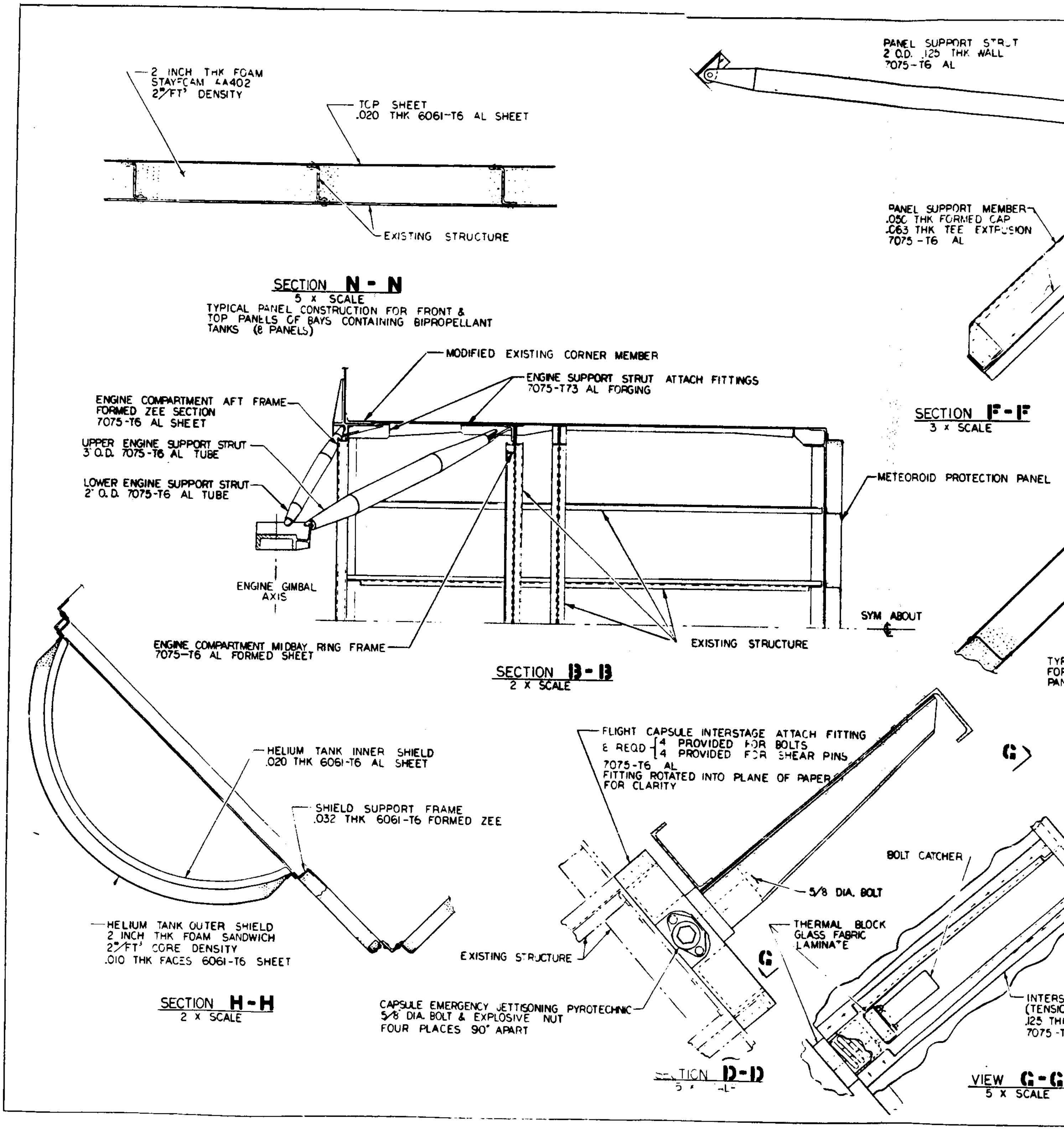
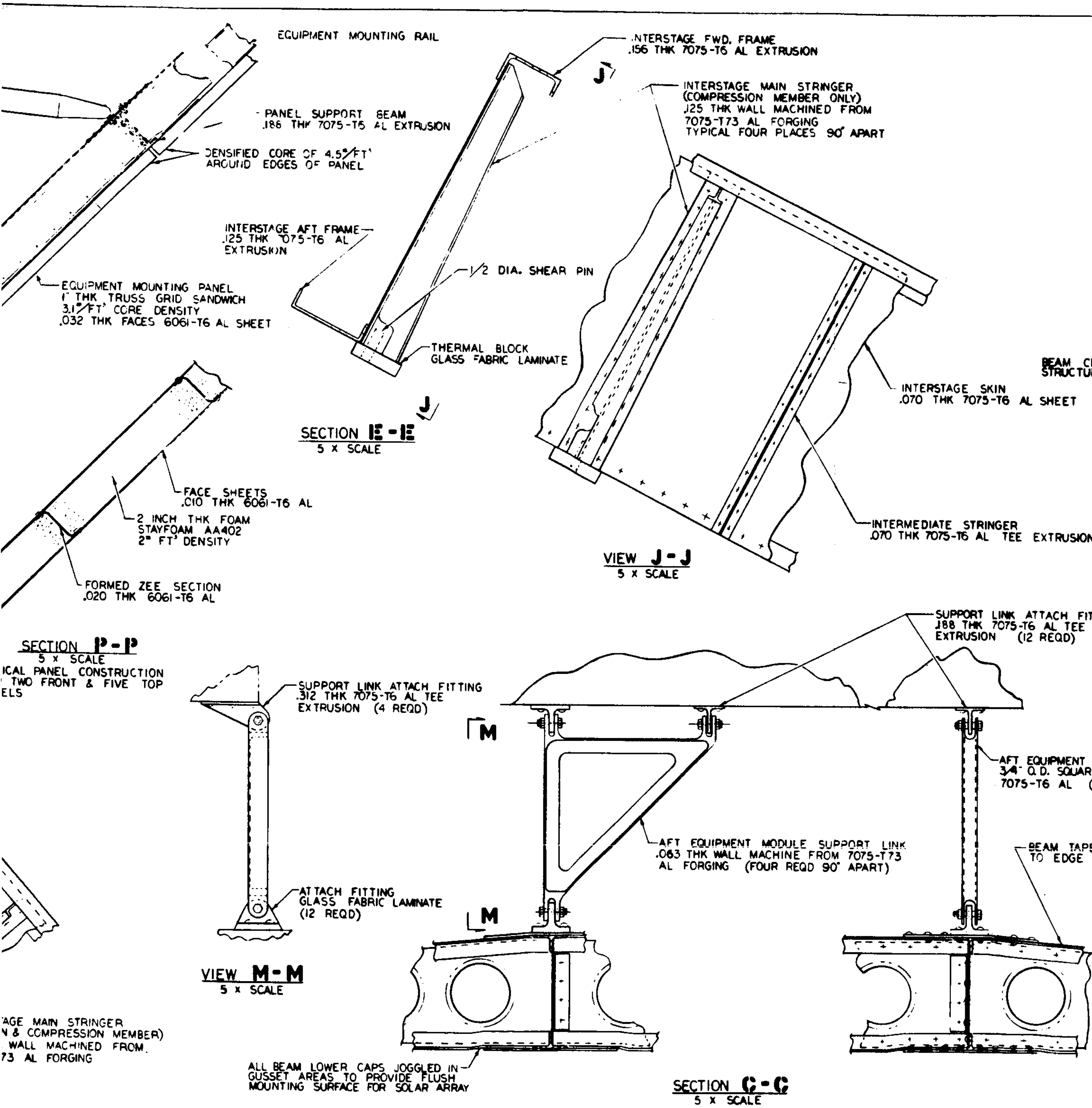
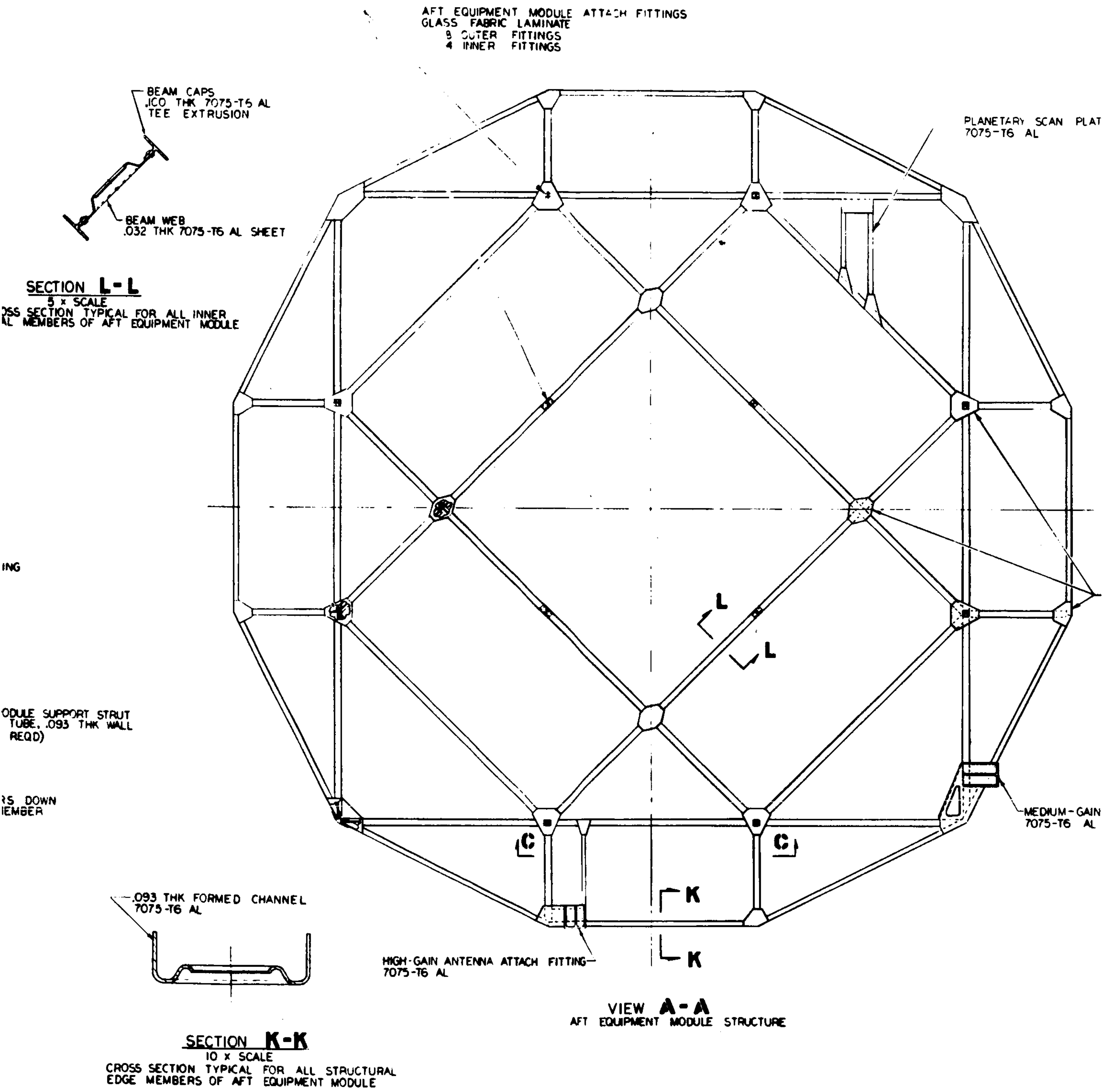


Figure 109. 1971 Voyager Spacecraft—Exploded View


2 INCH THK FOAM
STAYFOAM 4A402
2#/FT³ DENSITY
TOP SHEET
.020 THK 6061-T6 AL SHEET
EXISTING STRUCTURE
SECTION N - N
5 x SCALE
TYPICAL PANEL CONSTRUCTION FOR FRONT & TOP PANELS OF BAYS CONTAINING BIPROPELLANT TANKS (8 PANELS)
PANEL SUPPORT STRUT
2 O.D. .125 THK WALL
7075-T6 AL
PANEL SUPPORT MEMBER
.050 THK FORMED CAP
.063 THK TEE EXTRUSION
7075-T6 AL
SECTION F - F
3 x SCALE
MODIFIED EXISTING CORNER MEMBER
ENGINE SUPPORT STRUT ATTACH FITTINGS
7075-T73 AL FORGING
ENGINE COMPARTMENT AFT FRAME
FORMED ZEE SECTION
7075-T6 AL SHEET
UPPER ENGINE SUPPORT STRUT
3" O.D. 7075-T6 AL TUBE
LOWER ENGINE SUPPORT STRUT
2" O.D. 7075-T6 AL TUBE
METEOROID PROTECTION PANEL
ENGINE GIMBAL AXIS
SYM ABOUT ℄
ENGINE COMPARTMENT MIDBAY RING FRAME
7075-T6 AL FORMED SHEET
EXISTING STRUCTURE
SECTION B - B
2 X SCALE
HELIUM TANK INNER SHIELD
.020 THK 6061-T6 AL SHEET
SHIELD SUPPORT FRAME
.032 THK 6061-T6 FORMED ZEE
HELIUM TANK OUTER SHIELD
2 INCH THK FOAM SANDWICH
2#/FT³ CORE DENSITY
.010 THK FACES 6061-T6 SHEET
SECTION H - H
2 X SCALE
FLIGHT CAPSULE INTERSTAGE ATTACH FITTING
8 REQD 4 PROVIDED FOR BOLTS
4 PROVIDED FOR SHEAR PINS
7075-T6 AL
FITTING ROTATED INTO PLANE OF PAPER FOR CLARITY
5/8 DIA. BOLT
EXISTING STRUCTURE
CAPSULE EMERGENCY JETTISONING PYROTECHNIC
5/8 DIA. BOLT & EXPLOSIVE NUT
FOUR PLACES 90° APART
SECTION D - D
BOLT CATCHER
THERMAL BLOCK
GLASS FABRIC
LAMINATE
VIEW G - G
5 X SCALE

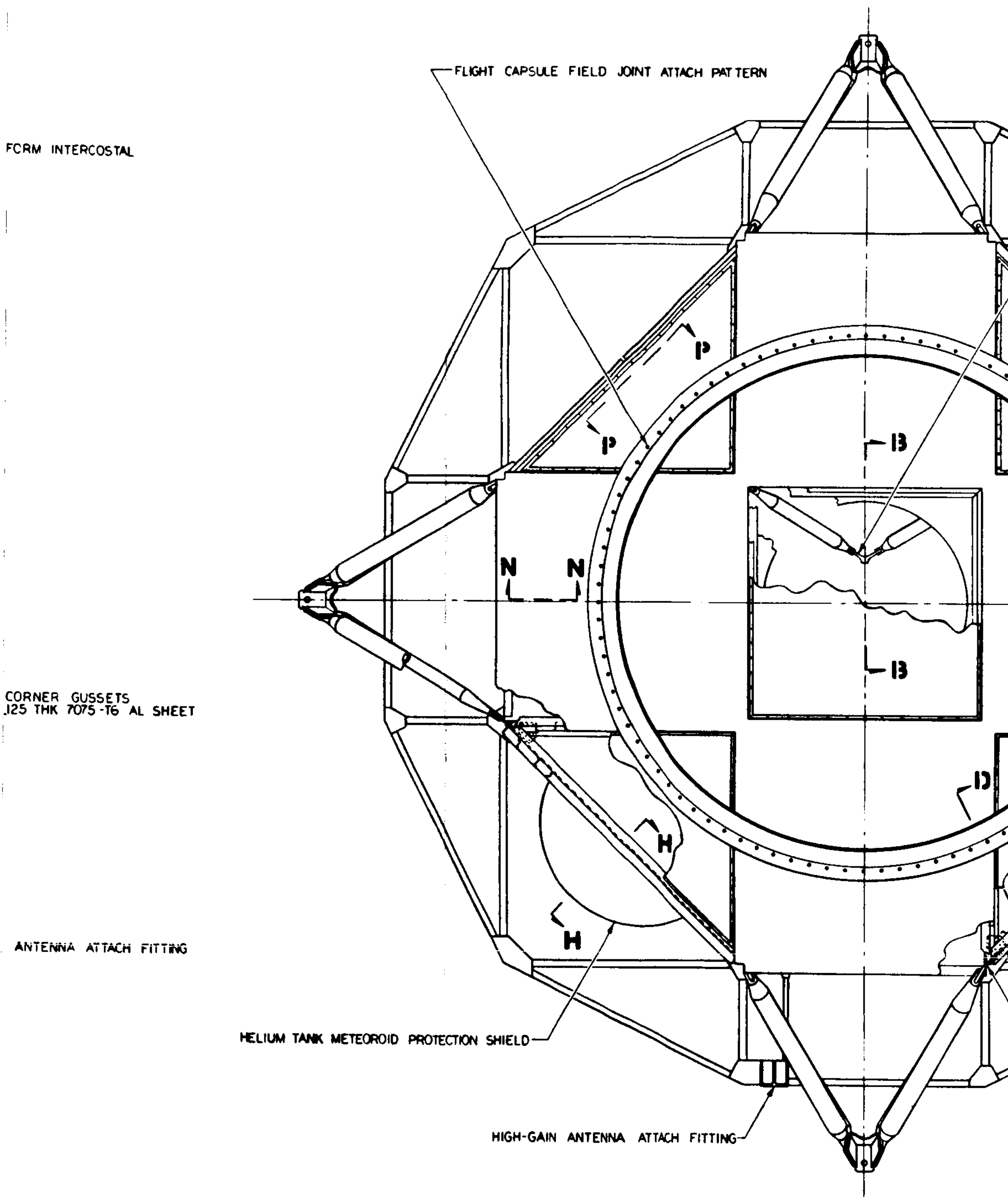

EQUIPMENT MOUNTING RAIL
PANEL SUPPORT BEAM
.186 THK 7075-T6 AL EXTRUSION
DENSIFIED CORE OF 4.5#/FT³
AROUND EDGES OF PANEL
INTERSTAGE AFT FRAME
.125 THK 7075-T6 AL
EXTRUSION
EQUIPMENT MOUNTING PANEL
1" THK TRUSS GRID SANDWICH
3.1#/FT³ CORE DENSITY
.032 THK FACES 6061-T6 AL SHEET
INTERSTAGE FWD. FRAME
.156 THK 7075-T6 AL EXTRUSION
J
1/2 DIA. SHEAR PIN
THERMAL BLOCK
GLASS FABRIC LAMINATE
J
SECTION E-E
5 X SCALE
INTERSTAGE MAIN STRINGER
(COMPRESSION MEMBER ONLY)
.125 THK WALL MACHINED FROM
7075-T73 AL FORGING
TYPICAL FOUR PLACES 90° APART
INTERSTAGE SKIN
.070 THK 7075-T6 AL SHEET
BEAM
STRUCTU
INTERMEDIATE STRINGER
.070 THK 7075-T6 AL TEE EXTRUSION
VIEW J-J
5 X SCALE
FACE SHEETS
.010 THK 6061-T6 AL
2 INCH THK FOAM
STAYFOAM AA402
2# FT³ DENSITY
FORMED ZEE SECTION
.020 THK 6061-T6 AL
SECTION P-P
5 X SCALE
ICAL PANEL CONSTRUCTION
TWO FRONT & FIVE TOP
ELS
SUPPORT LINK ATTACH FITTING
.312 THK 7075-T6 AL TEE
EXTRUSION (4 REQD)
ATTACH FITTING
GLASS FABRIC LAMINATE
(12 REQD)
VIEW M-M
5 X SCALE
M
M
SUPPORT LINK ATTACH FIT
.188 THK 7075-T6 AL TEE
EXTRUSION (12 REQD)
AFT EQUIPMENT MODULE SUPPORT LINK
.063 THK WALL MACHINE FROM 7075-T73
AL FORGING (FOUR REQD 90° APART)
AFT EQUIPMENT
3/4" O.D. SQUAR
7075-T6 AL
BEAM TAP
TO EDGE
AGE MAIN STRINGER
N & COMPRESSION MEMBER)
WALL MACHINED FROM
73 AL FORGING
ALL BEAM LOWER CAPS JOGGLED IN
GUSSET AREAS TO PROVIDE FLUSH
MOUNTING SURFACE FOR SOLAR ARRAY
SECTION C-C
5 X SCALE

2



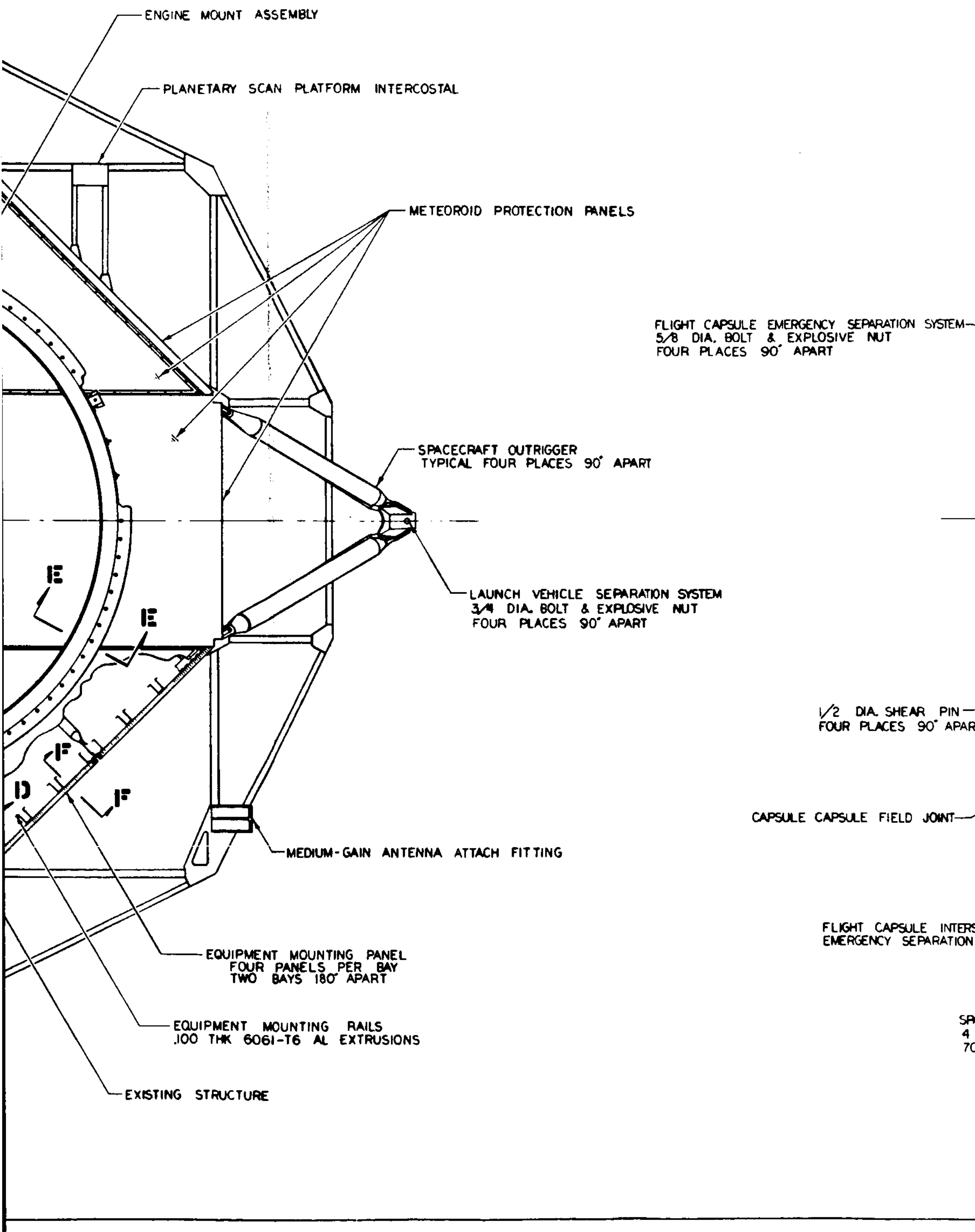
AFT EQUIPMENT MODULE ATTACH FITTINGS
GLASS FABRIC LAMINATE
8 OUTER FITTINGS
4 INNER FITTINGS
BEAM CAPS
.100 THK 7075-T6 AL
TEE EXTRUSION
BEAM WEB
.032 THK 7075-T6 AL SHEET
SECTION L-L
5 x SCALE
CROSS SECTION TYPICAL FOR ALL INNER
AL MEMBERS OF AFT EQUIPMENT MODULE
PLANETARY SCAN PLAT
7075-T6 AL
ING
ODULE SUPPORT STRUT
TUBE, .093 THK WALL
REQD)
RS DOWN
IEMBER
MEDIUM-GAIN
7075-T6 AL
.093 THK FORMED CHANNEL
7075-T6 AL
HIGH-GAIN ANTENNA ATTACH FITTING
7075-T6 AL
SECTION K-K
10 x SCALE
CROSS SECTION TYPICAL FOR ALL STRUCTURAL
EDGE MEMBERS OF AFT EQUIPMENT MODULE
VIEW A-A
AFT EQUIPMENT MODULE STRUCTURE

3



FORM INTERCOSTAL
FLIGHT CAPSULE FIELD JOINT ATTACH PATTERN
P
P
B
B
N
N
CORNER GUSSETS
.125 THK 7075-T6 AL SHEET
H
H
D
ANTENNA ATTACH FITTING
HELIUM TANK METEOROID PROTECTION SHIELD
HIGH-GAIN ANTENNA ATTACH FITTING

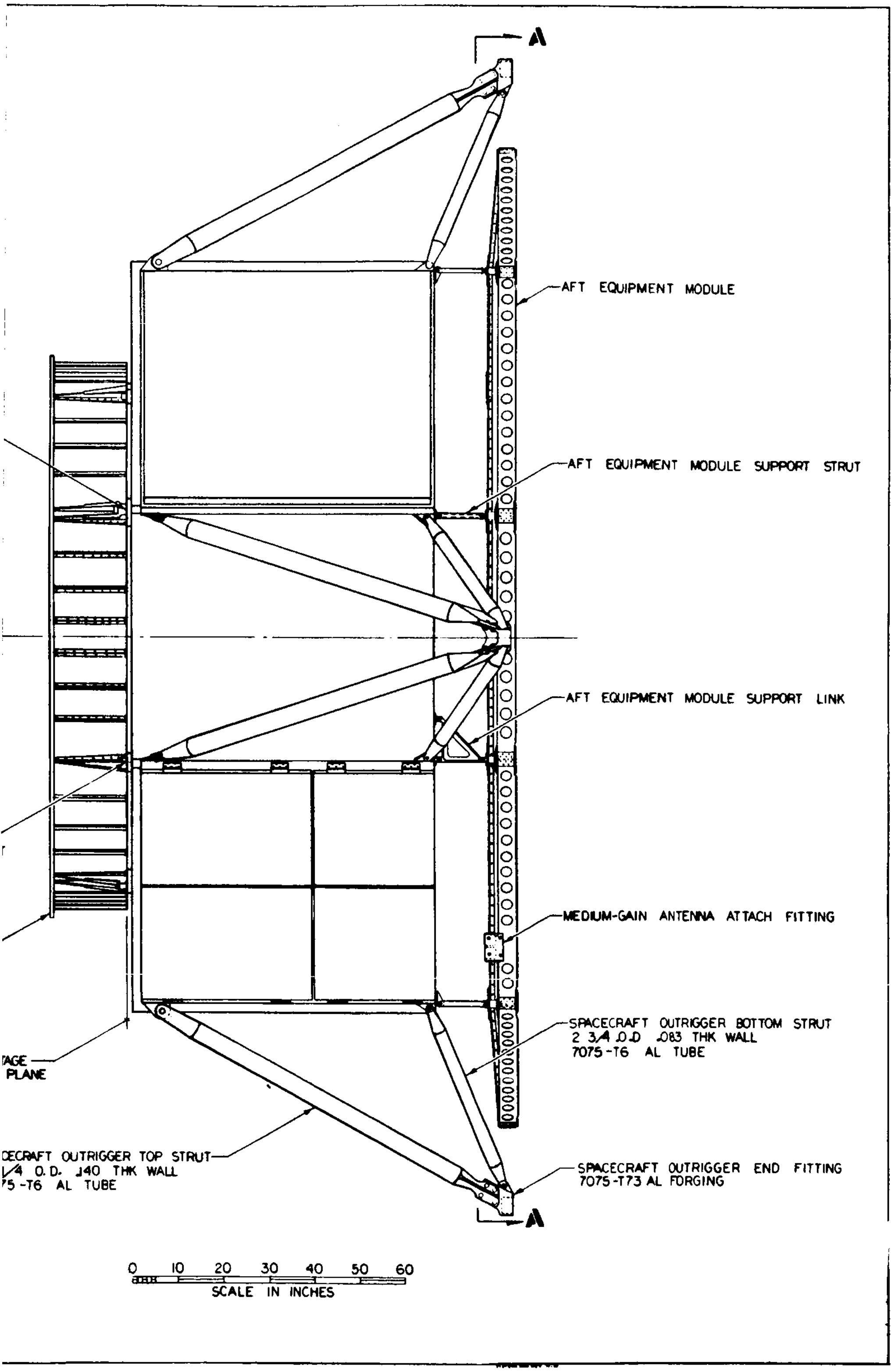
4



ENGINE MOUNT ASSEMBLY
PLANETARY SCAN PLATFORM INTERCOSTAL
METEOROID PROTECTION PANELS
FLIGHT CAPSULE EMERGENCY SEPARATION SYSTEM—
5/8 DIA. BOLT & EXPLOSIVE NUT
FOUR PLACES 90° APART
SPACECRAFT OUTRIGGER
TYPICAL FOUR PLACES 90° APART
LAUNCH VEHICLE SEPARATION SYSTEM
3/4 DIA. BOLT & EXPLOSIVE NUT
FOUR PLACES 90° APART
E
E
F
F
D
1/2 DIA. SHEAR PIN—
FOUR PLACES 90° APAR
CAPSULE CAPSULE FIELD JOINT—
MEDIUM-GAIN ANTENNA ATTACH FITTING
FLIGHT CAPSULE INTERS
EMERGENCY SEPARATION
EQUIPMENT MOUNTING PANEL
FOUR PANELS PER BAY
TWO BAYS 180° APART
EQUIPMENT MOUNTING RAILS
.100 THK 6061-T6 AL EXTRUSIONS
EXISTING STRUCTURE

Fi

5



;ure 110. Structural Configuration— 1971 Voyager Spacecraft

- Linear actuators
- Motor driven boom extenders
- Release bolts and springs

9.2 Requirements and Constraints

9.2.1 Mission Constraints

The design and operation of the structural and mechanical subsystem are constrained by three mission considerations; namely, the space environment, performance goals, and the sequence of events.

The micrometeorites that will be encountered by the Voyager spacecraft present a potentially hazardous environment from which pressurized units and sensitive electronic equipment must be protected. The particle flux, density and velocity data given in the JPL-Voyager Environmental Predictions Document, and the requirement that the flight spacecraft must perform its intended mission with the flight capsule removed from its interface, establish the level and amount of protection needed. The probability of successful operation of 0.97 in this environment is the current reliability allocation and assessment.

To achieve the quarantine goal of the mission, it is necessary for all structural materials and coatings to be stable in the space environment and compatible with the pre-launch ethyline oxide treatment. For composite structures, such as plastic laminates and sandwich panels, treatment with heat and/or ethyline oxide during fabrication should be considered so that all ejecta resulting from micrometeorite impact and penetration will be sterile. In addition, for a period of 50 years subsequent to launch, the integrity of all structural assemblies must be maintained and all mechanical devices used to initiate separation and appendage deployment must be contained to preclude Mars impact.

The use of magnetometers during interplanetary cruise dictates the utilization of non-magnetic materials and components in order to minimize the permanent, induced, and transient fields.

The mission events that impose constraints on the structural and mechanical subsystem are: sun stabilization which relates to the requirement that the bus structure be arranged to shadow the flight capsule after sun-acquisition; flight capsule emergency separation which demands that a separation system be incorporated at the interface of the capsule interstage and spacecraft and that sufficient structure be provided to support thermal insulation and micrometeorite shielding that would be required at this interface if the capsule were jettisoned; Mars orbit injection which requires special consideration of the deceleration program and its effect on appendage position during the retro-propulsion phase.

9.2.2 Subsystem Requirements

The spacecraft structure must be designed to have sufficient strength, rigidity and other physical characteristics to survive all loads induced during the operation of the Saturn V vehicle and the LEM descent engine. The stiffness characteristics are to be selected to avoid deleterious coupling with launch vehicle resonant frequencies in order to minimize the dynamic amplification of the forcing functions to the spacecraft subsystems and flight capsule.

The main equipment compartment module provides the only support for the flight capsule. The capsule inertia loads will be transmitted across the field joint at the forward end of the interstage structure, distributed through the capsule interstage, and transmitted across the emergency separation plane which is the interface of the interstage and the main equipment module. The field joint is the physical interface with the capsule and has a 10 foot diameter bolt pattern. The emergency separation joint will include provisions for structural continuity, support for mechanical and electrical separation system mechanisms, and a thermal barrier to minimize spacecraft heat losses. Deleterious dynamic coupling between the flight capsule and spacecraft is to be avoided.

All planetary vehicle inertia loads will be transmitted through the outrigger assemblies and into the adapter fittings at the nose fairing interface. The outrigger assemblies, in conjunction with the planetary vehicle adapters, must be designed to effect compatibility with the nose fairing dynamic envelope, complete mechanical interchangeability, efficient utilization of the nose fairing for the transmission of interface loads and to preclude resonance coupling with the launch vehicle. The planetary vehicle separation joint is located at the adapter forward end, at the outrigger interface, and will contain the mechanical and electrical separation system support provisions. The field joint is located at the adapter aft end and is the physical interface with the nose fairing.

Micrometeorite protection is provided for all pressurized units and sensitive electronic equipment within the spacecraft. Resistance to penetration will be inherent in the sandwich structures of the aft equipment module and electronic equipment mounting panels. Other exposed areas will be protected by suitable shielding which will be designed to disperse the particles encountered.

The structural and mechanical subsystem will be compatible with the subsystems it supports to assure compliance with the functional interface requirements of Paragraph 9.3.

9.2.3 Design Requirements

a. Structural Requirements

The spacecraft structure will be designed to withstand simultaneously the design limit loads and other accompanying environmental phenomena without experiencing excessive elastic or plastic deformation where such deformation would reduce the probability of successful completion of the mission. The design limit loads are the maximum loads that may reasonably be expected to occur in service for the design conditions under consideration multiplied by a hazard factor which is 1.0 for general structure.

The spacecraft structure will be designed to withstand simultaneously the design ultimate loads and accompanying phenomena without experiencing failure. Design ultimate loads are to be used for design, stress analysis and test as a means of assuring high probability of structural adequacy under the application of specified limit loads. Design ultimate loads are the product of the design limit loads and a factor of safety which is 1.25 for general structure.

The ability of the design to sustain the above loads will be assessed. The structure must have a positive margin of safety, equal to or greater than zero, where margin of safety is defined by the equation M. S. $= \frac{1}{R} - 1$. R is the ratio of the applied load or stress to the allowable load or stress.

The fatigue life of the structure will be predicated upon its capability to sustain the cyclic loading imposed during ground handling, acceptance vibration tests, transportation, launch, boost, separation, orbit correction and retropropulsion maneuvers. The consideration of the extended, 50 year life requirement will involve the evaluation of the fatigue strength as well as the selection and application of materials, processes, fabrication and assembly techniques that are compatible with the space environment.

The spacecraft strength and rigidity requirements of Paragraph 9. 2. 2 must be satisfied. The loads and stiffness criteria are presented in Paragraph 9. 2. 3. c.

Simple and conservative design techniques will be utilized in the development of the spacecraft structure. State of the art concepts, materials and analytical techniques will be used throughout this development. Advantage will be taken of the Ranger, Mariner and other NASA program experience where feasible. Standard and qualified parts and assemblies will be used, as exemplified by the utilization of the LEM descent stage which provides the major portion of the main equipment compartment module.

b. Mechanical Requirements

Redundant pyrotechnic devices are employed to initiate the separation of the planetary vehicle from the nose fairing, separation of the flight capsule from the flight spacecraft and the release of all appendages. The separation and release functions must be accomplished if either or both of the devices are actuated by the electrical signal supplied.

The flight capsule separation mechanisms for the 3, 000 pound and 10, 000 pound capsule systems will be identical. The separation mechanism will impart the impulse necessary to provide the flight capsule with a separation velocity of 0. 5 ± 0. 25 feet per second, relative to the flight spacecraft. The angular velocity imparted to the capsule will be less than one degree per second about each principal axis.

The planetary vehicle separation mechanisms for all planned missions will be identical. The separation mechanism will impart the impulse necessary to provide the planetary vehicle with a separation velocity of 0.5 ± 0.25 feet per second, relative to the launch vehicle. The angular velocity imparted to the planetary vehicle will be less than one degree per second about each control axis.

Although all appendages are released with pyrotechnic devices, only the low-gain antenna and planetary scan platform (PSP) are mechanically deployed. Redundant linear actuators are to be located at the root hinge of each appendage and store the potential energy which, at time of deployment initiation, is converted to the kinetic energy required for deployment. The actuators are to be designed so that the design load factor at hinge latching is limited to a maximum of 1.0g with either or both actuators operable.

For all mechanical equipment, compliance with the reliability requirements of the "Voyager Project Reliability Assurance Program Plan RA001BB" is mandatory.

Conformance to the range safety requirements, as delineated in AFETR 127-1 and M127-1, is essential for all pyrotechnic devices.

Standard and qualified parts will be used wherever possible. Materials and processes will comply with the "Voyager Project Process Selection List RA001BB-1B".

c. Assumptions and Criteria

The design limit loads imposed on the structural and mechanical subsystem are derived by the logical combination of the steady state and low frequency accelerations presented in the Voyager Environmental Predictions Document. These loads presumably account for the effects of all launch and in-flight quasi-static and transient phenomena which occur during the Saturn V operation. The dynamic amplifications of the forcing functions are based on the following assumptions:

1) At lift-off, the longitudinal and lateral excitations will decay to 10 percent of the initial values in 20 cycles, which corresponds to approximately 1.8 per cent of critical viscous damping for the launch vehicle.

2) The lift-off transient will occur in the first 2 longitudinal modes of the vehicle, $f_1 = 4.0$ cps and $f_2 = 8.7$ cps, with energy participation factors of 0.7 and 0.3, respectively.

3) The lift-off transient will occur in the first 3 lateral modes of the vehicle, $f_1 = 0.2$ cps and $f_2 = 2.1$ cps and $f_3 = 2.2$ cps, with one-third of the composite signal existing in each mode.

4) At engine cutoff, the appropriate transient will occur in the first longitudinal mode ($f_1 = 7.1$ cps) and the first lateral mode ($f_1 = 2.8$ cps) of the launch vehicle after the thrust has rapidly diminished to zero.

5) The equivalent viscous damping for the planetary vehicle support will be 5 per cent.

The design limit loads imposed on the structural and mechanical subsystem to account for the LEM descent engine induced loading during the midcourse correction, retropropulsion and orbit trim maneuvers are derived as follows. For the design of the engine support structure, a maximum axial thrust load of 9000 pounds is used. This is derived by combining the nominal thrust level of 7800 pounds with a 3σ thrust overshoot of 1200 pounds. The forcing function is characterized as a narrow band random oscillation having a center frequency of 35 cycles per second. For equipment other than the thrust structure, design limit load factors of 2.0g axial and 1.0g lateral are used. These factors are intended to account for the steady state and transient conditions which would occur at retropropulsion cutoff with the capsule having been jettisoned previously, and with the allocated propellant for interplanetary trajectory corrections having been consumed.

The external loads induced during ground handling are less than those design limit loads established to account for in-flight conditions.

During the flight acceptance vibration test, the input acceleration will be controlled through the primary cantilevered modes of the spacecraft so that the applied loads do not exceed the design limit loads specified in this paragraph.

During the type approval vibration tests, the input acceleration will be controlled through the primary cantilevered modes of the spacecraft so that the design ultimate loads are not exceeded. The design ultimate loads are the product of the design limit loads specified herein and a factor of safety of 1.25.

The pyrotechnic shock loading, in addition to the acoustic and random vibration environments specified in the Voyager Environmental Predictions Document, are not considered to be critical for the design of primary structural assemblies; however, the resultant loads will be used in the design and development of an environmental test specification for electronic equipment and mechanical hardware.

The design limit load factors for the critical conditions are given at the Planetary Vehicle center of gravity. The spacecraft is designed to sustain the combined loads for each event.

Table 63. Design Limit Load Factors

Flight Event	Steady State		Dynamic	
	Axial	Lateral	Axial	Lateral
Launch release	1.5g	0.5g	±3.6g	±0.45g
S-IC burnout	5.6g	0.5g	---	--
S-IC cutoff	--	--	±2.0g	±0.5g
LEM retropropulsion	1.0g	0.5g	±1.0g	±0.5g

The stiffness characteristics of major assemblies are selected to avoid deleterious coupling with the launch vehicle resonant frequencies and to minimize the dynamic amplification to the flight capsule. The rigidity criteria, expressed in terms of natural frequencies for major assemblies, are given.

- Spacecraft with respect to nose fairing (axial) > 10 cps
- Spacecraft with respect to nose fairing (lateral) > 5 cps
- Flight capsule with respect to spacecraft > 100 cps
- Appendage support with respect to spacecraft > 75 cps
- Aft equipment module with respect to spacecraft > 50 cps
- Equipment mounting panels with respect to main equipment compartment module > 75 cps

9.3 Functional Interfaces

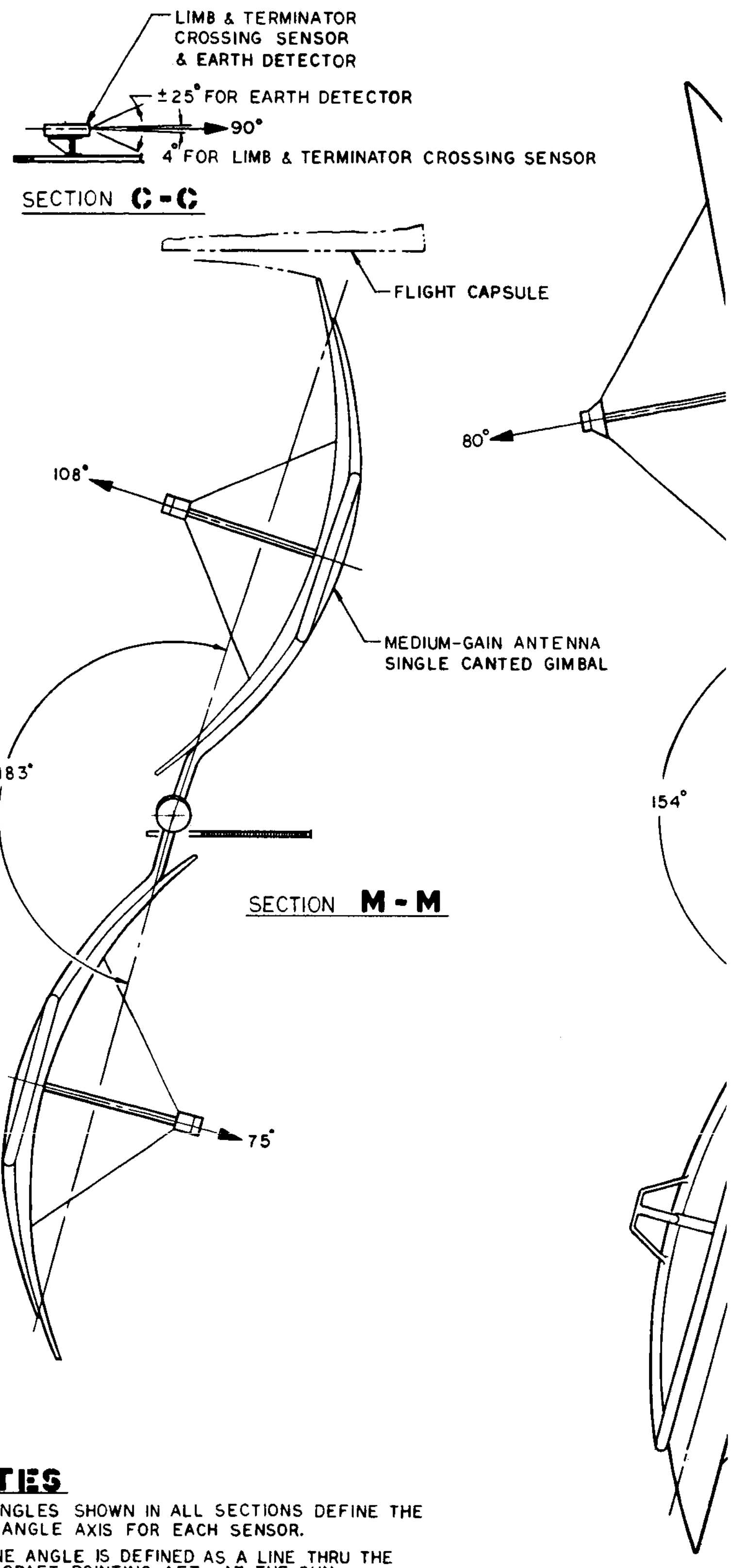
9.3.1 Science

The science equipment is arranged within the spacecraft as shown in Figures 111 and 112. The aft equipment module supports the Planetary Scan Platform, the magnetometer and ion chamber boom assembly, science antennas, and fixed science packages. One-half of equipment mounting Panel VII supports the data automation equipment, the power switching electronics and other ancillary equipment for the PSP and sensor instruments.

The support for the science assemblies mounted on the aft equipment module has sufficient strength and rigidity and other physical characteristics to assure compliance with the view angle and alignment accuracy requirements of Volume 1, Section III. The strength of the support for the fixed science and stowed appendages is capable of sustaining an acceleration of 15g in all directions when applied at the equipment package center of gravity. For the attachment holes, drill templates will be provided to assure interchangeability and precise alignment. The structural attachment consists of a pattern of 10-32 titanium screws. The support will exhibit a natural frequency greater than 75 cps. The flatness and surface finish of the mounting interface will be controlled. Rigid mounting brackets for the electrical harness are provided. To accommodate sun-oriented sensors, 3 inch diameter view ports are provided at the intersection of the solar array panels.

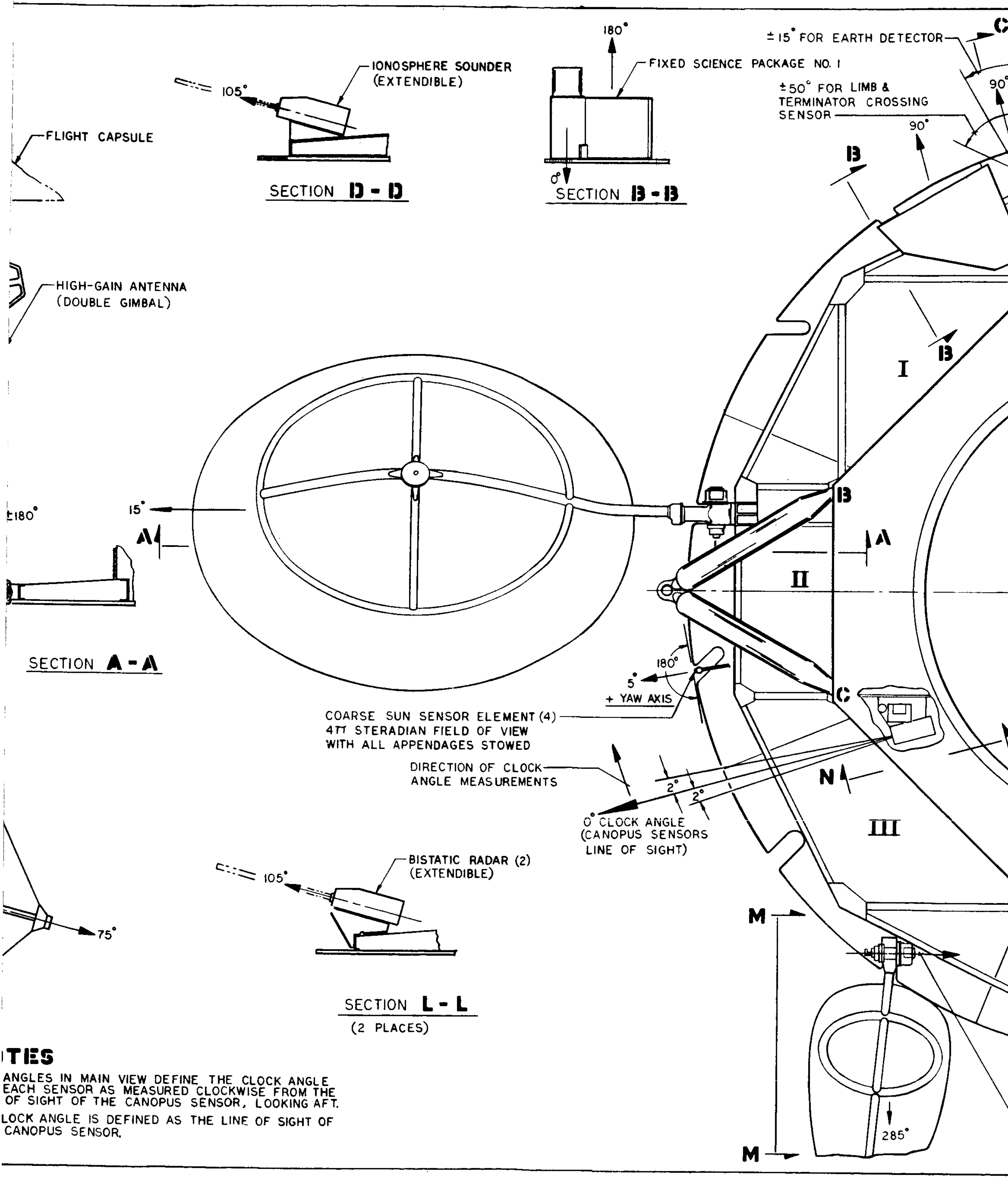
In the deployed configuration, suitable materials and coatings (ie, silver plated beryllium copper) must be selected for the science antennas to minimize thermal distortion in order to avoid a potential incursion into the field of view of the spacecraft sensors. Induced appendage oscillations during flight must be less than the electronic noise in the sensor circuits to avoid a deleterious interaction with the guidance and control system.

Structural provisions for the science equipment mounted on Panel VII of the main equipment compartment module will be identical to those



NOTES

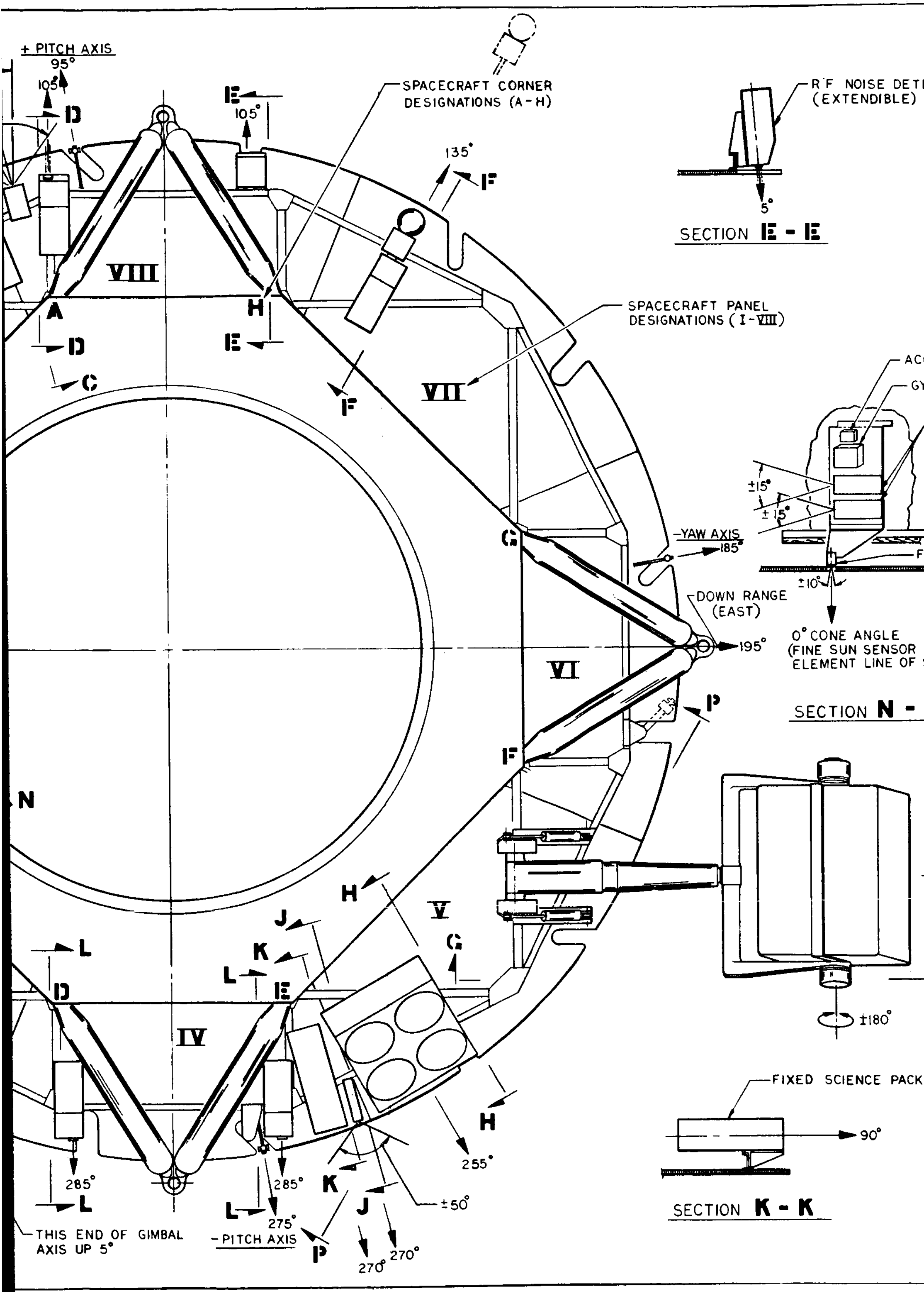
1. ALL ANGLES SHOWN IN ALL SECTIONS DEFINE THE CONE ANGLE AXIS FOR EACH SENSOR.
2. 0° CONE ANGLE IS DEFINED AS A LINE THRU THE SPACECPAFT POINTING AFT – AT THE SUN.
3. SECTIONS SHOWN HAVE BEEN ROTATED FOR CLARITY.



TES

ANGLES IN MAIN VIEW DEFINE THE CLOCK ANGLE EACH SENSOR AS MEASURED CLOCKWISE FROM THE OF SIGHT OF THE CANOPUS SENSOR, LOOKING AFT.

LOCK ANGLE IS DEFINED AS THE LINE OF SIGHT OF CANOPUS SENSOR.



+ PITCH AXIS
95°
105°
SPACECRAFT CORNER DESIGNATIONS (A-H)
R F NOISE DET
(EXTENDIBLE)
SECTION E - E
135°
SPACECRAFT PANEL DESIGNATIONS (I-VIII)
VIII
VII
VI
V
IV
-YAW AXIS
185°
DOWN RANGE (EAST)
195°
0° CONE ANGLE
(FINE SUN SENSOR
ELEMENT LINE OF
SECTION N -
±15°
±10°
±180°
FIXED SCIENCE PACK
90°
SECTION K - K
255°
±50°
285°
275°
270°
THIS END OF GIMBAL AXIS UP 5°
- PITCH AXIS

3

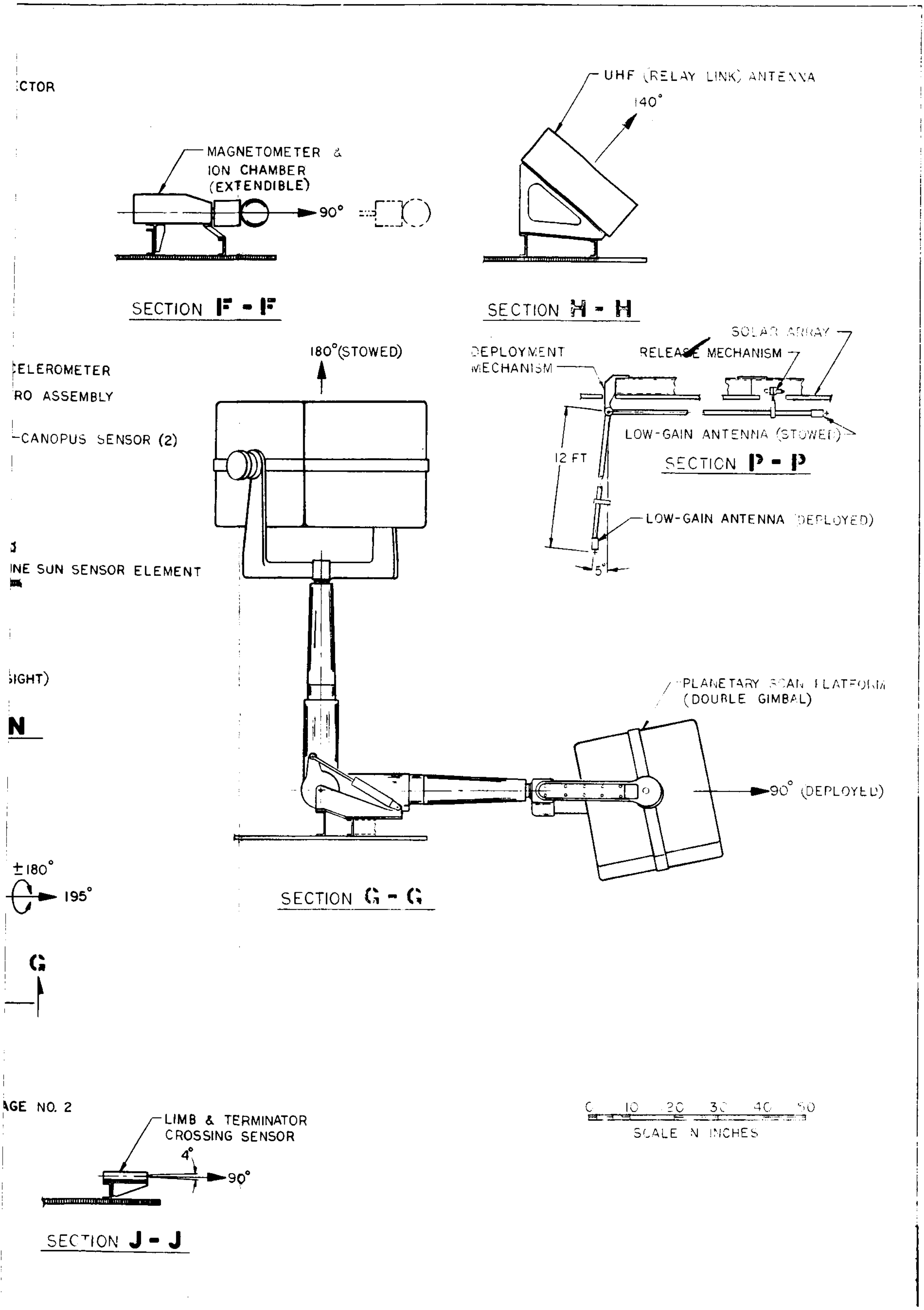
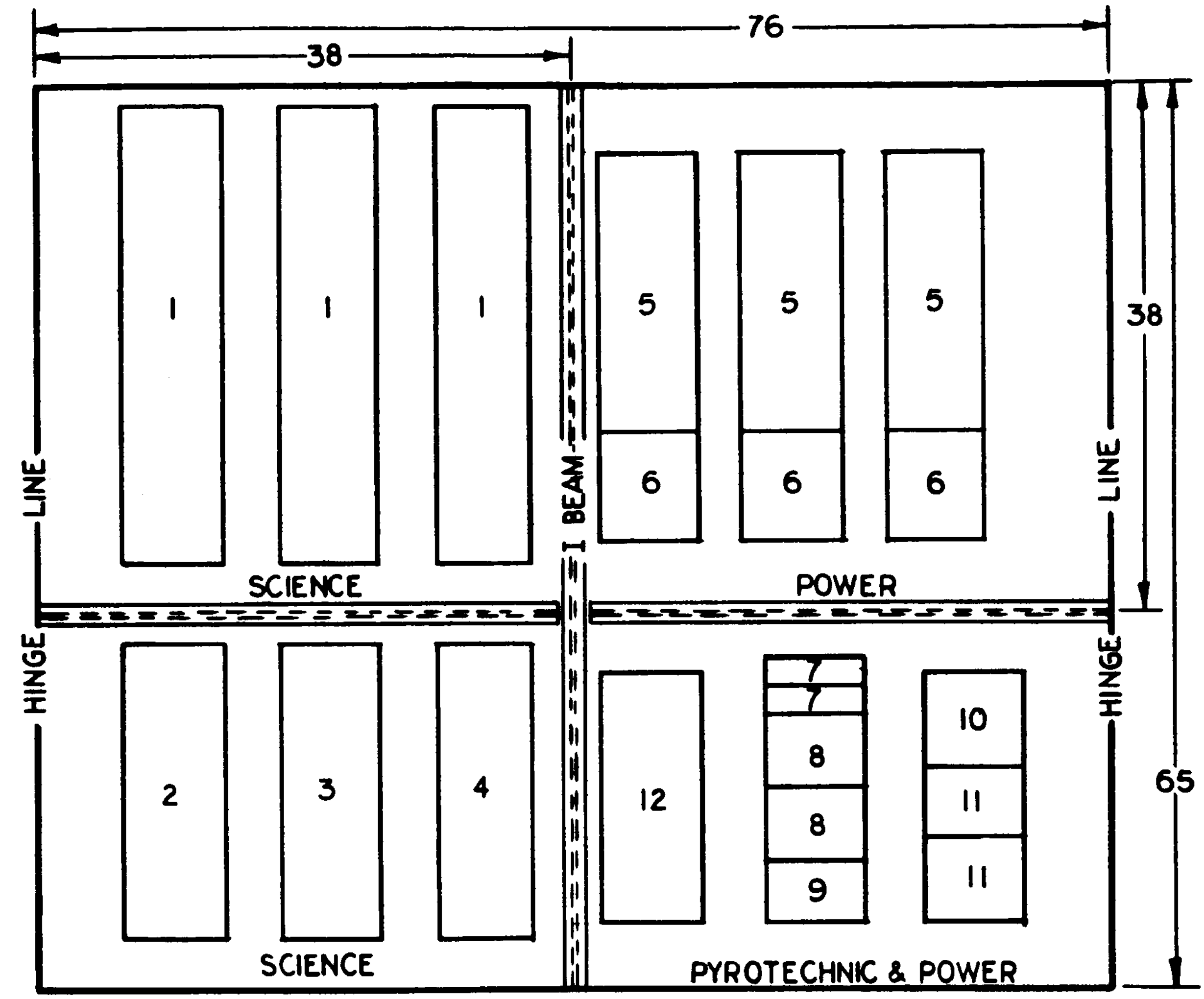


Figure 111. Sensor Geometry—1971 Voyager Spacecraft



	ITEM	QUANITITY	SIZE EACH	WEIGHT/ UNIT	TOTAL WEIGHT
SCIENCE					233.0
1	REMOTE HARDWARE P.S.P.	3	7×6×33.0	43.0	129.0
2	DATA AUTOMATION	1	7×6×22.0	37.0	37.0
3	REMOTE HRDW. DEPLOY. SEN.	1	7×6×22.0	23.0	23.0
4	" " NONDEPLOY. "	1	7×6×22.0	44.0	44.0
POWER					232.1
5	BATTERY	3	7×8×19.5	46.0	138.0
6	BATTERY REGULATOR	3	7×8×7.2	14.0	42.0
7	400 CPS INVERTER	2	7×6×2.0	3.0	6.0
8	SHUNT ELEMENT	2	7×6×5.2	8.0	16.0
9	POWER CONTROL UNIT	1	7×6×4.5	8.0	8.0
10	POWER DISTR. & SWITCHING	1	7×6×7.0	7.5	7.5
11	4.0 KC INVERTER	2	7×6×5.2	7.3	14.6
PYROTECHNIC					25.0
12	PYROTECHNIC CONTROL	1	7×6×17.0	25.0	25.0

Figure 112. Panel VII Equipment Installation

of the aft equipment module. Other mechanical and thermal interface provisions will be identical to the equipment support provisions delineated in the following paragraph 9.3.4.

9.3.2 Flight Capsule

The flight capsule is supported by the flight capsule interstage which provides an interface field joint. The joint consists of a ten-foot diameter, multi-bolt attachment pattern. The field joint provides for flight capsule—flight spacecraft alignment within the required tolerance. An emergency capsule separation joint is incorporated at the interface between the adapter and the propulsion module structure.

The stiffness characteristics for the flight capsule interstage are selected to produce an axial and lateral response which will minimize the dynamic loads imposed on the flight capsule during critical transient accelerations and to avoid violation of the nose fairing dynamic envelope during maximum lateral excitation of the launch vehicle at launch release and transonic flight. The stiffness of the flight capsule interstage is controlled by the material, effective stiffener area and height of the structure and the attachment to the propulsion module.

The flight spacecraft provides for an 8 point support of the flight capsule interstage. The hard points support all static and dynamic loads. Tension loads will be carried by separation bolts at four of the hard points. Shear loads will be carried by 4 shear pins rather than the separation bolts. The flight capsule adapter provides a matching pattern of hard points.

9.3.3 Planetary Vehicle Adapter

All planetary vehicle loads are transmitted to the planetary vehicle adapter through the four outrigger assemblies. The rigidity of the adapter and nose fairing are to be presumed infinite with respect to the planetary vehicle for purposes of calculating dynamic loads.

The 4 point field joint for planetary vehicle attachment is located at Stations 3340.13 and 3716.13. The planetary vehicle separation joints are at Stations 3344.63 and 3720.63 directly forward of the field joint attach points.

The planetary vehicle separation system complies with the following interface requirements:

a) The nose fairing will accept and distribute the 4-point loading from each of two planetary vehicles. The adapter ring sections must be joined at each interface to distribute the lateral shear loading.

b) The nose fairing will furnish the electrical harness and connectors for each interface.

c) The separation signal is generated by the launch vehicle and causes initiation of the separation system pyrotechnic devices.

d) Each spacecraft must be compatible with the nose fairing support points and dynamic envelope.

9.3.4 Interfaces With Other Spacecraft Subsystem Elements

a. Temperature Control

The main equipment compartment module serves as a thermally controlled environment for all internally mounted components. On the external surface provisions will be made for mounting thermal insulation and for attaching louver assemblies to the radiating areas of equipment panels. Radiative areas will be treated to have a low solar absorptivity (≈ 0.25); most internal surfaces will be treated to have a high infrared emmissivity (≈ 0.85).

Equipment panels will have a transverse thermal conductance of at least 7 Btu/hr ft^2 °F and a lateral thermal conductivity-thickness product of at least 6 Btu in/hr ft°F. Internal surfaces of the equipment panels, where components are to be mounted, will have a flatness of approximately 0.004 inch per foot.

Mechanical joints bridging the main compartment to the solar array structure and to the flight capsule will be designed to impede heat transfer. Wherever possible internal mechanical joints will be designed to promote heat transfer.

b. Propulsion

The propulsion module, which provides a major portion of the spacecraft structure, is a separate entity and can be tested as such prior to the integration of the other subsystem modules. The engine mount structure will locate the engine on the spacecraft centerline within a radial tolerance of .032 inches. The centerline of the engine will be parallel to the spacecraft centerline within a tolerance of 0.5 degree. The propulsion module structure also provides meteoroid protection for the propellant tanks and the feed system.

c. Electronic Packaging

Relatively heavy components are mounted so as to minimize dynamic amplification through the support structure and are secured in a manner that will provide adequate strength. Equipment mounting rails are utilized on the equipment mounting panels to facilitate the installation of electronic packages and to stiffen the panels such that the fundamental mode is above the primary modal frequencies of the launch vehicle and planetary vehicle. The structure is designed to assure intimate contact with the electronic components to effect adequate dissipation of the heat generated. Standard attach patterns will be provided for the installation of equipment. Hinges are provided along the outside vertical edge of each equipment mounting panel to facilitate access. Assembly harnesses are rigidly mounted and all connectors are accessible.

d. Electrical Bonding

All spacecraft primary structural members will be electrically bonded together per Military Specification MIL-B-5087 B (ASG), Bonding, Electrical, and Lightning Protection for Aerospace Systems, Class "L" (Lightning) and "R" (RF Potential). The structural interface with the planetary vehicle adapter will be conductive which will result in a low impedance path. This will serve as a lightning strike conductive path and will enhance RF shielding effectiveness.

e. Antenna Interfaces

Mechanical interfaces are provided for attaching the low-, medium-, and high-gain antennas to the spacecraft structure. All three antennas are rigidly supported from the aft equipment module near the edge of the solar array. Retention and release mechanisms are provided at the extremities of the antenna support booms at the interface with the spacecraft structure.

f. Solar Array

The solar array panels are attached to and supported by the aft equipment module. Interface interchangeability is inherent in the design. The design criteria developed for the solar array support structure include a stiffness requirement selected to preclude high dynamic loads on the solar array. The optimum stiffness is predicated on minimizing the dynamic amplification of transient accelerations at launch release and avoiding resonant coupling between the solar array and its support structure. A fundamental frequency of 50 cycles/second was selected to achieve this design goal.

g. Stabilization and Control

The support for the reaction control system on the aft equipment module and the guidance and control sensors within the main compartment module is designed to assure compliance with the view angle and alignment accuracy requirements detailed in Volume 1.

9.3.5 Engineering Measurements

Crystal and strain-gauge type accelerometers will be mounted on the planetary vehicle adapter to verify the steady state and low-frequency acceleration input to the planetary vehicle and to permit correlation with and analysis of ground test data. These measurements will be transmitted through the launch vehicle telemetry system and will be used to verify the adequacy of the test criteria.

Microphones (pressure transducers with a range from static to high frequency) will be similarly employed to verify sound pressure fields.

9.4 Design Description

9.4.1 Structure

The spacecraft structure is composed of five major elements: 1) the LEMDE propulsion module structure; 2) equipment mounting panels; 3) the aft equipment module structure; and 4) the flight capsule interstage structure; 5) the outrigger assemblies. The spacecraft structural arrangement was shown in Figure 110 and a preliminary specification summarizing the major properties of the structure is given in Figure 113.

The propulsion module structure serves as the unifying spacecraft element. This structure has the shape of an octagonal prism which, when interconnected with Items 2, 3, 4 and 5, forms the structural subsystem of the spacecraft. All planetary vehicle loads are transmitted through four outriggers which attach to the planetary vehicle adapter. This module supports the majority of electronic and ancillary science equipment. The electronic equipment is located on two of the eight side panels of the module. The spacecraft propulsion subsystem, consisting of four propellant tanks, pressurant tanks, the feed system, and the engine, is mounted in this module.

The flight capsule is supported by the flight capsule interstage structure. This element is a cylindrical semi-monocoque structure which interconnects the capsule with the forward face of the LEM propulsion module.

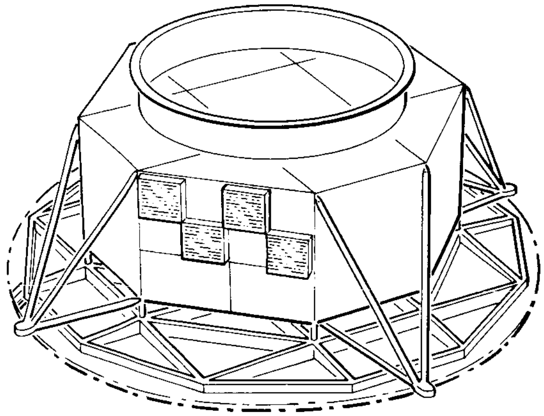
The aft equipment module is attached to the aft face of the LEM propulsion module and provides support for the solar array, antennas, Planetary Scan Platform, interplanetary science packages and the reaction control system.

a. Modified LEM Propulsion Module Structure

The external shape of the LEM module is a modified octagon 162 inches wide across the flats and 65 inches high. Two pairs of transverse beams arranged in a cruciform, together with an upper and lower bulkhead closures provide the main structural support. The space between the intersections of the beams forms the center compartment which incorporates mounting provisions for the engine support structure. The beams are of semi-monocoque construction with 7075-T651 aluminum webs, stiffeners and caps. The upper and lower bulkheads are also constructed of 7075-T651 aluminum skins with 2-inch deep "Z" section

PRELIMINARY SPECIFICATION

Structural Subsystem



Subsystem Function

Provides structural integration, support and environmental protection for the bus subsystems and mounting provisions for the flight capsule.

Subsystem Characteristics

The subsystem is composed of a main propulsion support structure, capsule interstage, outriggers and solar array and equipment support structure.

Performance Characteristics

LOAD FACTORS	LONGITUDINAL Static	LONGITUDINAL Dynamic	LATERAL Static	LATERAL Dynamic
Lift Off	1.5	±3.6	0.5	±0.45
1st Stage Burnout	5.6	—	0.5	—
1st Cutoff	—	±2.0	—	±0.5
Retrofire	1.0	±1.0	0.5	±0.5

FACTORS OF SAFETY	Yield	Ultimate
General Structure	1.00	1.25

METEOROID PROTECTION

Spacecraft surface area ______ 364 Square feet

Mission time ______ 177 Days

Probability of zero penetration ______ 0.916

Mission reliability ______ 0.97

Performance Characteristics

Configuration

COMPONENT	OVERALL DIMENSIONS	WEIGHT	MAT'L AND CONSTRUCTION
Main propulsion support structure	166 hex x 138 inches high	1152 pounds	7075 Al Semi-monocoque
capsule interstage	120 dia x 17 inches high	149 pounds	7075 Al Semi-monocoque
Outriggers	82 x 54 x 46 inches	180 pounds	7075 Al tube truss
Solar array and equipment support structure	234 diameter x 6 inches deep	219 pounds	7075 Al beam gridwork
SUBSYSTEM	—	1700 pounds	—

Interfaces

FLIGHT CAPSULE - Bolted flange 120 inches diameter bolt circle

P/V ADAPTER - 4 Points - 250 inches diameter

Figure 113. Preliminary Specification—Structural Subsystem

stiffeners. Bulkheads, of the same construction as the upper and lower bulkheads, close off the four ends of the cruciform beam assembly. Diagonal beam assemblies join adjacent corners of the cruciform structure at the upper and lower surfaces.

Additions to the basic LEM propulsion structure are required to accommodate the flight capsule interstage attachment and to provide meteoroid protection, equipment mounting panel attachment, support for the aft equipment module, and retention interfaces for the PSP and antennas. Minor modifications are required for attachment of new outriggers and relocated engine.

The flight capsule interstage attaches to the LEM structure at eight points on a 120-inch diameter circle. The attach points will be located at the intersection of the circle and the transverse beams. The attachment fittings, as shown in Figure 110 are 2-inch thick 7075-T73 fittings that attach to the beam caps and the two adjacent "Z" web stiffeners. A one-inch thick fiberglass spacer is installed over the forward face of the fittings and serves as a heat block between the spacecraft and the flight capsule. The interface of the fiberglass spacer with the interstage is the emergency separation plane.

Additional meteoroid protection above that inherent in the basic LEM structure is needed to satisfy the Voyager requirements. On the upper bulkheads in the bays above the propellant tanks and on the cruciform end bulkheads, 2-inch thick low density foam will be bonded to the existing webs between the "Z" stiffeners and an additional .020 6061-T6 sheet will be attached to the outer legs of the stiffeners. The top surface of the center compartment, the four triangular top panels and the side panels in the two pressurant tank bays will consist of sandwich panels with .010 6061-T6 skins and 2-inch thick low density foam core. Meteoroid protection for the aft face of the module will be provided by the solar array. The equipment mounting panels will supply the protection for the two equipment bays.

As shown in Figure 110, the two equipment bays will be modified to provide attachment and support for the equipment mounting panels. Each bay will be divided into four rectangular areas. A vertical "I" beam will run from the top diagonal beam to the bottom diagonal beam and a horizontal "I" beam will run between adjacent main transverse beams. A horizontal tube will extend from the intersection of the horizontal and vertical "I" beams to the intersecting corners of the adjacent transverse beams.

The four outriggers are similar to the general arrangement of the LEM outriggers and attach to the same points on the LEM structure. However, they extend further outboard and aft for the Voyager application. The new design consists of two 4-1/4 inch diameter 7075-T6 aluminum tubes, which run from the planetary vehicle adapter interface to the upper caps of the transverse beams, and two 2-3/4-inch diameter 7075-T6 aluminum tubes which attach to the beam lower caps. The fitting that attaches to the outboard end of the four outrigger tubes has a cup-shaped cavity to accept the planetary vehicle separation system as defined in Section 9.4.2. The attach fittings on the ends of transverse beam caps will be modified due to the geometry change and the associated change in loads.

For Voyager, the engine mount must be lowered three feet to provide thrust vector control. This requires a new engine thrust mount which will attach to the lower caps of the beams and to existing structure 23 inches above the lower caps. The beam attach fittings will be new. The tubular structure is changed only to the extent necessary to pick up the new attach points for the revised engine support location. Local attachment fittings will be required for attachment of the aft equipment module.

b. Equipment Mounting Panels

The eight equipment mounting panels, which are installed on two of the LEM propulsion module structure side facets, serve as mounting panels for spacecraft and science electronics equipment. In addition, they provide meteoroid protection and serve as radiators for the thermal control subsystem. The thermal louver assemblies mount on the outer faces of the panels. The panels are of sandwich construction with one-inch thick aluminum truss grid core and 0.032 inch 6061-T6 aluminum face sheets. Extruded rails are attached to the inner side of the panels and serve as the mounting members for all electronic equipment. The rails also serve to increase the fundamental frequency of the panel. The panels are bolted to the equipment mounting bay structure with an interchangeable bolt pattern on all four sides. A hinge is provided on the vertical outside edge of the panels to facilitate equipment servicing.

c. Aft Equipment Module Structure

The aft equipment module structure consists of a gridwork of beams as shown in Figure 110. All beams, with the exception of the peripheral members are 7075-T6 aluminum "I" beams with extruded Tee caps and sheet metal webs. The members are 6 inches deep except for the tapered sections which attach to the peripheral members. The outer members are 4-inch deep 7075-T6 sheet metal channels. The attach points for the eight solar array segments are located so that panels can have identical insert patterns with a maximum of two unused inserts per panel. Attach fittings for antennas, the PSP, interplanetary science packages, and reaction control jets are mounted to the outer members of the gridwork.

The temperature difference between the aft equipment module structure and the LEM structure causes a relative growth of 5/16 of an inch across the maximum width of the LEM module. To accommodate this relative displacement the gridwork is supported from the lower surface of the LEM module by twelve pivoted links. The links are located at the corners of the engine compartment and at the ends of the cruciform beams. Four of the links, 90° apart, have only one degree of freedom and transmit both vertical and lateral loads. The other eight links carry vertical loads but are free to move laterally to accommodate the expansion across the ends of the cruciform. All links are constructed of 7075-T6 aluminum. All lugs attaching the links to the gridwork are made of fiberglass to provide a heat block between the LEM and aft equipment module.

d. Flight Capsule Interstage Structure

The capsule interstage structure provides attachment of the flight capsule to the spacecraft and is a 10-foot diameter, 17-inch high semi-monocoque structure. Loads are transmitted from the spacecraft to the interstage through 8 fittings, 4 of which carry tension and compression and 4 which carry compression and shear. The interface is the emergency separation joint for the flight capsule. The four tension fittings incorporate the separation system which is described in Section 9.4.2. The fittings are 7075-T73 aluminum bath tub fitting with a cavity at the lower end to accommodate the separation device. The compression/shear fittings are similar with the aft ends modified to accept shear pins. The fittings run the full depth of the adapter and taper to a small cross-section of the top. The cylindrical structure consists of 7075-T6 skin with extruded Tee section stringers 7 inches on center. Four-inch deep 7075-T6 extruded channel frames are used at the forward and aft ends of the interstage. The structure is designed to pick up loads at 8 points at the spacecraft interface and distribute the loads circumferentially at the flight capsule adapter interface.

9.4.2 Mechanical Systems

a. Separation Mechanisms

Pyrotechnic separation mechanisms are provided for separation of the two planetary vehicles from the launch vehicle and for emergency separation of the flight capsules. The systems are designed to provide a minimum separation velocity of 0.25 fps at both interfaces, based on maximum 1977 mission weights.

The planetary vehicle/launch vehicle separation subsystem consists of four separation fittings at the extremities of the spacecraft outriggers. A similar concept is used for flight capsule emergency separation. In this configuration, the separation mechanisms are incorporated in four of the eight fittings of the interstage. The location and arrangement of components are shown in Figure 114.

A typical separation mechanism is shown schematically in Figure 115. The mechanism consists basically of a separation bolt and ejection spring, a split holding nut with dual actuating cartridges, and a separation spring. The physical installation includes a suitable housing for the separation spring and retention provisions for all parts released on separation. The geometry is varied to accommodate the specific structural arrangement at the planetary vehicle and flight capsule interfaces. The split holding nut, which is the actual separation element, is held together in the installed position by a cross stud. The stud is threaded into non-outgassing, non-fragmenting pyrotechnic separation nuts at each end.

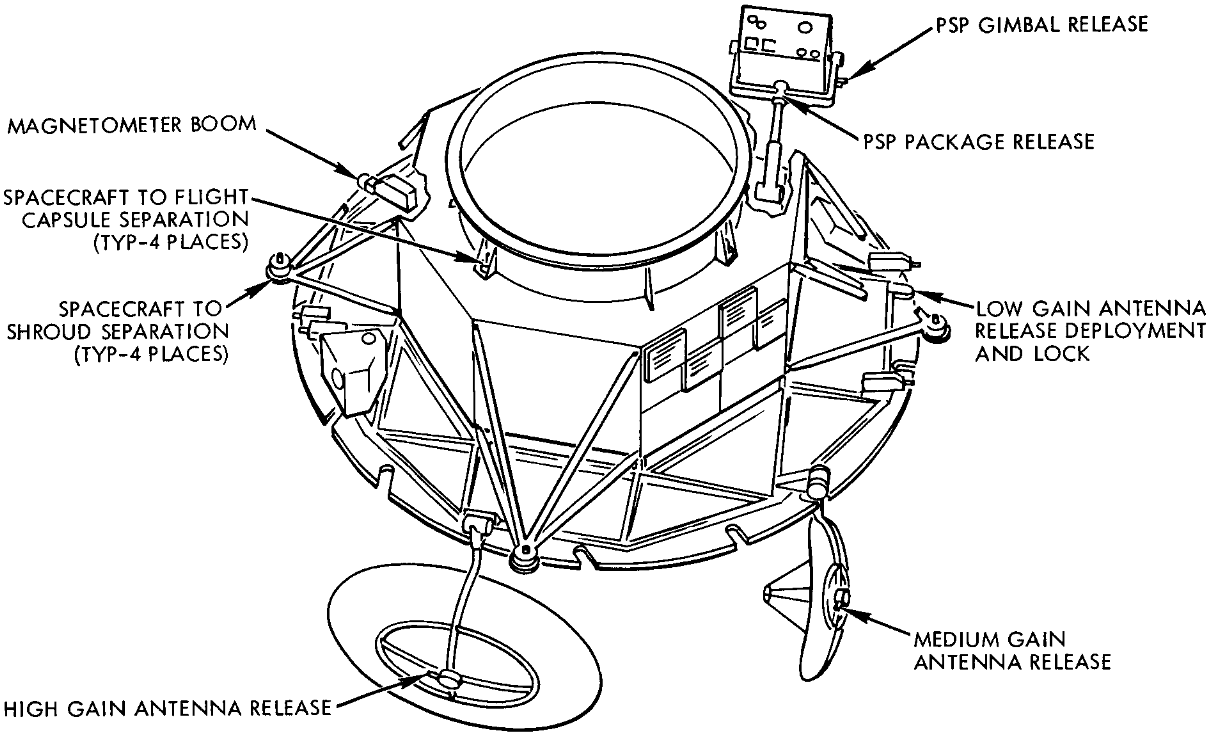


Figure 114. Mechanical Isometric Sketch

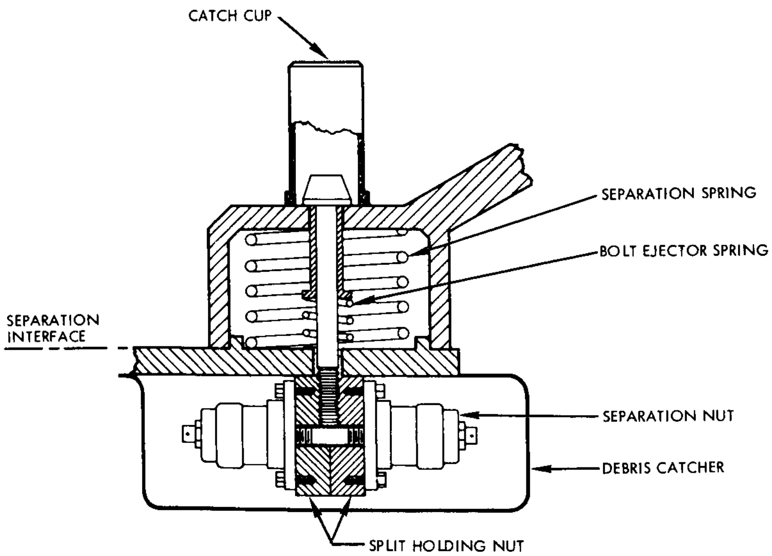


Figure 115. Typical Separation Mechanism

When separation is commanded, power is applied to dual bridgewires in the separation nut cartridges. Firing of the cartridges causes release and ejection of the holding nut halves by reaction against the cross stud. The separation bolts, which are now free, are driven out of the interface and held in the retracted position by the ejection springs. On release of the holding nuts, the compressed separation springs impart the impulse necessary to achieve the required separation velocity.

For planetary vehicle separation from the launch vehicle, all four separation points are initiated simultaneously on command from the launch vehicle. To reduce spacecraft pyrotechnic electrical power requirements, however, emergency separation of the flight capsule is sequenced such that two points at 180° are released initially. Release of the remaining two attachments (and actual separation of the vehicles) is delayed 10 minutes to allow recharging of capacitors.

The separation system is fully redundant having dual separation nuts, dual bridgewire cartridges and dual power supplies. Details of the separation nuts and cartridges are presented in Section 10 of this Volume.

b. Release Mechanisms

Release mechanisms are provided to retain the low-, medium-, and high-gain antennas, and PSP in a stowed position prior to deployment. In addition, a pitch axis gimbal release is provided for the PSP. The location and arrangement of these components are shown in Figure 114.

All of the antenna and PSP release mechanisms are similar and employ the same basic concept as the vehicle separation mechanisms, differing only in size and element geometry. A typical configuration is shown in Figure 116. Operation of these components is commanded by the spacecraft sequencer and is identical to that described previously for the separation mechanisms.

The outboard PSP gimbal axis is retained in a locked position by a dual pin puller. Dual cartridges each capable of actuating the pin against a 3000-pound side load provide the necessary redundancy. The mechanism is shown schematically in Figure 117.

On command from the C&S, power applied to dual bridge wires ignites the cartridges which actuate separate pistons in the pin

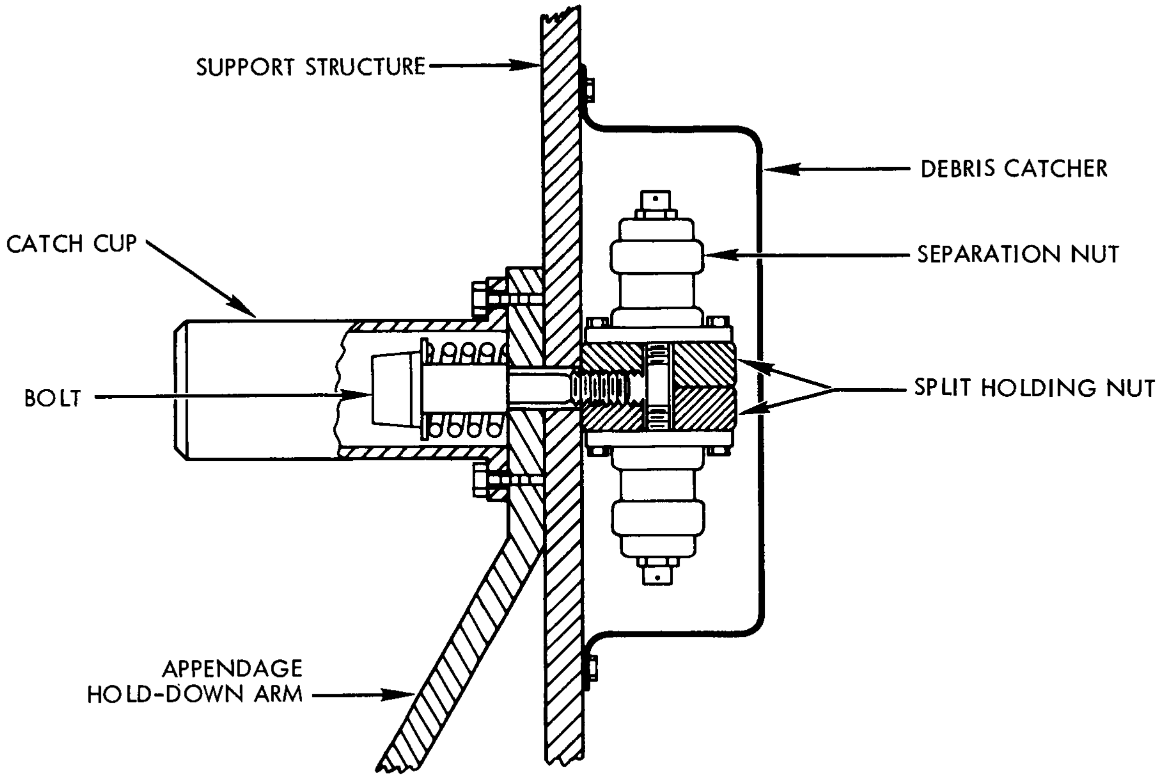


Figure 116. Typical Stowage Release Mechanism

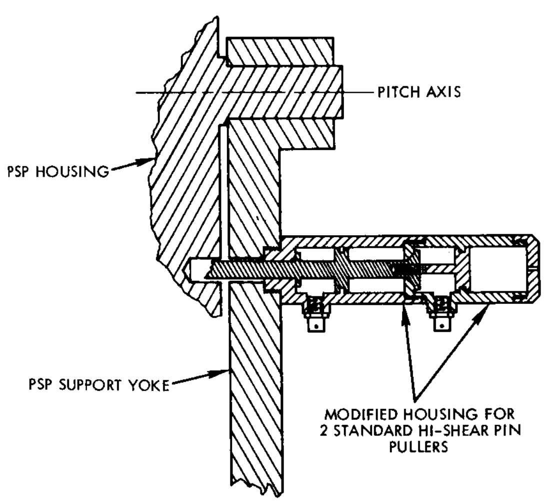


Figure 117. PSP Gimbal Latch System

puller. This action shears a small pin in the main locking member which is then withdrawn, completing the releasing function.

c. Deployment Mechanisms

Mechanisms are provided for deploying the low-gain antenna, PSP, and the magnetometer from the stowed to the operating positions. The low-gain antenna and PSP hinge actuators are designed for deployment only while the magnetometer actuator is capable of either extension or retraction on command. The location and arrangement of these installations is shown in Figure 114.

The low-gain antenna and PSP actuators, shown schematically in Figure 118, are conventional linear type actuators incorporating an integral lock at the full travel position. Rate damping is provided by a silicon fluid hydraulic damper. Locking is achieved at the end of the stroke by expansion of a locking ring into a relieved step in the bore.

Magnetometer deployment is accomplished by means of a DeHavilland, Model A-32, storable, tubular, extendable member (STEM). This particular model, shown in Figure 119, was developed specifically for magnetometer applications.

The boom is a one-inch diameter element, formed from several layers of beryllium copper strips of various lengths. When retracted, the element is stored in a strained, flattened condition on a drum. As the circular element is retracted, it is smoothly transformed into a flattened condition by passing it through a suitable guide system. The element can be extended to any length up to 20 feet or retracted by rotating the drum in the appropriate direction. The unit is powered by an electric motor and extension or retraction positioning is achieved by controlling the time period of power application. The leadwires to the sensor are deployed through the tube by a system integral with the STEM unit.

9.5 Parameters and Performance Summary

9.5.1 Performance Parameters

Using the load criteria and factors of safety shown in Section 9.2.3, and planetary vehicle weight allocations, the internal load distribution for the selected configuration was determined for the various flight conditions.

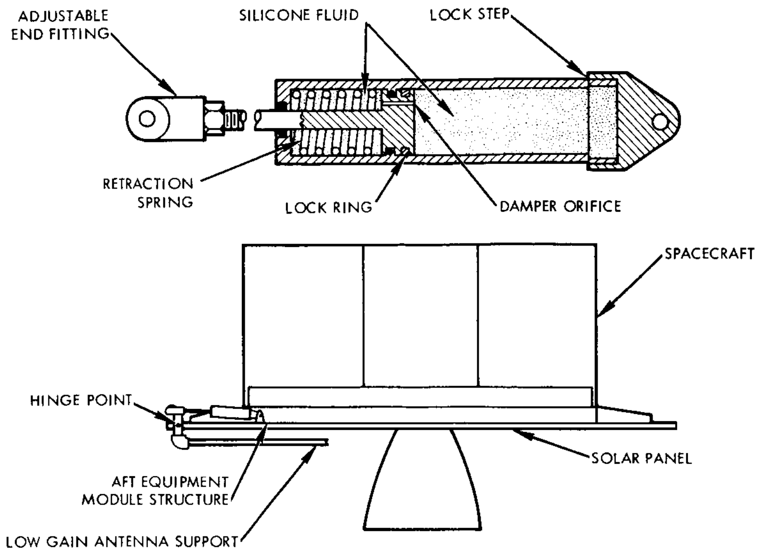


Figure 118. Low-Gain Antenna and PSP Deployment Mechanism

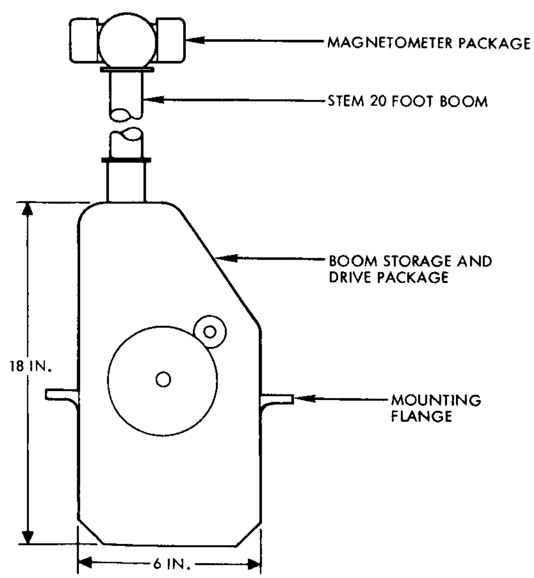


Figure 119. Magnetometer Boom Deployment Mechanism

Major structural members were analyzed using the critical loads to insure positive margins of safety and to provide a sound basis for structural weight calculations. The structural analysis is not included because of the page limitation, but it is available for review. Table 64 lists the members analyzed, their critical load and condition, and their margins of safety. A comparison of the required and achieved spacecraft stiffness characteristics is presented in Table 65.

The flight spacecraft structural and mechanical subsystem weight presented in Table 66 is divided between the spacecraft structure and the spacecraft propulsion. The spacecraft structure includes items associated with the electronic equipment support modules, the outriggers, and the flight capsule adapter. All other items, including the basic LEM propulsion module structure and all meteoroid protection, are incorporated in the spacecraft propulsion. Latches, panel hinges, miscellaneous supports, equipment rails, and reaction control system supports are based on data generated during Task A. The basic LEM propulsion module structure is derived from assumptions and JPL information. All other weights are based on stress analysis information. The weight summary for the PV adapter is shown also. Six percent is allowed for attachments and miscellaneous items which have not been detailed at this time.

The structural resistance to micrometeorite penetration was determined using the environment shown in the Voyager Environmental Prediction Document and errata transmitted 18 October. The techniques are the same as those used in Task A. The design point shown on Figure 120 is based on the selection of the Summers and Charters penetration equation and a 177 day nominal cruise time. The surface area is based on the premise that the spacecraft must perform its mission without the flight capsule, and that the solar array and support structure completely shield the bottom of the spacecraft.

Figure 120 also shows the effect on probability of no puncture of the meteoroid shield due to the variation in penetration equations and capsule sizes.

It is clear that to design a meteoroid shield of sufficient thickness to reduce the probability of no penetrations to a level such as 0.95 or

Table 64. Margins of Safety

Item	Description	Load	Margins of Safety
Truss System			
Truss A-B & A-C	4-1/4 OD x .140 x 7075-T6 Tubular	45,200 lbs. comp. Launch release	0 (stability)
Truss A-D & A-E	2-3/4 OD x .083 x 7075-T6 Tubular	21,900 lbs. Launch release	+.11 (stability)
Separation Bolts			
Adapter	3/4-in. dia. x 160 KSI H. T. Bolt	33,900 lbs. Launch release	+.86 (tension)
Interstage	5/8-in. dia. x 160 KSI H. T. Bolt	22,800 lbs. Launch release	+.91 (tension)
Separation Nut Mechanism			
Adapter	5/8-in. dia. x 160 KSI H. T. Retainer Bolt	23,500 lbs. Launch release	+.85 (tension)
Interstage	3/8-in. dia. x 160 KSI H. T. Retainer Bolt	14,700 lbs. Launch release	+.03 (tension)
Structure of Aft Module	6-in. deep beams	--	Designed for stiffness
Solar Array Support Links	7075-T6 Alum. Alloy Forging	482 lbs. Launch release	+.03 (crippling)
Capsule Interstage			
Skin	.070 x 17 (7075-T6 alum)	605 lbs/in. shear Launch release	+.002 (buckling)
Stiffeners	.75 x 1 x .070 tee (7075-T6 alum)	2090 lbs compr. Launch release	+.05 (stability)
Lower Ring	1 x 4 x .125 Channel (7075-T6 alum)	11,880 lbs. shear Launch release	+.35 (crippling)

Table 64. Margins of Safety (Continued)

Item	Description	Load	Margins of Safety
Upper Ring	1 x 1-1/2 x 3 x .156 Channel (7075-T6 alum)	199 lbs/in. tension Launch release	+.81 (bending)
Fittings	Machined fitting (7075-T6 alum)	14,970 lbs. compr Launch release	+.05 (stability)

Table 65. Stiffness Characteristics

Item	Natural Frequency - cps	
	Required	Achieved
Spacecraft with respect to nose fairing (axial)	> 10	12.9
Spacecraft with respect to nose fairing (lateral)	> 5	6.6
Flight capsule with respect to spacecraft	> 100	210
Appendage support with respect to spacecraft	> 75	75
Aft equipment module with respect to spacecraft	> 50	50
Equipment mounting panels with respect to main equipment compartment module	> 75	138

Table 66. Structural and Mechanical Subsystem Weight Breakdown

Item		Weight (lbs.)
Spacecraft Structural and Mechanical Subsystem		(776.7)
Equipment latches		2.4
Equipment panel hinges		2.2
Miscellaneous supports		8.7
Outriggers		180.0
Aft equipment module structure		176.1
Equipment mounting panels		106.3
Equipment mounting rails		78.0
Aft Module support linkage system		30.0
Flight capsule adapter		149.0
Attachments and miscellaneous		44.0
Spacecraft Propulsion Module Structure		(923.6)
LEM basic structure		671.5
Reaction control system supports		11.7
Meteoroid protection		226.1
Attachments and miscellaneous		14.3
Flight Spacecraft Structural and Mechanical Subsystem		(1700.3)
Planetary Vehicle Adapter Weight Allocation		1500.0
Adapter ftgs., separation hardware & misc.	37.0	
Nose fairing reinforcement & misc.	385.0	
TOTAL	422.0	
Net Weight Margin		1078.0

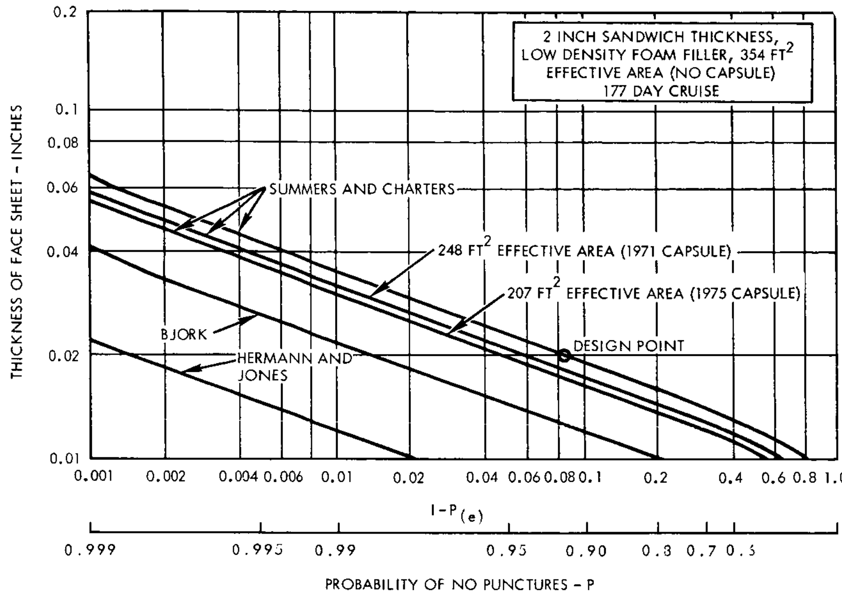


Figure 120. Sandwich Face Sheet Thickness Versus Probability of No Puncture

0.99 would be prohibitive in terms of the weight required. This also would be grossly conservative since it would tacitly be assuming that any penetration of the shield would lead to mission failure. This is by no means the case.

The mission reliability analysis performed accounted for the probability of damaging internal spacecraft components given a penetration of the meteoroid shield. Figure 121 shows the results of this analysis as a function of shield face thickness and mission time. The three lower curves represent various stay times in Mars orbit. Since the analysis showed that the puncture of the propellant tanks was an extremely important parameter in determining mission reliability, these three curves represent orbital times for which the propellant tank remained pressurized and active. Therefore, if the propellant tanks were depressurized after one month in Mars orbit, mission reliability would not significantly change even though the spacecraft continued in orbit for a total of six months.

For this study, the existing LEM propulsion module external structure was taken as aluminum sheet 0.02 inch thick with 2-inch deep "Z" sections stiffeners. This structure is modified as shown in Figure 122 to provide added meteoroid protection. Not shown on this figure is the aft heat shield which provides protection for the bottom of

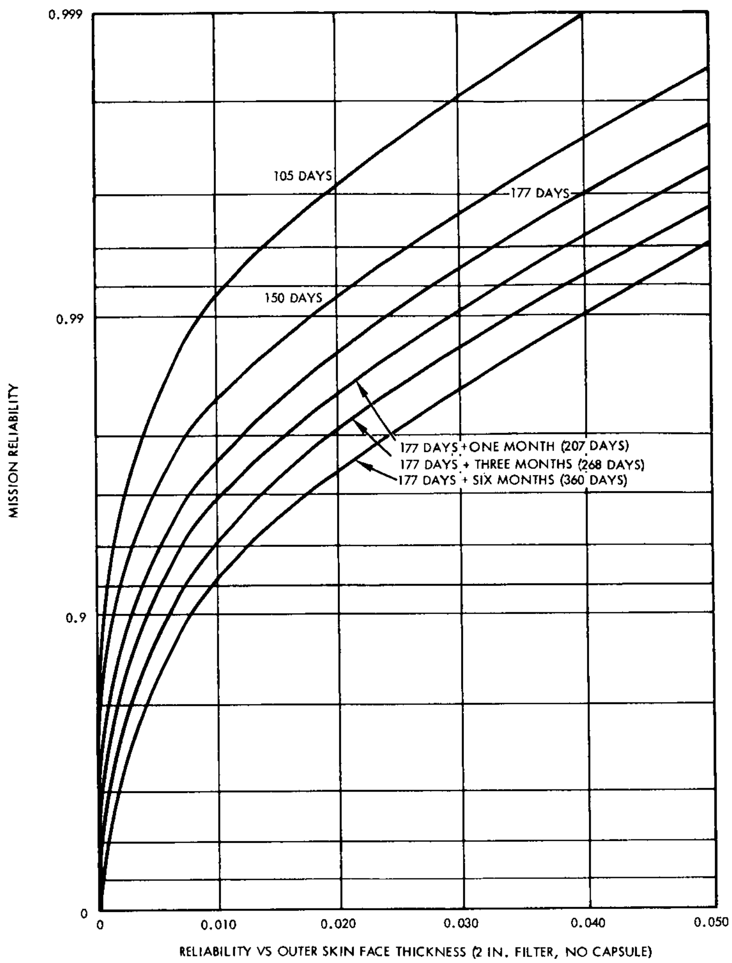


Figure 121. Reliability Versus Mission Time and Face Thickness

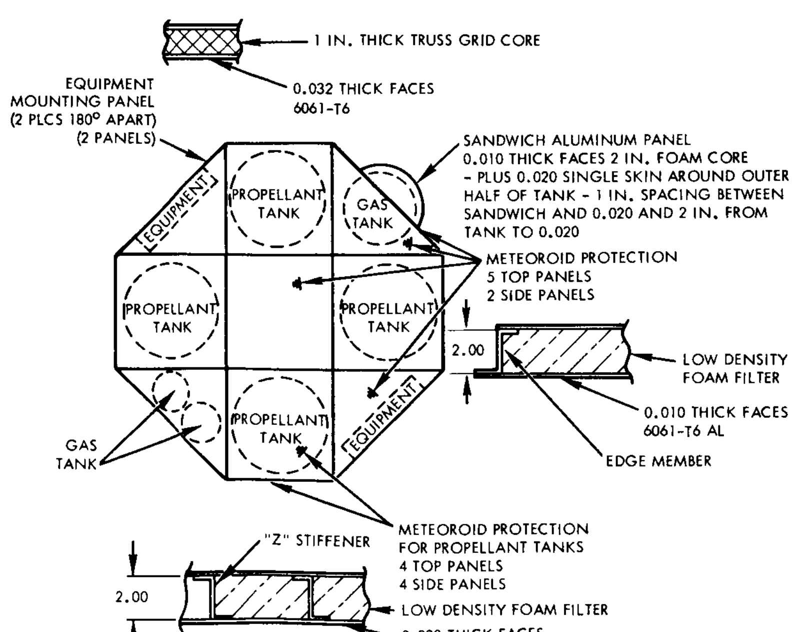


Figure 122. Panel Design for Meteoroid Protection

the spacecraft. Because of conflicting requirements for the thermal control subsystem and weight section, all the panels could not be of sandwich design with 0.02 face thickness and a low density foam core. Therefore, meteoroid protection reliability was calculated for the configuration shown on Figure 122. The resulting reliability values are as follows: for one planetary vehicle - 0.9791; for at least one of the planetary vehicles - 0.99956; and for both vehicles performing the mission - 0.9586.

a. Reliability Estimates

A statistical stress versus strength approach was employed based upon a safety factor of 1.25 and a margin of zero for each structural member. It was assumed for the lower limit that all structural elements are in series, i.e., failure of any one element results in mission failure. Since many multiple load paths exist in the structure, this assumption results in a conservative reliability estimate.

In addition to obtaining a best estimate of the structural reliability, R_s, a 3σ confidence interval was obtained to indicate the degree of uncertainty associated with reliability of the spacecraft structure. This was done for two cases, a single spacecraft and for two spacecraft. The results are shown in Table 67.

Table 67. Structural Reliability

Reliability	One Spacecraft	Two Spacecraft
R_s (lower limit)	0.9664	0.9346
R_s (best estimate)	0.99613	0.99227
R_s (upper limit)	0.999569	0.999130

For this analysis, it was assumed that catastrophic failure of any one structural member would result in mission failure. It was also assumed that all structural members have equal reliability, since they are being designed according to the same general guidelines. These two assumptions allowed calculation of a single member's reliability, which was then raised to the 75th power (the estimated number of independent structural members per spacecraft) to obtain an estimate of the reliability of the structure subsystem for one spacecraft. Reliability estimates for the various release and deployment mechanisms have been generated. Each actuator includes dual pyrotechnic devices with dual bridgewires. With this approach, all mechanisms have a reliability assessment of at least 0.9999.

10. PYROTECHNIC SUBSYSTEM

10.1 General Description

The pyrotechnic subsystem is comprised of Voyager flight spacecraft functional elements actuated by electro-explosive devices.

The subsystem can be divided into three major areas:

- The pyrotechnic control assembly
- The electro-explosive devices
- The mechanical attach-release devices

The pyrotechnic control assembly contains all of the electrical circuits required for the operation of the subsystem (see Figure 123). This includes the safe-arm circuit which controls application of power to the subsystem, the power conversion circuitry which rectifies the AC input to provide the proper DC voltage for the energy storage circuits, and solid state firing circuits which provide initiating current to individual explosive devices on command.

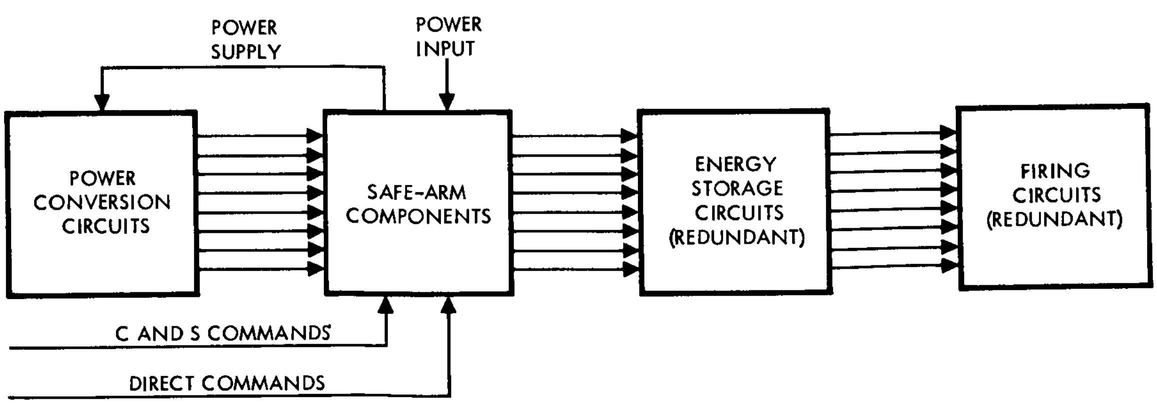


Figure 123. Diagram of the Simplified Pyrotechnics Control Subsystem

The attach-release devices are mechanical assemblies which utilize the explosive pressure impulse as the source of motive power. Each type of device is described in the structures and mechanisms subsystem description, Section 9.4 of this volume. The locations of the devices on the spacecraft are shown in Figure 124.

In a typical operation, the subsystem will be armed on receipt of a command from the spacecraft sequencer. The capacitors in the energy storage circuits will be charged, and on receipt of a second command from the sequencer the firing circuit will provide a path through the desired bridge wire for discharge of the capacitively-stored energy.

10.2 Requirements and Constraints

The following constraints are placed on the pyrotechnics by the mission:

- The AFETR P80-2 requirements for category "A" devices will apply to all devices in the pyrotechnic subsystem.
- Installation and electrical connection of the explosive devices will be in an explosive safe area as late in launch preparations as possible.
- Following prelaunch checkout, the electrical connector at the planetary vehicle adapter separation plane will be pyrotechnically separated. An indication of connector separation will be provided through the launch vehicle hardline umbilical links.

The launch vehicle supplies a pyrotechnic control subsystem for the planetary vehicle adapter separation (which releases the planetary vehicle from the launch vehicle). The pyrotechnics will not be armed prior to separation of the planetary vehicle from the launch vehicle. Normal separation of the capsule from the spacecraft shall occur under control of the capsule, based on a signal from the spacecraft.

The pyrotechnic subsystem will satisfy the following functional requirements:

- Provide a safe-arm capability for the subsystem.
- Provide explosive devices for disconnect of the planetary vehicle-vehicle adapter umbilical connector, not including the firing circuits for these devices.

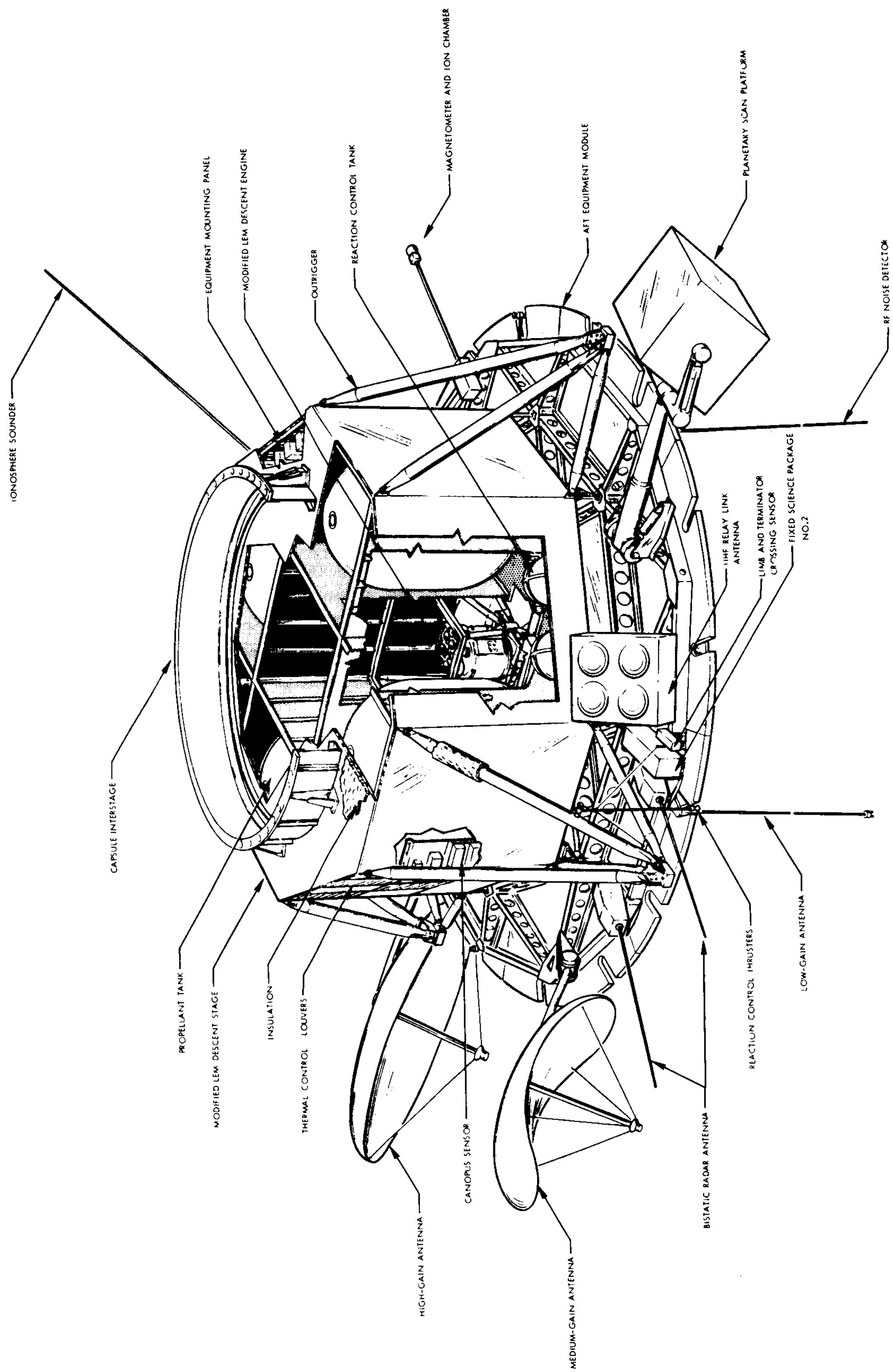


Figure 124. 1971 Voyager Spacecraft—In-flight Configuration Isometric Drawing

- Provide redundancy in both the arming and firing circuits.
- Provide explosive devices and firing circuitry for six pyro-initiated functions:
  1) Release of high- and medium-gain antennas from stowed position.
  2) Release low-gain antenna and PSP gimbal arm from stowed position.
  3) Uncage PSP gimbal.
  4) Actuate a total of 21 explosive valves controlling propulsion subsystem operation.
  5) Actuate electrical disconnect on the flight capsule-flight spacecraft separation plane prior to emergency jettison of the flight capsule.
  6) Actuate a separation mechanism to provide emergency separation of the flight capsule.

The ordnance devices in the pyrotechnics subsystem will be nondetonating, nonventing, and nonrupturing. Exploding bridge wires will not be used. All electro-explosive devices will contain redundant bridge wires, both terminating in a connector rather than pigtails. The connectors, keyed to prevent misconnection, will contain an environmental seal with the mating plug. Connector receptacles will be an integral part of the explosive device, with the male pins on the explosive side of the connection.

A minimum of 3.5 amperes applied to either bridge wire is required for firing. The explosive devices will accept 1 ampere or 1 watt for 5 minutes on each of two bridge wires simultaneously without firing and will meet this requirement without using external shunts. In addition, the static discharge of 25,000 volts from a 500-picofarad capacitor applied between pins or between pins and case will not fire the explosive devices.

Power for the actuation of all pyrotechnic devices will be supplied by capacitor discharge without imposing deleterious transients on the spacecraft power system or the other firing circuits. Current-limiting devices will be provided to prevent excessive current drain due to vehicle wiring or malfunctions in the firing circuits.

Redundant firing circuits will be provided and they will not be routed through in-flight separation connectors. Firing circuits will utilize solid state switching rather than relays for control of the initiating current. Firing circuit electrical conductors will be twisted to maintain electrical balance and reduce induction, and mating of connectors will provide for shield connection prior to making contact with the firing circuit. Firing circuits will be isolated from structure by at least 10,000 ohms. A minimum firing current of 5 amps for 50 milliseconds will be provided; the current will not exceed 22 amps. Current-monitoring devices for ground checkout and instrumentation will be included in the return leg of the firing circuits.

The possibility of a terminal-to-terminal or terminal-to-case ohmic resistance of any value from zero to infinity during or after firing will be anticipated in the circuit design. Anticipated environmental conditions will not degrade the performance of the firing circuits.

To assure electromagnetic compatibility, pyrotechnic circuits will be routed separately from all other vehicle wiring. Continuous circumferential shielding containing no electrical discontinuities and grounded to structure at both ends will be utilized from the explosive device either to the point at which the firing circuit leads are shorted together or to the firing power source. Shielding and cases will provide a minimum of 40 db attenuation over the frequency range of 150 kc to 10,000 Mc. The isolation between ground monitor and control circuits and the firing circuits will be at least 40 db from 150 kc to 10,000 Mc. The explosive devices will be be installed with a metal-to-metal contact resulting in a bonding impedance of less than 2.5 milliohms DC resistance and 80 milliohms RF impedance over the frequency range of 200 kc to 20 Mc. When installed the explosive devices will survive in an electromagnetic field of 2 watts per square meter from 150 kc to 10,000 Mc.

Qualification will include a no-fire and all-fire Bruceton analysis on each lot of devices. No single or common failure mode (including procedural deviation) will be capable of arming and firing any device in the pyrotechnic subsystem. Power to the pyrotechnic subsystem will be blocked by the normally-open contacts of the safe-arm circuits. A signal will be provided at spacecraft separation to arm the pyrotechnic subsystem.

10.3 Functional Interfaces

The power subsystem will provide AC power, 50 volts, 4 kc square waves, for input to the power conversion circuits and regulated 50 VDC for the operation of solid state drive circuits.

As part of the prelaunch sequence, electrical disconnects at the planetary vehicle-adapter separation plane are pyrotechnically actuated. The disconnect mechanism and explosive devices performing this function are supplied as part of the pyrotechnic subsystem, while the firing signals to the explosive devices are supplied by the launch vehicle. Verification of successful separation will be provided. Following booster powered flight, the forward planetary vehicle is separated from its adapter. After the intervening section of shroud is folded back and out of the way, the aft planetary vehicle is separated from its adapter. The separation mechanisms are in the pyrotechnic subsystem. The firing circuits for these separations are provided by the launch vehicle.

The computing and sequencing subsystem will provide all primary commands controlling the pyrotechnic subsystem. These commands will be 50-millisecond pulses with a 10-volt peak amplitude, capable of supplying 25 milliamps of current. Commands will be issued by the C and S to the pyrotechnic subsystem with backup either by redundant provisions in the C and S or by direct commands via the command subsystem. Commands provided are:

- Safe-arm
- Arm relays K4 through K11
- Release high- and medium-gain antennas
- Deploy low-gain antenna and PSP
- Uncage PSP gimbals
- Propulsion start-stop
- Pressurize propellant tanks

The command subsystem will provide backup commands for all of the C and S commanded pyro functions except propulsion start-stop, as well as alternate mode functions which are not a normal part of the mission sequence. The commands are in the same form as those from the C and S. The commands available are those listed above (except propulsion start-stop) together with

- Separate capsule electrical disconnect
- Separate capsule

Mechanical interfaces exist at each of the mechanical attach-release devices in the pyrotechnic subsystem. These interfaces are discussed in Section 9 of this volume.

10.4 Design Description

10.4.1 Pyrotechnic Control Assembly

The pyrotechnic control assembly contains the electrical circuits that provide power and control for all spacecraft-initiated pyrotechnic functions. A functional block diagram is shown in Figure 125. The assembly provides safe-arm relays for control of power to the subsystem, power conversion for rectification of the AC input to DC, energy storage circuits, relay control of the DC input to the individual energy storage circuits, and the individual firing circuits for each bridge wire.

a. Safe-Arm

The function of the safe-arm circuit is to block prime power application to the pyrotechnic subsystem, thereby preventing premature charging of the energy storage circuit capacitors and operation of solid-state devices. This function is accomplished by routing of prime power through the normally open contacts of redundant latching relays. A safe-arm monitor circuit is also incorporated to provide a status indication for the EOSE. The safe-arm commands are generated within the sequencer as a result of planetary vehicle separation from the adapter. Solid state relay drivers provide power for relay actuation.

A second set of arm relays is included between the power conversion circuits and the energy storage circuits (see Figure 125) to remove power during long flight periods when no pyrotechnic events occur. In addition, they provide a capability for isolation of any energy storage circuit should it malfunction. The relays are energized by

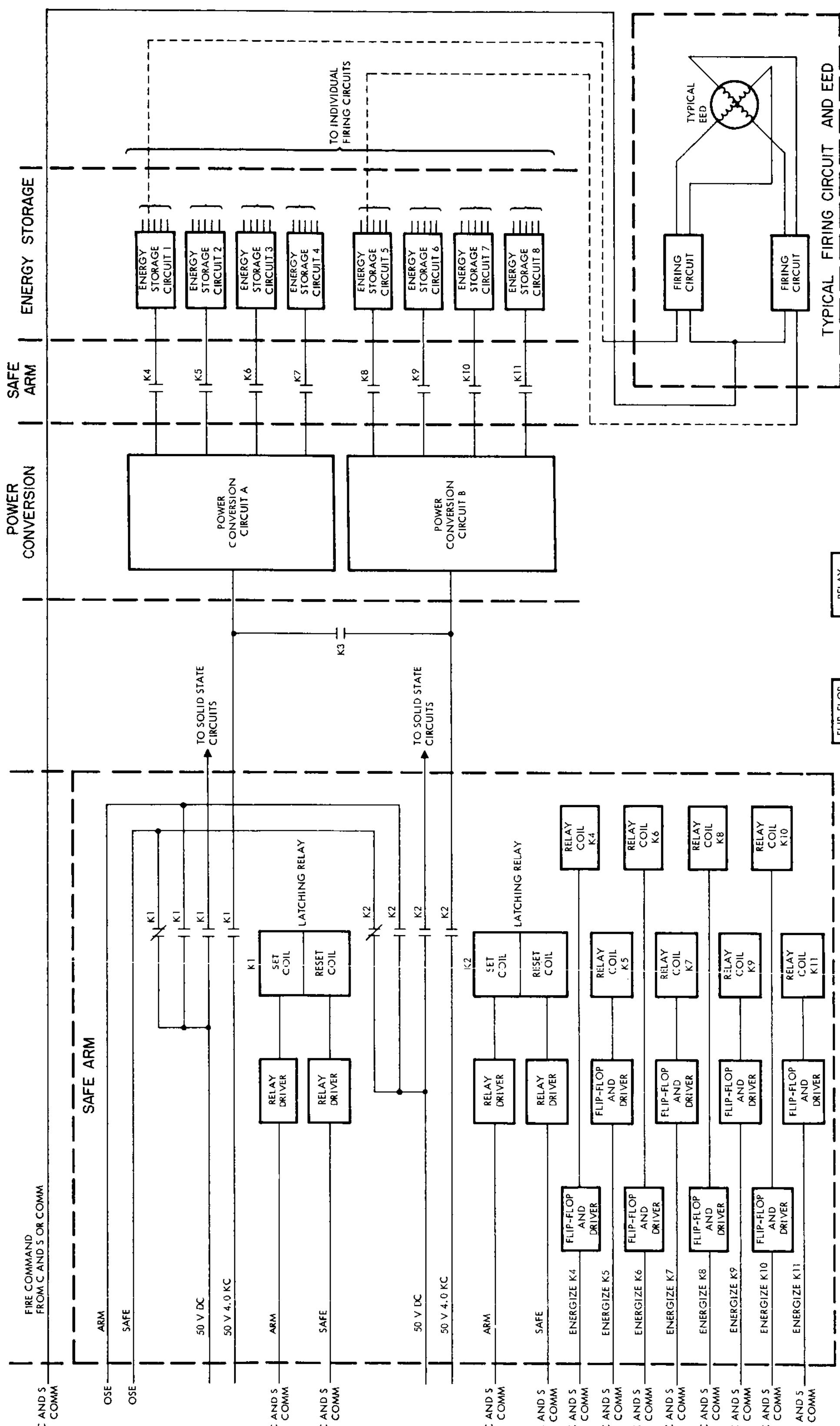


Figure 125. Pyrotechnic Subsystem Block Diagram

individual arm commands from the sequencer and backup commands from the command subsystem. The relays are maintained in an energized condition by a solid-state flip-flop and driver circuit until reset by command from sequencer or command subsystem.

b. Power Conversion

The prime source of power for subsystem operation is the 50-volt 4-kc bus. Since the energy storage cirucits require a DC input, rectification is necessary. The AC input is routed to a transformer which increases the voltage to the desired level. This increase in voltage allows the use of smaller capacitance in the energy storage circuits. The transformer output is then supplied to solid-state rectifiers for conversion to DC.

c. Energy Storage Circuits

The energy storage circuits contain capacitors in which energy can be stored over a comparatively long period of time (on the order of 5 to 10 minutes) prior to the initiation of an explosive device. The desired charge period is controlled by proper sizing of resistors in the circuits. The demand on the AC bus is reduced by extending storage time to the longest time available consistent with mission requirements.

The division of total energy storage capability into eight individual circuits (see Figure 125) is based on an analysis of the number of explosive devices that may require initiating current at any one time. By proper mission sequencing, this number can be limited to a maximum of eight. A number of firing circuits are connected to each energy storage circuit. With eight storage circuits provided, firing circuits can be arranged so that an individual storage circuit can be utilized for each bridge wire being fired at any one time. This approach provides maximum redundancy and serves to isolate a malfunction should one occur.

d. Firing Circuit

The firing circuits control the initiating current to the bridge wires. Solid-state switching is utilized and each firing circuit contains a current-limiting resistor. Should actuation result in a short circuit, this resistor would open, preventing excessive current drain from the source. Redundancy is provided by utilizing a separate firing circuit for each bridge wire. Initiating current controlled by each firing circuit is supplied from separate energy-storage circuits.

The level of redundancy in command circuits, energy storage circuits, and firing circuits will be reviewed during Phase IB to assure optimum subsystem performance.

10.4.2 Electro-Explosive Devices

Every design feature necessary to satisfy the requirements of the explosive devices has been incorporated in one device or another; however, no single shelf item exists which embodies all of the features. The Apollo Standard Initiator (ASI) comes the closest in that it meets all of the requirements except the required need to withstand a 25,000-volt static discharge between pin pairs. It does withstand 25,000 volts pin-to-pin and shunted pins to case. Means for meeting the pin-pair requirement will be defined in Phase IB.

The connector normally used with the ASI makes contact with the case before making contact with the pins by a margin of 0.01 inch. The desirability of increasing this margin will be studied.

The closeness of fit between connector and case will provide adequate protection against electrostatic discharge, providing the connector is conductive and not coated with an insulating material as standard ones are.

Material normally used in the manufacture of the ASI is 700-Series stainless steel. Prototypes have been made of 300-Series stainless steel. Nonmagnetic cases of aluminum will not withstand the ASI pressure test requirement but will easily meet the requirements of the Voyager specification.

10.4.3 Mechanical Attach-Release Devices

The mechanical devices are described in detail as a part of the mechanism subsystem (Section 9). The types of devices used in the subsystem are reviewed here, with the functions performed by each type identified.

a. Electrical Disconnects

The electrical disconnects are dual-cylinder piston devices which rotate the two elements of a standard twist lock electrical connector in such a manner as to unlock it and allow the elements to be separated. The energy for unlocking is supplied by redundant explosive devices. Two disconnects identical except for size will be used, planetary vehicle-vehicle adapter electrical connection and flight spacecraft-flight capsule electrical connection.

b. Release Nuts

The release nut is the basic active element in all of the separation mechanisms. Structural elements to be separated are restrained by bolts and split nuts at the attach points. The split nut is held together by a rod, threaded at both ends, to which release nuts are attached. The pyrotechnic actuation of either release nut allows the split nut to spread. The bolt, under spring pressure, is then withdrawn from the nut. Metal covers prevent the release components from flying free. The release nut is utilized for a variety of functions associated with the spacecraft:

- Planetary vehicle separation from the launch vehicle
- Emergency flight capsule jettison
- Release of high- and medium-gain antennas
- Deployment of low-gain antenna and PSP

The nuts are identical except for size in these applications.

c. Pin Pullers

The dual pin puller is a device containing two pistons on a single shaft located in individual cylinders within the device body. The shaft has sufficient freedom of movement to protrude from the device body, serving as a pin which can be used as a restraining device. Pressure from the explosive device is admitted to the cylinders, causing the pistons to draw the shaft into the device body, releasing the restrained elements. The only mechanism using a pin puller is the release of the PSP gimbal axis.

10.4.4 Explosive Valves

Explosive valves are utilized in the propulsion subsystem for the valve openings and closing to control propulsion operation. The valves are part of the propulsion subsystem and are described in detail in Section III. The explosive devices used in valve actuation are part of the pyrotechnic subsystem.

10.4.5 EMC Design Considerations

The pyrotechnic subsystem will be designed to comply with the AMR range safety document and, more importantly, the subsystem will be designed to be completely compatible with the Voyager system. To achieve this goal, the subsystem will comply with the Voyager electromagnetic compatibility requirements as detailed in the preliminary EMC control plan (see Volume 3).

In general, the pyrotechnic control assembly will be designed to electromagnetically isolate the explosive devices and firing circuits from the remaining subsystems. The unit enclosure will be constructed with solid metal skin providing a shielding integrity in excess of the EMC requirements. All cabling, both power and signal, except the shielded cables for the explosive devices, will be decoupled immediately upon entering the unit enclosure. The internal capacitor charging and firing circuits will be designed to minimize interaction between firing circuits by minimizing common impedance elements and wiring crosstalk. This type of command activation circuit is utilized to reduce the possibility of coupling enough energy from other spacecraft operations into the firing circuit to false trigger the firing circuit.

10.5 Parameters and Performance Summary

The parameters and performance summary of the pyrotechnics subsystem are shown in Table 68, and the preliminary specifications are shown in Figure 126.

Table 68. Pyrotechnics Subsystem Parameters and Performance Summary

Subsystem Function	Provide circuitry, electroexplosive devices and mechanisms necessary to accomplish flight spacecraft associated pyrotechnic functions
Characteristics	
Input Voltage:	50 volt 4.0Kc 5 watts 50 volt regulated — 0.5 watts
Input Commands:	10 volt 50 millisecond pulses
Number of Command Inputs:	27 from C and S 23 from command subsystem
Number of Electroexplosive Devices	53 total
Type of Electroexplosive Devices	Pressure producing • 2400 psi • 4200 psi
Initiating Current Source	Capacitor discharge
Type of Firing Circuit	Solid state
Type of Mechanisms	Release nuts, pin pullers, electrical disconnects, explosive valves
Physical Characteristics	
Weight (lbs)	
Release and deployment system	7.7
Flight capsule emergency separation	13.7
Pyrotechnic control assembly	25.0
Explosive devices	0.6
Electrical connectors	2.2
Attachments and miscellaneous	1.5
Total	50.7
Size	
Pyrotechnic Control Assembly	7 x 7 x 20 inch
Reliability (including pyrotechnic)	.97733 (encounter plus 1 month)

PRELIMINARY SPECIFICATION

Pyrotechnic Subsystem Elements

Electroexplosive Devices

Function

Produce pressure and retain in pyrotechnique devices.

Performance Characteristics

Pressure seal good to 8000 psi (nitrogen gas) after firing.

Hermetic seal-leakage rate not to exceed 10^{-6} cc/sec (helium) at one atmosphere differential pressure.

Recommended firing current: 5 amps applied to one bridgewire over a range of -250 to 165°F results in functioning time of 4 ms or less.

No-fire current: 1 amp - 1 watt for 5 minutes simultaneously on two bridgewires.

Minimum firing current is 3.5 amps to one or both bridgewires.

Output pressure as specified 2400 or 4200 psi in a 10 cc bomb.

Insulation resistance:

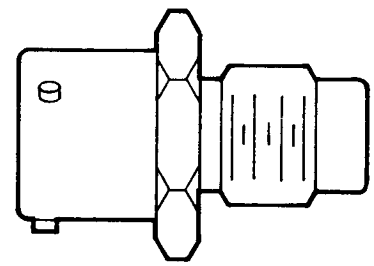
100 megohms minimum resistance when 100 vdc is applied between shorted contact pins and case.

Static sensitivity:

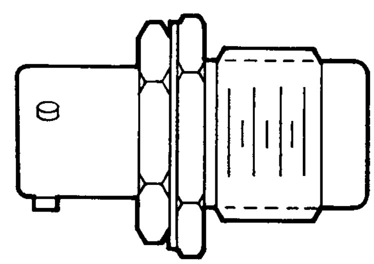
Withstand static discharge of 2500 volts from a 500 picofarad capacitor applied pin-to-pin or between pins and case at all atmospheric pressures.

Environment

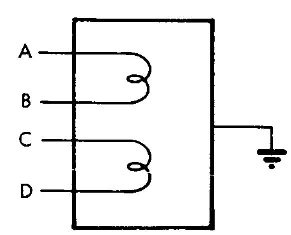
Dielectric strength withstand 500 vac (rms) applied between shorted pins and case for one minute without breakdown.



EED PRESSURE TYPE
LOW PRESSURE OUTPUT



EED PRESSURE TYPE
HIGH PRESSURE TYPE



CIRCUIT

Physical Characteristics

Material of body to have as low a magnetic signature as is consistent with performance requirements above.

Size: 2400 psi units, 3/8 - 24 UNF x 1 inch long, with 9/16 inch hex
4200 psi units, 5/8 - 18 UNF x 1.2 inch long, with 7/8 inch hex.

Weight: Under 0.05 pound

Initiator: Head identical on both units

Connector to make contact with case before making contact with pins.

Figure 126. Preliminary Specification—Pyrotechnic Subsystem Elements

11. TEMPERATURE CONTROL SUBSYSTEM

11.1 General Description

The temperature control subsystem ensures that spacecraft components will experience satisfactory temperatures throughout the Voyager mission. The design features include a) surface finishes to attain desired radiometric properites, particularly on external equipment, b) appropriate distribution of electronic components, and c) structural design to achieve various degrees of thermal coupling, generally close coupling within the main compartment and poor coupling between the main compartment and solar array and capsule, and between the external equipment and the solar array backup structure. Temperature control hardware to be used includes multilayer aluminized Mylar insulation, bimetal-actuated louver assemblies, and thermostatically-controlled heaters.

Temperature control of the main compartment is acheived by insulation on the external surface, outriggers, and capsule adapter, and by the use of louvers covering the uninsulated radiating areas on the equipment panels. The louvers serve to regulate the temperature of the electronic equipment, while the insulation serves to reduce the temperature gradients within the compartment, thereby providing a proper thermal environment for the remaining internal equipment.

Insulation covering the back side of the solar array reduces the cooling rate during eclipses, thereby permitting survival of longer eclipses. Appropriate radiometric properties of the front side limit maximum array temperatures to acceptable values.

External equipment such as antenna drive motors will use a combination of insulation and thermostatically controlled heaters for temperature control. Much of the external equipment will be passively controlled using appropriate surface finishes alone, or in combination with insulation.

11.2 Requirements and Design Constraints

This section is a summary of requirements and design constraints imposed by the mission specification and by the selected spacecraft configuration which have particular relevance to the temperature control subsystem. The summary, divided into three subsections, gives 1) the conditions (mission constraints) under which the subsystem is to perform, 2) the performance requirements the subsystem is to meet, and 3) the design requirements which must be satisfied. Collectively, these are taken as ground rules to be observed in the development of this subsystem.

11.2.1 Mission Constraints

- Sterilization: insulation and solar array will be exposed to elevated temperatures (≈250°F, 50 hours). When encapsulated in the fairing the spacecraft will be exposed to a 100°F gaseous environment (12 percent ethylene oxide, 88 percent Freon) for about 10 hours. This will be followed by a nitrogen purge.

- Air Conditioning: cooling of the spacecraft, when encapsulated in the fairing, will be accomplished by a flow of air (45 to 65°F) within the fairing. Flow rate, humidity and particulate matter content to be specified.

- Mechanical Loads: louver assemblies and insulation will experience acceleration and mechanical and acoustic vibration during boost.*

- Shroud Venting: insulation will experience a decreasing pressure environment.*

- Heating: shroud heating during boost will affect, mainly, external surfaces of insulation and planetary vehicle separation mechanism. Aerodynamic heating after nose fairing jettison will affect, mainly, the insulation radially outboard of the flight capsule. Heating limits are to be specified. Prior to injection the environment will include a partial fairing, a parking orbit coast period (2 to 90 minutes) during which earth albedo, infrared emission, and eclipse will be experienced.

- Orientation: the planetary vehicles will be attitude stabilized, using the sun and Canopus as reference objects, except for periods associated with earth eclipse; initial stabilization; maneuvers for midcourse corrections, retropulsion firing, Mars orbit trim and capsule orientation; and Mars eclipses. A Mars eclipse of 2.3 hours represents the longest eclipse anticipated.

- Environment: solar thermal radiation, corpuscular radiation, micrometeoroid distribution, and Mars albedo and infrared emission.*

*As given in JPL "Voyager Environmental Predictions Document," 18 October 1965.

- Other: nominally, the capsule will be separated from the spacecraft, at the in-flight separation joint, in the vicinity of Mars. Non-nominal operations include never separating the capsule, and separation at the field joint. Satisfactory subsystem performance shall extend for a period of two months after Mars encounter, with a design goal of 6 months.

11.2.2 Subsystem Performance Requirements

It is required that the temperature control subsystem maintain all spacecraft components in a temperature band which served as the basis for thermal environmental exposure during type approval and flight acceptanc testing.

11.2.3 Design Requirements

In developing the temperature control subsystem to satisfy the performance requirements, a number of design requirements will have to be satisfied. These are grouped, somewhate arbitrarily, into two categories. The first, reliability, includes those requirements which clearly pertain to acquiring the greatest confidence in the successful performance of the subsystem; the second, discipline, includes specific requirements which reflect good design practice.

a) Reliability

- Adequate design margin to allow for irreducible uncertainties such as: 1) material properties measurements, 2) effect on material properties of prolonged exposure to the mission environment, and 3) component heat generation history.
- Systematic identification and elimination of unreliable design features.
- Keep design simple to enhance validity of analyses and tests, and to ease fabrication, assembly, inspection, and checkout.
- Maximize use of techniques which have proved to be successful on other spacecraft programs.

b) Discipline

- Design to accommodate the selected science payload.
- Replacement of thermal hardware to be direct without requiring special adjustments or calibrations.

- Thermal paths between components and structure to be compatible with electrical grounding requirements.
- Particular attention to be given to effects of heat-up from the propulsion subsystem.
- Satisfy spacecraft requirements regarding RF and magnetics control.
- Design to be compatible with spacecraft decontamination procedures.

11.3 Functional Interfaces

11.3.1 Electrical

The baseplate of each component, including those of the science subsystem, must have an area large enough to limit maximum power dissipation densities to a nominal value of 0.3 watt per square inch (of baseplate area). In instances where this requirement may create significant penalties, for example in the case of the 50 watt TWT and charge regulators, power densities up to 1.4 watts per square inch will be permissible; this higher density, however, will be associated with higher operating temperatures of approximately 185°F and 150°F, respectively. For these higher power dissipating components special baseplate designs (thicker and longer) will be required.

The flatness requirement of mounting surfaces is approximately 0.004 inch per foot. Thermal filler material will only be used if compelling reasons exist.

Packages mounted to the inside surface of equipment panels will have high emissivity ($\approx$ 0.85) finishes on all surfaces; point or anodic treatment generally will suffice. Surface finish constraints necessary for adequate electrical grounding will be specified.

Permissible operating and nonoperating temperature limits will be as given in Table 5.5-1 of Section IV, Volume 1.

For the purpose of maintaining satisfactory temperature control of the equipment (exclusive of science) auxiliary heater power (at 50 VDC, nominally) will be required. For this purpose the power subsystem will allot 54 watts prior to Mars encounter and 34 watts thereafter. In addition, the array power margin shown in Figure 17 of Section 1 will also be made available for this purpose.

Provisions will be made for ground-commanding ON or OFF as many as four separate heater circuits.

Provision will be made for as many as 40 temperature sensors (type, ranges, and locations to be specified) for monitoring equipment mounting panel, solar array, appendage articulation mechanisms, and structural elements. These sensors are in addition to those required for monitoring temperatures of other subsystem components.

11.3.2 Mechanical

The design of the spacecraft elements will conveniently accommodate the installation of insulation assemblies.

Equipment panels will have a transverse thermal conductance of at least 7 Btu/hr ft^2 $^{\circ}$F, and a lateral thermal conductivity-thickness product of at least 6 Btu in/hr ft $^{\circ}$F. Provisions will also be made to permit installation of a louver assembly on the outside surface. The external surface (at least that portion covered by the louver assembly) will be treated to obtain a value of solar absorptivity no greater than 0.25, and a hemispheric infrared emissivity of at least 0.85. The surface flatness requirement is approximately 0.004 inch per foot.

Mechanical joints, fittings, etc. between the main compartment (basically, the propulsion module structure) and adjoining external equipment will be designed to impede heat flow (see Section 11.4.1). All joints between the structure and associated panels will be designed to enhance heat flow.

Many different surface finishes are likely to be required. All surfaces internal to the main compartment will be treated to have hemispheric infrared emittances of at least 0.85, although exceptions to this may be required. External exposed surfaces will have varied finish requirements.

To prevent adverse heating of the solar cells and neighboring inboard equipment, thermal radiation from the nozzle and nozzle extension will be limited by the use, respectively, of high temperature insulation secured to the engine compartment bottom-frame (see Figure 129 in Section 11.4.1) and by the use of an ablatively cooled nozzle extension.

Non-sun oriented periods which occur as a result of midcourse, retropulsion, orbit trim and capsule orientation maneuvers will be limited, nominally, to 2 hours.

11.3.3 Capsule System

The capsule system is thermally coupled to the spacecraft essentially by conduction alone, through the capsule interstage structure. Radiation coupling with the external equipment, the louvered radiative areas, and the insulated main compartment and solar array is small. The design of the capsule interstage structure will be such as to minimize the heat loss from the spacecraft to the capsule.

11.4 Design Description

11.4.1 Spacecraft Assemblies

To describe the design of this subsystem it is convenient to consider, separately, the different principal assemblies of the spacecraft. These, shown schematically in Figure 127 are the main compartment, solar array, engine, and external equipment. Although these are considered separately, it must be recognized that thermal interactions exist.

a. Main Compartment

The main compartment houses both electronic and mechanical components. The electronic components, grouped by subsystems, are mounted on equipment panels located as shown in Figure 128. Mechanical components, the majority of which are associated with the propulsion subsystem and portions of the guidance and control subsystem, are mounted to the structure. Temperature control of the contents of the compartment is achieved principally by:

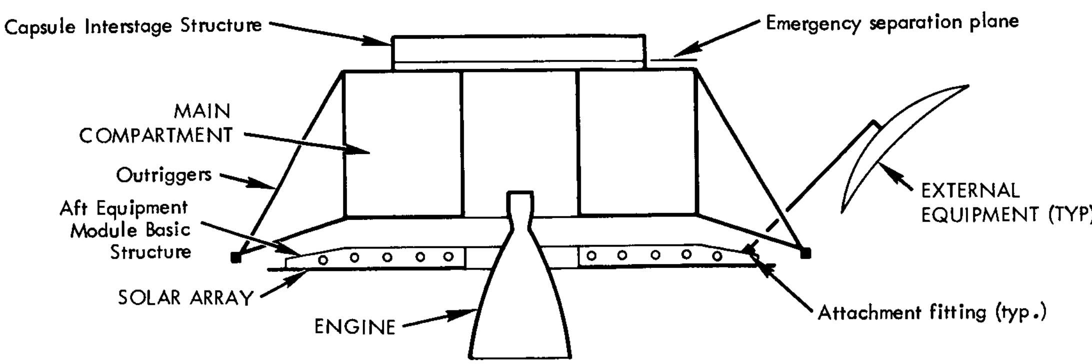


Figure 127. Principal Thermal Assemblies of the Spacecraft

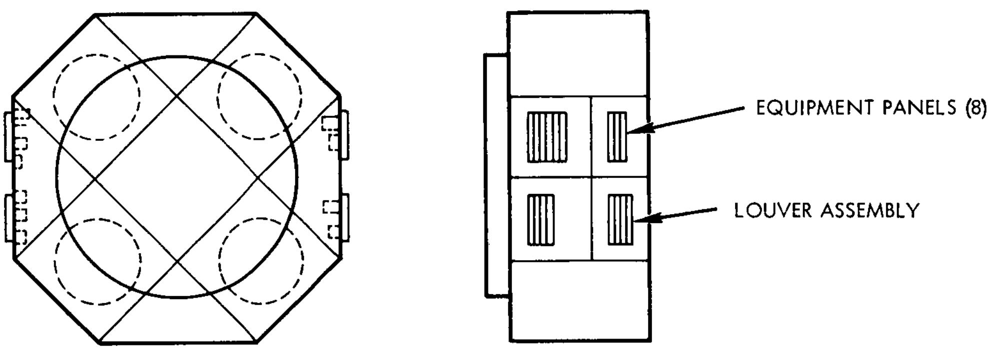


Figure 128. Equipment Panel Locations

- Virtually complete insulation on external surfaces.
- Minimizing heat shorts (thermal coupling) to the capsule, aft equipment module, and through miscellaneous external fittings to space.
- Appropriate distribution of internal heat generating components.
- Appropriate radiant and conductive interchange within the enclosure.
- Use of thermal louvers mounted on a portion (radiating area) of the outer surface of the equipment panels.

External insulation, shown schematically in Figure 129 will cover virtually all external surfaces except the louver-covered radiators. The prime function of the insulation for the Voyager application is to minimize

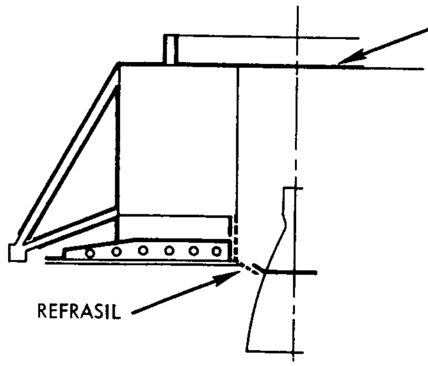


Figure 129. Insulation Locations

heat loss to space; a secondary function is to limit heat gains when irradiated by the sun. Because the heat sources are localized rather than distributed, a high performance insulation is required to prevent remotely located passive equipment from getting too cold. The insulation to be used, described more fully in Section 11.4.2, is of the multilayer radiation shield (aluminized Mylar) type. The covering on the aft surface of the engine compartment will be a high temperature insulation. Its purpose, in addition to minimizing heat loss from the main compartment, is to prevent adverse heating of the inboard area of the solar array subsequent to engine firing.

Local heat shorts (heat paths) bridge the main compartment to adjoining equipment and to space. Depending on the circumstances, these paths afford an opportunity for the compartment to gain or lose heat. Thermal design of these local heat shorts will minimize this heat exchange. Figure 130 shows, schematically, the heat short design associated with the capsule adapter.

Location of the bulk of the mechanical components (propulsion subsystem) is fixed by virtue of the modularized concept adopted; i.e., the propulsion subsystem is virtually integral with the main compartment structure. The electronic components will be grouped by subsystem and mounted on equipment panels. Within the constraints of center-of-gravity requirements and subsystem functional requirements, the distribution of the components will be such as to divide the heat generation nearly equally between opposing panel groups to reduce temperature gradients within the structure. On each of the equipment panels the components will be arranged to achieve approximately uniform heat distribution.

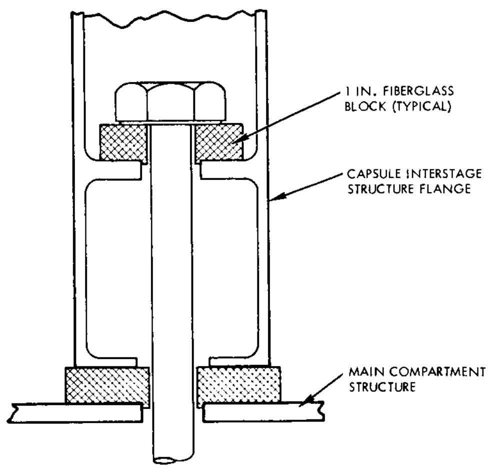


Figure 130. Typical Heat Short Design

Figure 131 illustrates a typical component mounting technique. Component design constraints, such as flatness and power dissipated per unit base area, generally will preclude the need for thermal filler material between component and mounting surface; only under compelling circumstances will the use of such filler be considered. Special packaging and mounting techniques are required for very high heat generating components, the notable example being the TWT. Mounting techniques necessary to satisfy electrical grounding of the component case to the panel are compatible with the thermal requirements.

By way of complementing the external insulation, internal temperature gradients will be minimized by enhancing conductive and radiative heat transfer within the compartment. Thus, high thermal conductivity aluminum alloys will be used on external panels and, where advantageous, internal surfaces will be treated to attain a high emittance.

As a result of the preliminary study, approximately 18 square feet of louver-covered radiating area is required on the main compartment. Louver-assemblies (see Section 11.4.3) of the type used on OGO, Pioneer, and Mariner, will be mounted to the external surface of equipment panels. Their main purpose is to limit the temperature excursion of electronic components by regulating the heat loss locally.

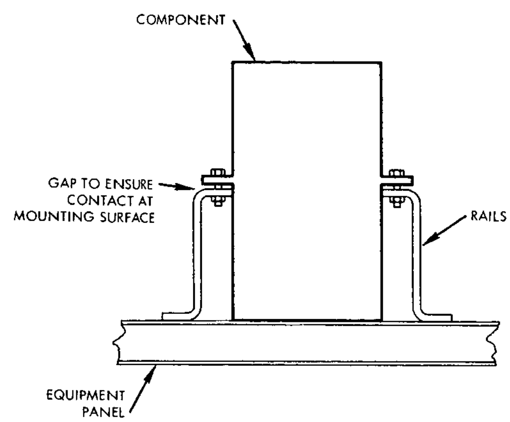


Figure 131. Typical Component Mounting Technique

b. Solar Array

For purposes of this discussion, the solar array includes the solar panels, the aft equipment module basic structure, and the complex of attachment points which serve to secure the external equipment. Effort during this study was limited to determining the gross aspects of the thermal design which would be necessary to limit solar panel temperature extremes to acceptable values, taken to be -120°C and +120°C. These are the operating temperature limits for RCA's flight-qualified design of the solar modules for LOP. Thermal behavior characteristics of the solar array are presented in Section 11.5.3.

The lowest array temperature during the Voyager mission will be at the end of the longest Mars eclipse. To accommodate a maximum eclipse of about 2.3 hours, using a reasonable weight per unit area, it was found necessary to insulate the back side of the array (see Figure 129). The conductance of this insulation need not be as low as that required for the main compartment because substantially all of the heat lost by the array is emitted from the front surface.

The maximum array temperature will occur when the array is sun-oriented in the vicinity of the earth; its temperature will be nominally at about 107°C by virtue of the radiometric properties of the cell-covered surface. During the first midcourse correction the very low intensity of thermal radiation from the plume (throttled-down engine) is not likely to cause significantly higher temperatures.

c. Engine

No special thermal design of the engine appears necessary for the Voyager mission, engine temperatures near earth ($\approx 185^{\circ}F$) and Mars ($\approx 70^{\circ}F$) being within tolerable limits. Post-firing soak-back heating of the propulsion subsystem hardware housed within the main compartment was estimated and found to pose no Voyager-peculiar problems. The radiatively cooled skirt used for the Apollo mission, however, will be changed to an ablatively cooled one to prevent excessive heating of the inboard solar panels during engine firing.

d. External Equipment

The external equipment is comprised of the remaining spacecraft components which are separately exposed to the spacecraft and space environment, and which are attached to pads on the aft equipment module basic structure. A preliminary examination of the external configuration revealed that satisfactory temperature control of the fixed components and the drive mechanisms could be achieved by techniques described in the Phase IA Task A report. These techniques, used successfully for external equipment on spacecraft such as OGO, include appropriate conductive coupling to the pads, appropriate surface treatment, insulation and, when necessary, thermostatically controlled heaters. A detailed study in Phase IB would will result in the optimum combination of these techniques.

The thermal effect of plume impingement on the low-gain antenna was studied. This convective heating combined with radiant heating from the sun (at 1AU), the plume, and the arrays produces an antenna temperature of about 150° for reasonable radiometric properties. The contribution of convective heating is negligibly small.

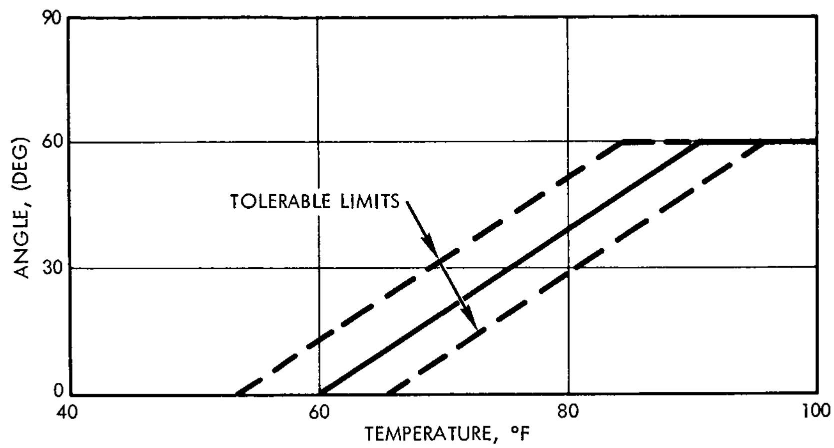
11.4.2 Subsystem Elements

a. Louver Assemblies

The louver assemblies will be of the bimetal actuator type used on Mariner, OGO, and Pioneer. From the three designs those features which best suit the Voyager application will be selected. A preliminary specification for the louver assemblies is given in Figure 132. For

PRELIMINARY SPECIFICATION

Louver Assembly



Function

To prevent excessive excursion of radiator temperature by varying its local effective emittance.

Performance Characteristics

1. The relationship between lower angle and actuator temperature shall be as shown in graph.
2. When mounted to an isothermal radiator (approximately 3 square feet) whose emissivity is 0.85, the effective emissivity of the radiator shall be:

   < 0.10, with louvers fully open

   > 0.68, with louvers fully closed
3. The assembly shall tolerate temperatures resulting from exposure to solar radiation at 1 AU with a radiator solar absorpitivity of 0.20.
4. The performance shall not be compromised by exposure to the mission environment for 1.5 years.

Physical Characteristics

1. The assembly weight, for areas greater than 3 square feet, shall not be greater than 0.5 pound/foot^2.

Figure 132. Preliminary Specification—Louver Assembly

the purpose of general description, however, the essence of the OGO configuration (see Figures 133 and 134) is given below.

Each assembly will include a number of louver blades, each being about 2 inches wide, made from two pieces of 0.005-inch aluminum suitably shaped to provide adequate strength. A bearing pin is attached on the central longitudinal axis at each end of the louver blade through interposed insulation blocks.

Continuous louver support brackets formed from small gage aluminum contain the louver bearings. Integral features serve to secure one end of each bimetal actuator and to limit louver angle excursions (0 to 90 degrees). A pair of brackets, joined together at their ends, form a frame containing the louvers. The final louver assembly (less the actuator shields) thus becomes one part for which handling and installation fixtures are built.

Actuators for the louver blades will employ alloy S992 (W.M. Chase Spring Co.), the type used on the Pioneer spacecraft for its low magnetic field intensity (3 gamma at 3 inches). Each actuator is a spiral coil, the inner end of which is secured to the louver axis by a device which also permits setting the louver angle. The other end is formed to extend down to the base of the support bracket where it is held fixed. Actuators are placed at alternate ends of neighboring louvers.

To permit the actuator to be strongly responsive to the local temperature of the radiator surface, a radiation shield is provided. It is comprised of multilayer aluminized Mylar, formed to cover the actuators and the entire length of the support brackets. When installed, the only openings present are those required to permit free rotation of the louver shaft and blade.

b. Insulation

Insulation used on the spacecraft will consist principally of blankets of multilayer aluminized Mylar. To ease its handling, each blanket will have 3-mil aluminized Mylar covering both faces; the outside layer has the non-aluminized surface facing outward. These blankets can be made of either crinkled sheets (NRC-2, made by National Research Corporation) or of alternate layers of flat and dimpled sheets (Dimplar,

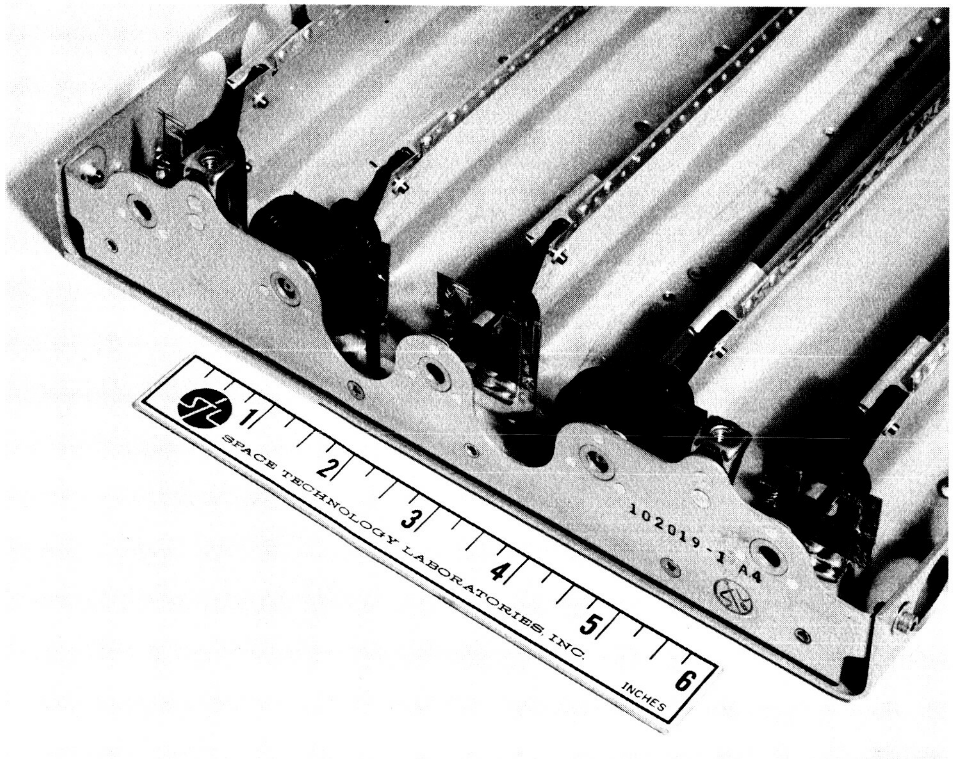


Figure 133. OGO Louver Assembly Details

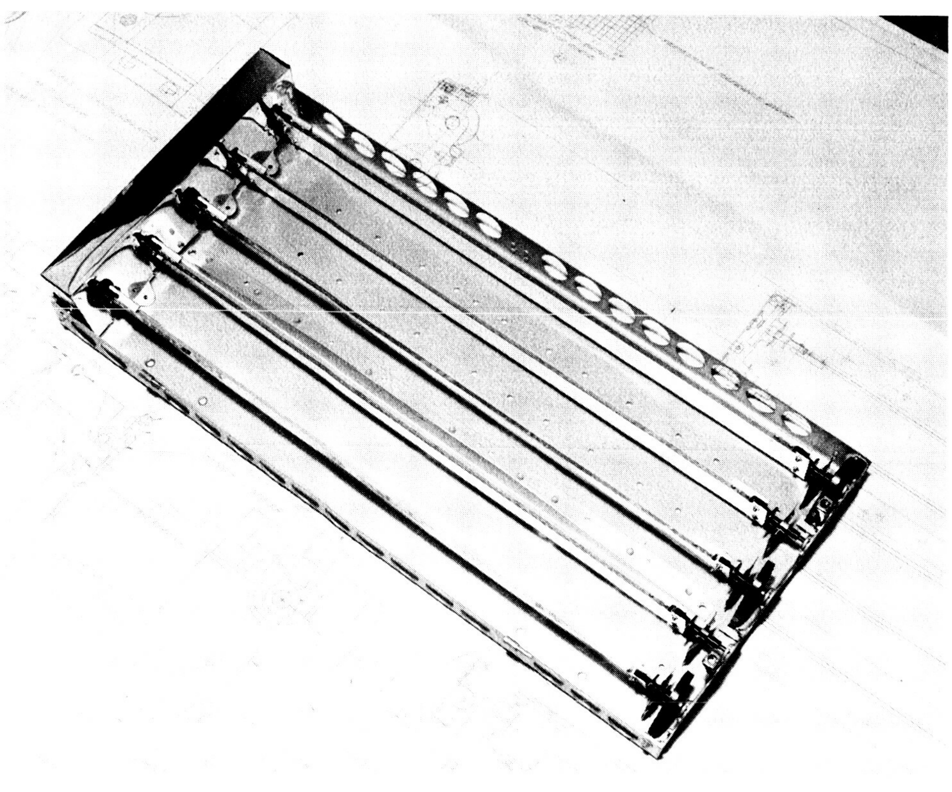


Figure 134. Typical OGO Louver Assembly

made by Quality Electric Company). Either system can be made to yield the required effective thermal conductance (0.001 Btu/hr ft^2 ^oF) for the main compartment if a sufficient number of layers are properly assembled. The NRC-2 system will require about 35 to 50 sheets and the Dimplar system about 15 to 20, both resulting in a blanket of about 1-inch nominal thickness (see Figure 147). Insulation requirements for the solar array will be satisfied with a blanket thickness of about 1/2 inch. An effort during Phase IB will be directed toward selection of the system.

The high-temperature insulation for the aft end of the engine compartment will be 1/2 inch Refrasil batt sandwiched between a spun-metal enclosure of low emissivity.

c. Heaters and Thermostats

The heaters to be used for temperature control of external equipment will be of the strip type. These are thin, variably sized, flexible (e.g., silicone rubber) units which can be adhesively-bonded to a surface. The resistive wire is bifilar-wound to reduce the magnetic field. These units can be obtained from commercial sources in virtually and physical size and power rating. Associated with probably all of these strip heaters will be thermostats having an ON-OFF range appropriately selected. Commercial units are available in a broad spectrum of operating points. TRW Specification PT2-2004, for example, identifies ON-OFF differentials ranging from 9 to 20^oF, setting accuracies of ±2 and ±5^oF, and mean operation levels from 30 to 110^oF.

11.5 Parameters and Performance Summary

11.5.1 Main Compartment

A gross heat balance on the main compartment was made for appropriately conservative assumptions regarding "hot" and "cold" conditions. In such a balance the compartment is regarded to be isothermal. The assumptions included internal power dissipation, and external heat losses and gains. The heat balance for the hot condition served to determine the approximate radiating area required; the cold condition indicated the approximate insulation effectiveness required. Results of this gross heat balance showed that proper temperature control could be achieved if the compartment were approximately isothermal, as postulated.

To determine what the actual temperature distribution in the main compartment is likely to be, a mathematical, multi-node model was generated for computer solution. Figure 135 shows the general configuration of the compartment as well as the solar array and engine, which were represented as nodes. Thermal symmetry permitted using a quarter-section of the configuration, as shown in Figure 136. This section was divided into the (isothermal) nodes, identified in Table 69. This table together with Figure 135 serve to show exact nodal locations. Heat transfer paths (resistances) between nodes are shown in Figure 137. Values of resistance were calculated from available geometric information and from postulates given below:

Conduction

1) External panels: 0.05 inch thick 7075 aluminum sheet. (A sandwich-type construction actually exists.)

2) Internal panels: 0.03 inch thick 7075 aluminum sheet.

3) Structure joints: conductance is very large compared with that across panels.

4) Solar panel to main compartment: conductance through all attachments is 0.27 Btu/hr ^{o}F.

5) Equipment panel (radiator): transverse conductance is 7 Btu/hr ft^2 ^{o}F; lateral conductivity-thickness product is 6 Btu in/hr ft ^{o}F.

6) Electronic components to equipment panel: Mounting conductance is 5 Btu/hr ft^2.

7) Insulation: conductance values used were 0.001, 0.003, and 0.005 Btu/hr ft^2 ^{o}F.

Heat Inputs

1) Solar vector: directed into engine nozzle.

2) Solar intensity: 433 Btu/hr ft^2 (earth) and 159 Btu/hr ft^2 (Mars).

3) Outriggers: a total loss of 32 Btu/hr (earth) and 44 Btu/hr (Mars).

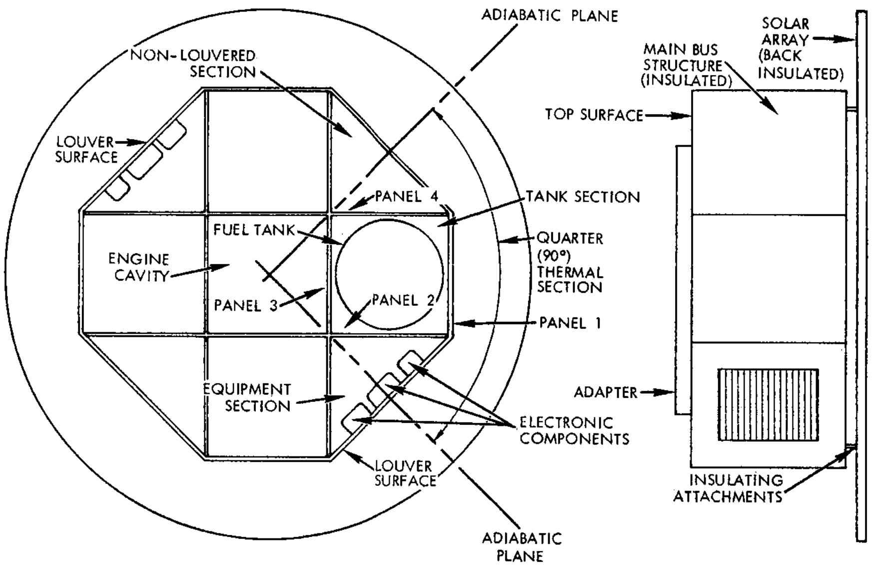


Figure 135. Mathmatical Model Node Designation

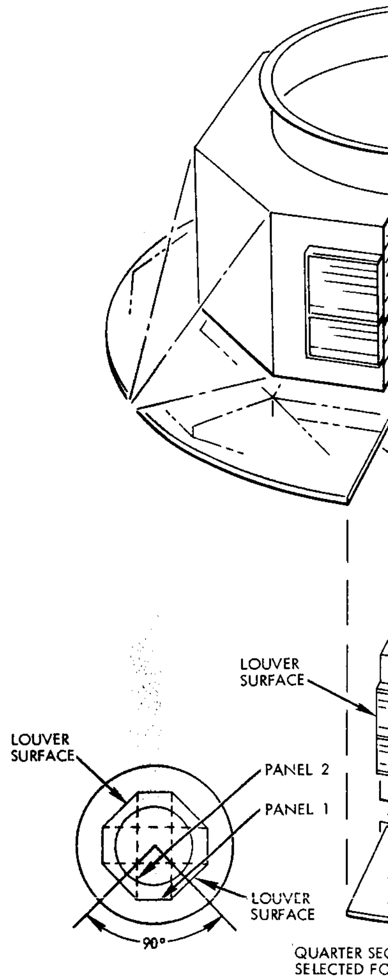
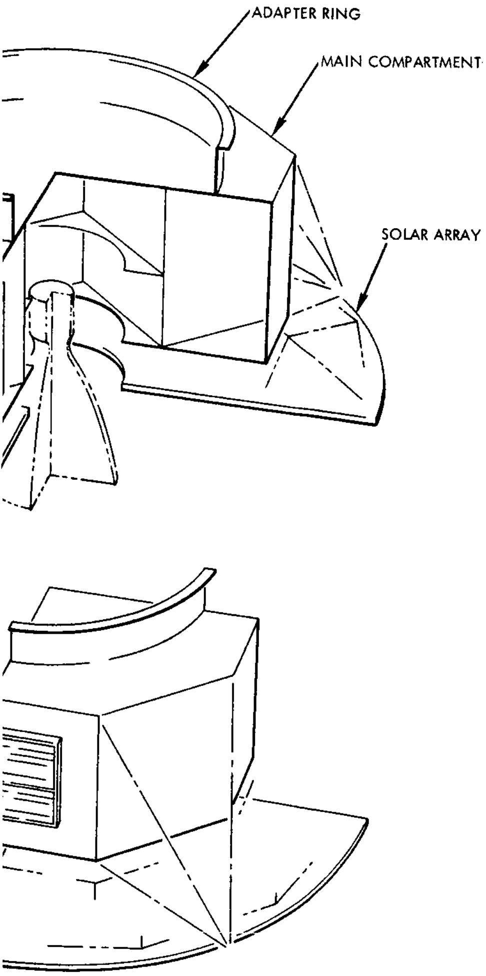


Figure 136.



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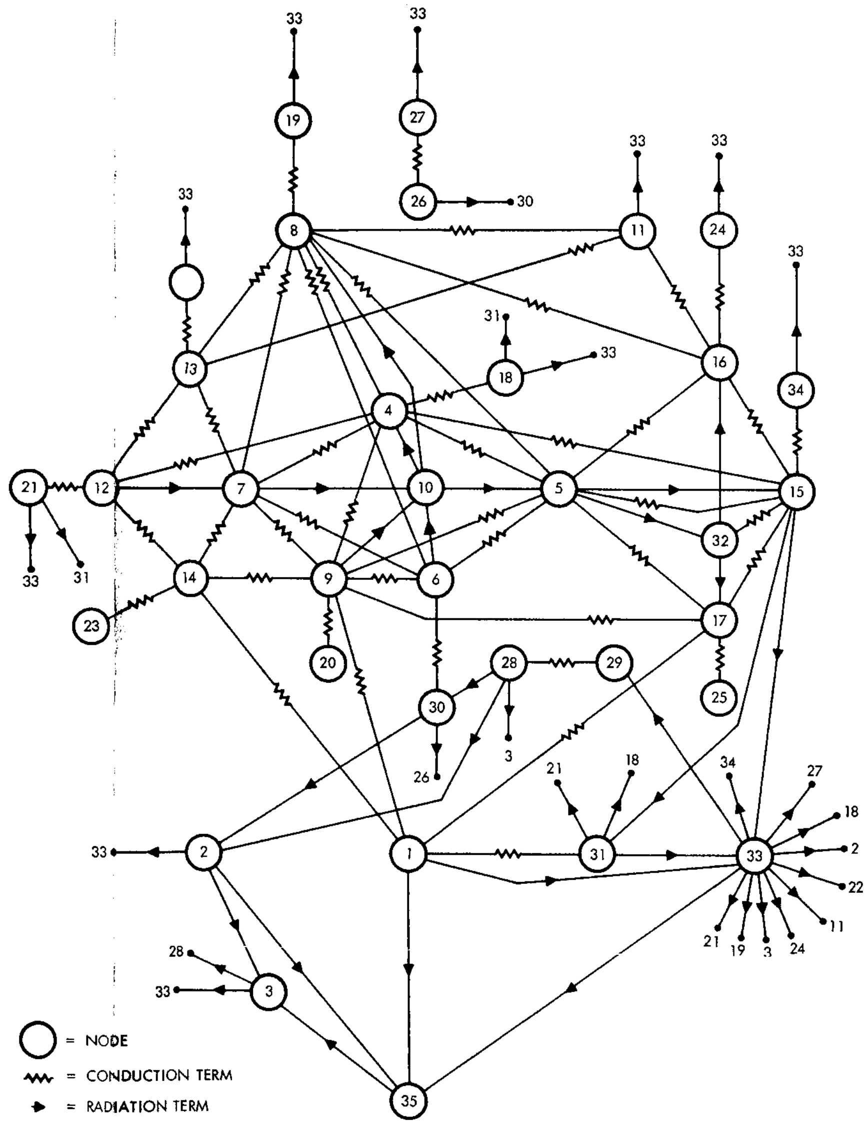


Figure 137. Nodal Network

Table 69. Node Listing

Node Number	Description
1	Solar Array
2	Engine Case
3	Engine Nozzle
4	Tank Section Panel 1
5	Tank Section Panel 2
6	Tank Section Panel 3
7	Tank Section Panel 4
8	Tank Section Top
9	Tank Section Bottom
10	Fuel Tank
11	Capsule Adapter
12	Noncontrolled Section—Outside
13	Noncontrolled Section—Top
14	Noncontrolled Section—Bottom
15	Equipment Section—Outside (Equipment Panel)
16	Equipment Section—Top
17	Equipment Section—Bottom
18	Tank Section Panel 1
19	Tank Section Top
20	Tank Section Bottom
21	Noncontrolled Section Outside
22	Noncontrolled Section Top
23	Noncontrolled Section Bottom
24	Equipment Section Top
25	Equipment Section Bottom
26	Engine Cavity Top
27	Engine Cavity Top Outer Insulation Sheet
28	Engine Cavity Bottom Inner Surface
29	Engine Cavity Bottom Outer Surface
30	Fuel Tank Panel 3 Inner Surface
31	Solar Panel Superinsulation, Outer Sheet
32	Lumped Electronics
33	Space
34	Equipment Section, Outer, Outside Sheet
35	Nozzle Extension

4) Capsule adapter: a loss of 24 Btu/hr.

5) Internal electronics: 319 watts (min) and 392 watts (max), equally distributed between two equipment panels, total mounting surface area of 14 ft^2.

Louvered Area

1) Radiating area: 9 ft^2 on each of the two panels.

2) Effective emittance: 0.1 (closed) and 0.7 (open), linear variation with louver angle.

3) Actuation range: closed at 60°F and open at 90°F, linear variation.

Radiation

1) Internal surfaces: $\epsilon = 0.85$

2) Electronics: radiating area equals 14 ft^2.

3) Outer insulation: $\epsilon = 0.78$.

4) Solar panel: $\alpha/\epsilon = 0.9$ and $\epsilon = 0.82$.

A number of steady-state computations were made to determine how the temperature distribution is affected by the internal power dissipation and by the effectiveness of the insulation. Solutions were also obtained for near earth and near Mars conditions, to include the effect of external losses and gains that depend on the distance from the sun.

Results of this parametric study are summarized in Table 70 and in Figures 138 and 139. Table 70 presents for each of the 12 prime cases studied the temperature of the nodes regarded to be significant, i.e., the radiator panel and the fuel tank; the temperature of the solar array, for reference purposes; and the maximum gradient (ΔT) in the structure (given also for horizontal and vertical planes within the structure). Figures 138 and 139 show the temperature distribution for two of the cases studied. The "hot" case corresponds to good insulation (0.001), high power, near earth; the "cold" case to good insulation, low power, near Mars.

This preliminary analysis indicates that the insulation required for the main compartment should have a thermal conductance of about 0.001 Btu/hr ft^2 °F. The combined effect of louvers and this insulation should

Table 70. Summary of Computed Results

	Near Earth						Near Mars					
Electronic Power, watts	392			319			392			319		
Insulation Conductance, Btu/hr ft^2 ^oF	0.005	0.003	0.001	0.005	0.003	0.001	0.005	0.003	0.001	0.005	0.003	0.001
Temperature (^oF)												
Radiator Panel	72	75	78	68	71	74	65	68	71	60	63	67
Fuel Tank	57	64	75	53	60	71	32	41	51	27	37	47
Solar Array	225	225	225	225	255	225	72	73	72	72	73	73
Maximum Structure ΔT	36	27	22	35	26	21	47	37	28	46	36	27
In horizontal plane	22	15	6	21	15	6	41	32	24	40	31	25
In vertical plane	26	20	17	25	19	17	24	18	14	22	18	13

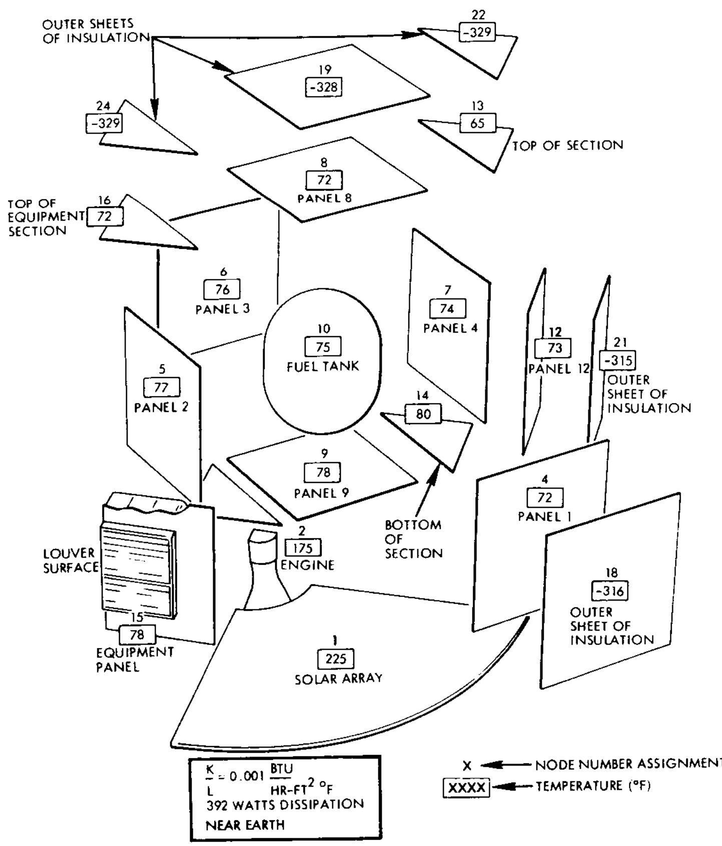


Figure 138. Temperature Distribution "Hot Case"

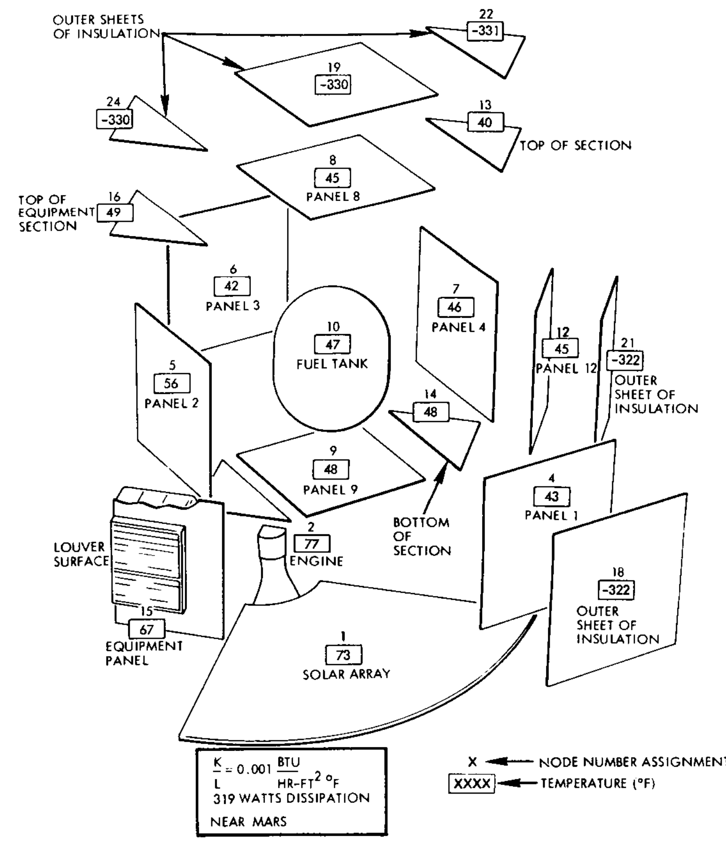


Figure 139. Temperature Distribution "Cold Core"

maintain the equipment panels and remote components such as propellant tanks at acceptable levels.

The computed value of 47°F for the propellant tank ("cold" case), 3°F below the desired lower limit, warrants discussion. Notwithstanding the fact that this temperature depression would in actuality not be regarded as critical, tank temperature histories will be explored more fully as part of the detailed main compartment analysis and design effort. In the event that higher tank temperatures are desired, several thermal design modifications can be made either by the use of local heaters or by radiatively coupling the relatively warm solar array to the main compartment. An analysis has shown that a total of 40 watts of heater power would raise the tank temperatures by about 10°F with a negligible rise in equipment panel temperatures. The spacecraft power margin, for a postulated worst case array capability, would permit the use of at least 60 watts for this purpose through the last orbit trim. Radiative coupling could employ a louver system which would regulate the heat transfer by sensing the temperature of the aft end of the main compartment.

Transient thermal behavior of the main compartment was only superficially examined during this Tank B study. However, from considerations of the large thermal capacity of the compartment, the highly effective insulation, and the studies performed as part of Phase IA, Task A, no significant temperature transients are anticipated.

11.5.2 Plume Heating

The character of the LEMDE plume, as a heat source, was determined in order that its thermal effect on the solar array and other external equipment could be evaluated. Both its radiative and convective characteristics were determined.

A prediction of the radiant heat flux of the plume was made using data from the characteristic net analysis as a basis for a conservative estimate of the plume temperature. The apparent emissivity of the plume was taken to be 0.1. Results, based on the method of analysis described in the Phase IA report, are displayed in Figure 140. The curves in this figure, called "iso-flux" lines, represent the locus of differential areas

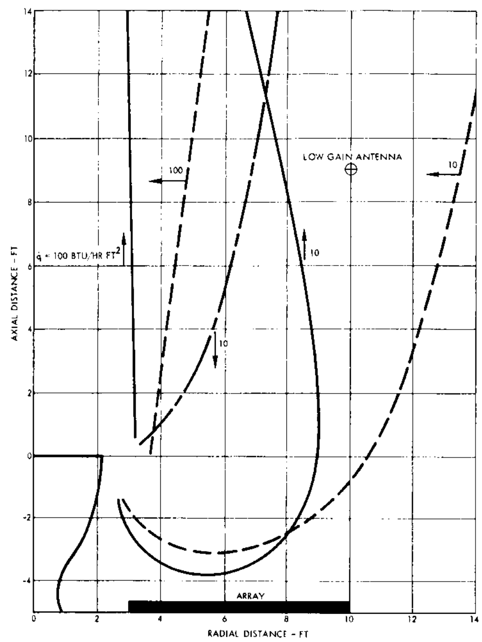


Figure 140. Plume Radiant Heating Characteristics

which receive a particular level of irradiation when facing in the direction given by the arrow, either downstream, upstream, or radially toward the centerline. It is apparent that the incident flux on the aray is small, less than 10 Btu/hr ft^2 and represents no problem.

Flow properties within the LEMDE plume were generated using a characteristic net program. A technique was applied which gives the convective heat transfer independent of the configuration of the heated body. The kinetic energy of the gas stream is equated to heat transfer. It is assumed that all of the energy of the exhaust is transferred to the body independent of body size, shape, or surface inclination, but dependent only on position within the plume. This technique lends itself to a gross examination of the convective thermal environment. The equation for heat transfer becomes:

$$q = \frac{\dot{m}V^2}{2} = \frac{\rho V^3}{2}$$

where

q = heat transfer

$\dot{m}$ = mass flow rate through a unit cross section area

V = gas velocity

ρ = gas density

The heat transfer distribution for 100 psia (high thrust) chamber pressure is shown in Figure 141; at lower pressures heating is approximately proportional to the pressure. While there is no plume impingement on the solar array, there will be impingement on the low-gain antenna, with heating of about 20 Btu/hr ft^2.

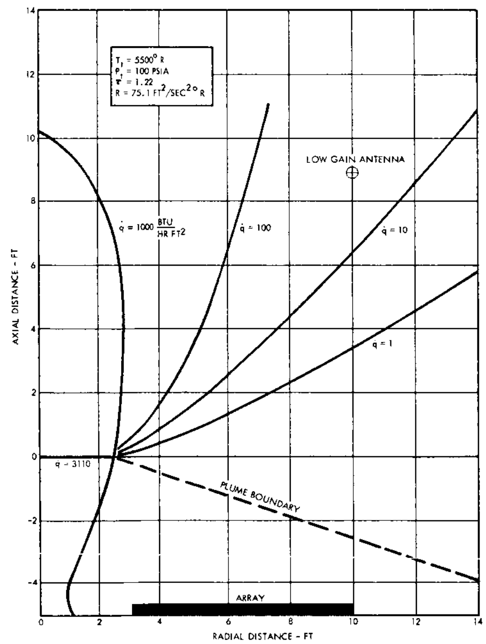


Figure 141. Plume Convective Heating Characteristics

11.5.3 Solar Array

Several thermal analyses of the solar array were performed. All were simplified by assuming the array to be an isothermal object, neglecting the effects and the backup structure, external equipment mounts, and presence of the engine nozzle.

a. Equilibrium Temperatures

Steady state temperatures were determined from the following equation:

$$\sigma T^4 = \frac{(\alpha - \eta)}{(\epsilon + \epsilon_B)} \frac{G}{D^2}$$

where

T = array temperature, ^oR

G = solar intensity at 1 AU (433 Btu/hr ft^2)

D = distance from the sun, AU

σ = Stefan-Boltzmann constant (0.173×10^{-8} Btu/hr ft^2 $^oR^4$)

α = solar absorptivity of front face

ϵ = hemispheric emissivity of front face

ϵ_B = apparent emissivity of back face (negligibly small)

η = solar cell efficiency

Figure 142 shows the equilibrium temperature as a function of distance from the sun. The two curves represent the extreme values of

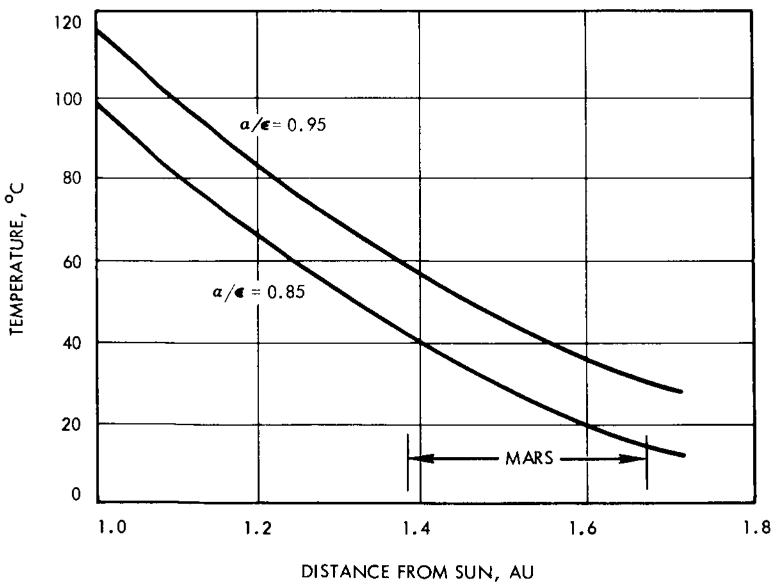


Figure 142. Solar Array Equilibrium Temperature

α/ϵ of the glass-covered solar cells. Because a partial shunt regulator is used, a fraction of the solar array will not convert incident solar radiation to power; this will lower the overall efficiency of the array. For the calculation of temperatures shown in this figure, a cell efficiency of zero was used for the case of α/ϵ = 0.95; an efficiency of 100 percent of the a actual value was used for α/ϵ = 0.35.

b. Transient Temperatures

The major temperature transients which require consideration occur (1) during the earth eclipse which occurs after injection (the fairing limits temperature decay prior to injection), (2) during the retrofire maneuver, and (3) during Mars eclipses. Thus, the transients will generally be of two kinds: cooling due to eclipses, and heating due to sun-reacquisition and/or plume and nozzle thermal radiation.

Figure 143 shows the approximate array temperature decay during eclipses for different array specific weights for starting temperatures

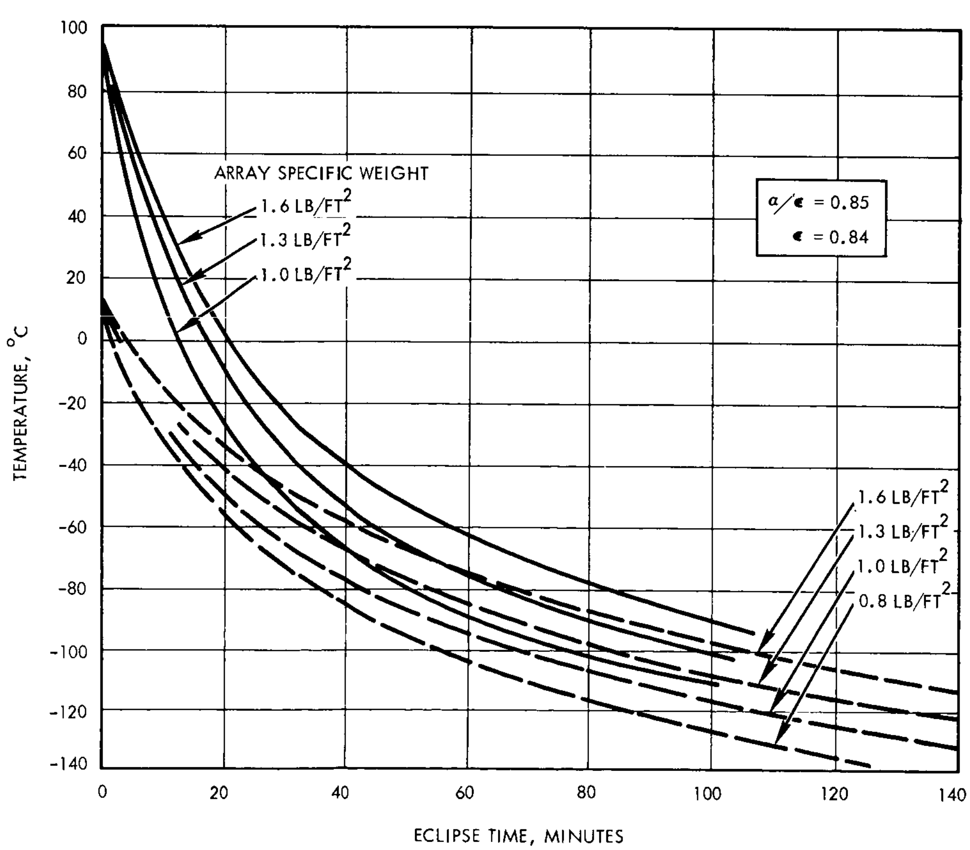


Figure 143. Solar Array Temperature Decay During Eclipses

corresponding to near earth and near Mars. For conservatism the starting temperatures correspond to the lower design value of α/ϵ (0.85) and for the farthest position of Mars (1.67 AU). In addition, the largest design value of emissivity (0.84) was used for the cooling period. For a Mars orbit eclipse of the nominal maximum duration of 2.3 hours, this figure indicates that the array temperature will drop below $-120^{\circ}C$ for the selected array specific weight of 1.0 lb/ft^2. One of the simplifying assumptions made in this analysis was that during the eclipse period the solar array received no thermal input. In actuality the backup structure and cabling, the combined weight of which being approximately equal to that of the array, do in fact add heat to the array during the cooling period and, in this way, serve to reduce the cooling rate. While this effect was not carefully studied, it is not unreasonable to anticipate that the actual array cooling rate would correspond to that for an equivalent specific weight of at least 1.5 lb/ft^2 leading to a lower temperature of approximately $-115^{\circ}C$.

Figure 144 shows the approximate array temperature increase for different step-inputs of irradiation, the irradiation presumed to have the spectral distribution of solar energy. The starting temperature of $-110^{\circ}C$ is an estimate of the array temperature at the time of retropulsion firing. It corresponds to the temperature at the end of an 85 minute cooling period for an array weight of 1.0 lb/ft^2, and (α/ϵ) of 0.85, and an ϵ of 0.84. These curves indicate that the plume heating (less than 0.03 of the solar intensity at 1 AU) for a duration of 400 seconds will cause only a small increase in cell temperature. Because the heat pulse from the ablatively-cooled engine skirt will not be a step-function, these curves are not applicable. However, with the present skirt design the heat pulse, in the vicinity of both earth and Mars, will cause only an insignificant increase in array temperature.

11.5.4 Insulation

From the results of tests and analytical studies performed by Douglas Aircraft Co. (Report SM-48806) on NRC-2, the following generalized equations for the net heat loss through insulation over a flat, isothermal surface have been formulated:

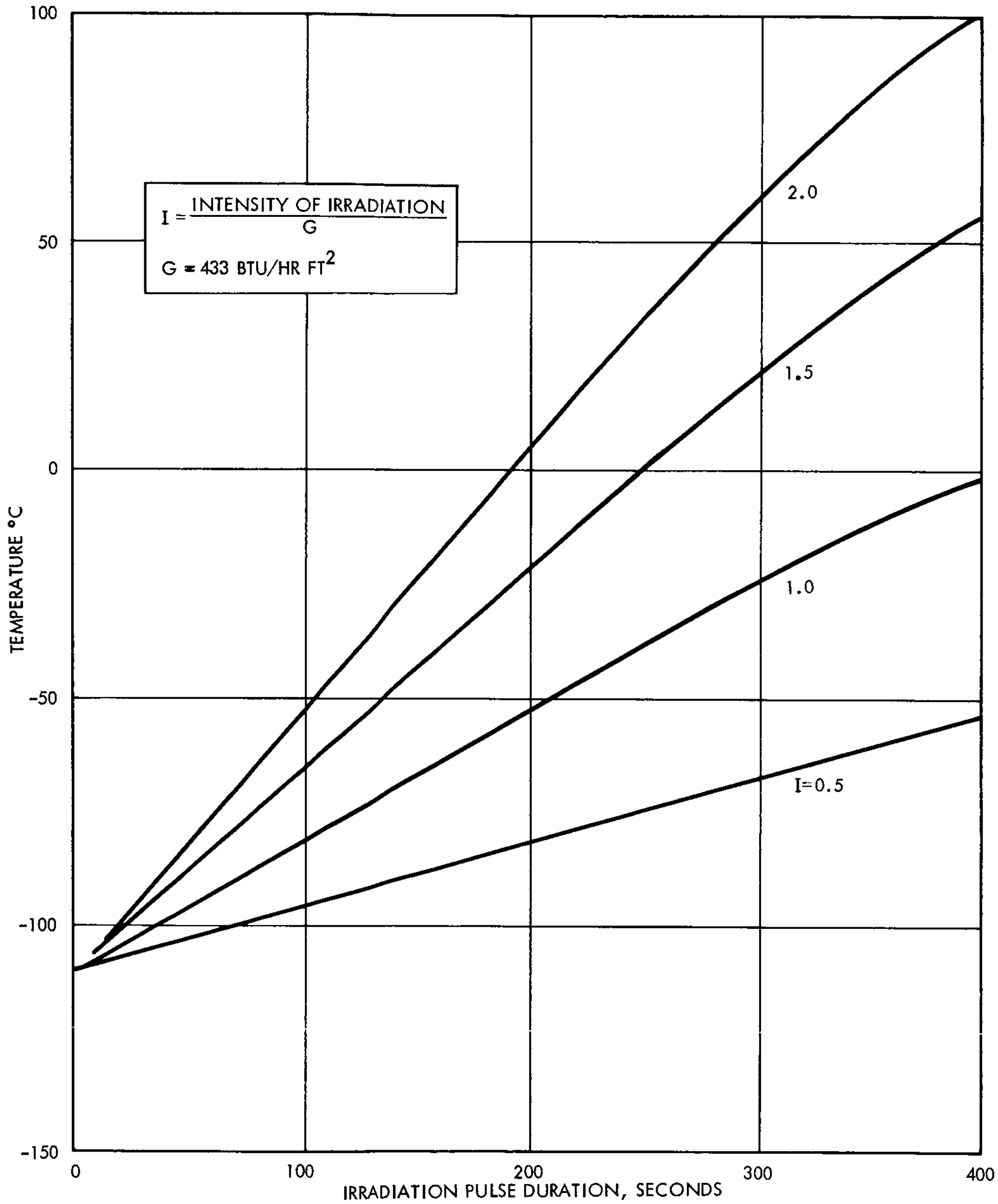


Figure 144. Solar Array Temperature Rise During Step-Input Irradiation

$$Q_{NET} = \sigma \epsilon_C T_C^4 - \alpha_C G = \frac{1}{35.66\,(N-1)}\,\sigma\left(T_H^4 - T_C^4\right)$$

$$+ \frac{8.16 \times 10^{-5}\,D^{1.186}}{(N-1)}\,(T_H - T_C)$$

where

Q_{NET} = net heat loss through insulation, Btu/hr ft^2

σ = Stefan-Boltzmann constant (0.173×10^{-8} Btu/hr ft^2 $^oR^4$)

ϵ_C = exterior surface emissivity (0.78)

T_C = exterior surface temperature, ^oR

α_C = exterior surface solar absorptivity (0.24)

G = solar intensity, Btu/hr ft^2 (433 at 1 AU)

T_H = interior surface temperature, ^oR

N = number of sheets of insulation

D = number of sheets per inch (55)

Figure 145 shows the net heat loss as a function of the number of sheets and for several values of the interior surface temperature. Curves are shown for two cases, the sun normal to the insulation and for no external thermal input.

When making thermal analyses of an insulated compartment, it is convenient to characterize the insulation by its apparent conductance (k/l), as if it behaved as solid insulation. Figure 146 shows this conductance, based on the information given in Figure 145, as a function of its composite thickness and number of sheets (assuming a packing density of 55 sheet/in.). Also shown in Figure 146 is a test point for Dimplar insulation; this was recently obtained at TRW in connection with an experimental investigation of its use as insulation for cryogenic fluids. Inasmuch as the main compartment analysis (Section 11.5.1) indicated that an overall (flat surfaces, corners, etc.) value of k/l of about 0.001 is required, it appears likely that 35 to 50 sheets of NRC-2 or 15 to 20 sheets of Dimplar may be used.

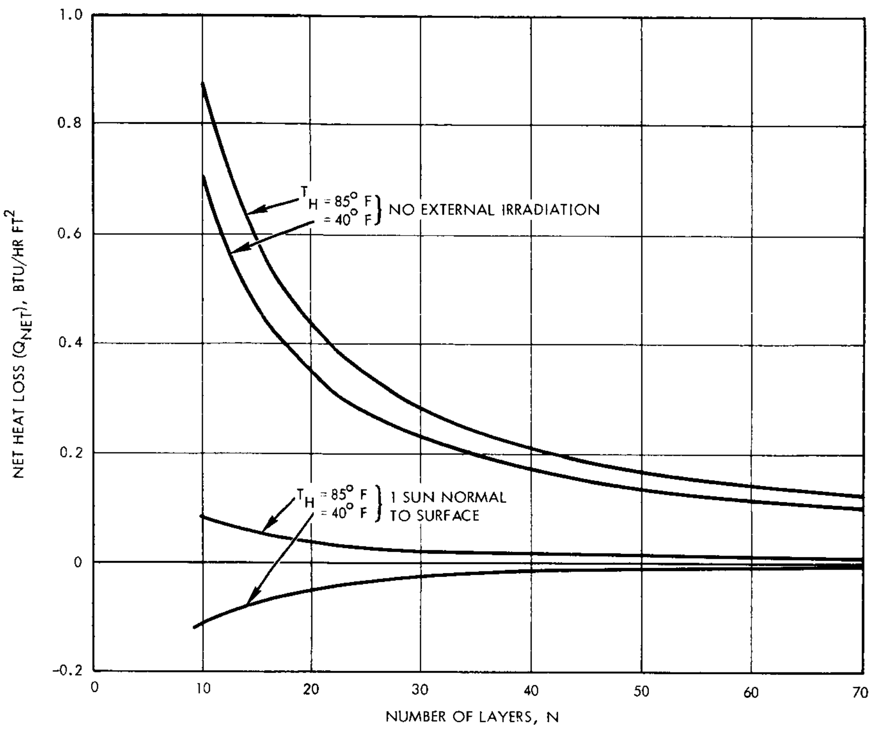


Figure 145. Heat Loss Through NRC-2 Insulation

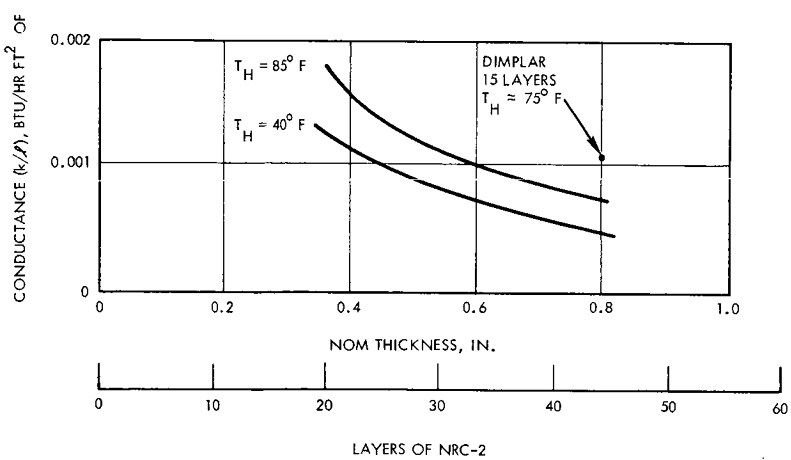


Figure 146. Conductance of Multilayer Insulation

11.5.5 Error Sources

Error sources in a temperature control subsystem are limited almost entirely to uncertainties in thermal properties and characteristics of the spacecraft materials, particularly as these may be affected by a prolonged mission enviornment. During the development period an effort will be directed toward assessing these effects. Recognizing that, at best, there will remain uncertainties, conservatism in design and spacecraft test conditions are employed to provide a margin of safety.

Reliability of louvers, heaters and thermostats have been determined for this mission from reasonable engineering assumptions (see report for Phase 1A Task A, Vol 4, Appendix B). The basic redundancy feature of louver assemblies for temperature control purposes will require that a large number of cases (environments, equipment duty cycles, etc.) be examined during development phases to determine prime failure modes.

12. CABLING SUBSYSTEM

12.1 General Description

The spacecraft cabling and electrical distribution subsystem consists of the following elements:

- All spacecraft wiring harnesses except those furnished as integral parts of GFE assemblies
- Junction boxes for the distribution and integration of electrical functions
- Umbilical cabling associated with the spacecraft
- System level test points including hardline test connectors.

The functions of the cabling subsystem are to distribute electrical signals and power throughout the spacecraft bus, to integrate all electrical subsystems into the overall bus, to integrate the science, capsule, and launch vehicle electrically with the spacecraft, and to provide the system level test points. A spacecraft system block diagram illustrating functional power and signal flow by the cabling subsystem is shown in Volume 1. This diagram illustrates the functional interfaces between subsystems and is a starting point for preparation of detailed wiring diagrams.

12.2 Requirements and Constraints

12.2.1 Electromagnetic Compatibility

The distribution of electrical signals and power throughout the spacecraft must be implemented properly to assure electromagnetic compatibility among all the spacecraft electronic subsystems. The major design requirements to be considered in this respect are bonding, grounding, shielding, and cable routing.

a. Electrical Bonding

Bonding requirements in the design of cabling encompass the electrical bonding of connector shells to the spacecraft structure via the subassembly and/or cable trough to form part of the spacecraft electrical reference system. Electrical bonding will be accomplished by metal-to-metal contact over the entire mating surface of the connector shell in contact with the subassembly chassis or cable trough. A bonding resistance of 2.5 milliohms maximum between any connector shell and mating structure is a design requirement. Specific provisions will be made to preclude contamination of bonding surfaces with nonconductive oxides and finishes.

b. Grounding

The Voyager spacecraft will utilize a common reference plane grounding configuration, with the electrically bonded spacecraft structure providing the low-impedance common reference plane.

The spacecraft AC power ground of the 400 cps and 4 kc square-wave power will be connected to the spacecraft structure at one location only. This location will be the power distribution unit in the power subsystem. AC power will be DC isolated from the chassis ground and all other grounds everywhere in the system.

Primary DC power, both the regulated 50 vdc and the unregulated DC, will be chassis grounded within the battery package. DC power will be isolated from AC power and grounds everywhere else in the spacecraft.

Secondary DC power is defined as the DC output of power supplies within individual subsystems operated from the primary AC power buses.

All secondary DC power will be connected directly to the chassis at the power conversion point.

All signals which are unbalanced to ground, including pulse, digital, and radio frequency circuits, will be connected to the assembly chassis at both the sending and receiving ends of the circuit. This requirement does not alter the DC isolation requirement between secondary and primary power.

All signals which are balanced to ground will be connected to the chassis at the point of balance.

Analog signal circuits will be connected to the subassembly chassis only at the load end of the circuit.

c. Cable Shielding

Electromagnetic shielding will be employed on all cables which interconnect assemblies, equipment, spacecraft, and EOSE containing circuits that are capable of generating electromagnetic fields or are susceptible to electromagnetic fields. The degree of shielding will be sufficient to insure a compatible system with an adequate compatibility margin.

Primary power cables, both AC and DC, will generally not be shielded, except for cables which carry switched actuation power to such units as solenoids or pyrotechnic devices. Secondary power cables will be shielded as dictated by the circuit design of the equipment.

RF (S-Band and UHF), both sending and receiving, including the interface cables between receivers and detectors will employ solid-jacketed or double-shielded coax cables.

Pulse, both digital and change-of-state, and synchronization signal cables will employ single-shielded cables, preferably a flat braid shield with 90 percent coverage as a minimum. Analog signal cables will be shielded as dictated by the interfacing circuit design. The cable trough will be designed to provide some degree of continuous RF shielding.

d. Cable Routing

The cabling subsystem routing will be designed to minimize common circuit impedance, cross-talk, radiation and pickup. Signal types (primary power, RF input, RF output, analog, and pyrotechnics) will be isolated from each other and from the remaining system cabling, consisting primarily of pulse circuit cables, within the cable troughs where possible. Cables will be routed in direct contact with the equipment mounting panels or some other structural member. Provisions will be made to accommodate spacecraft checkout probes without disconnecting components and subassemblies.

12.2.2 Magnetics

Cables and J-boxes will be fabricated from nonmagnetic materials. All power leads will be twisted to eliminate stray magnetic fields from this source.

12.2.3 Reliability

The cabling subsystem will use proven materials and connectors chosen from the Voyager Approved Parts List. Redundancy of critical circuits will be provided as found necessary during studies to be conducted during Phase 1B.

12.2.4 Assembly and Test Provisions

Design of the spacecraft cabling will allow installation of complete equipment mounting subpanels, including the flight harness interconnecting the units mounted on the panel so that the entire subpanel including associated flight harness can be checked as a subsystem by EOSE.

Separate connectors will be provided for the introduction of all OSE test cables that support the system and subsystem level testing so that normal flight cabling does not have to be disturbed.

12.3 Functional Interfaces

12.3.1 Bus Subsystems

It is the function of the cabling subsystem to provide the hardware electrical interfaces between the spacecraft bus subsystem. One of the products of the interface definition documents will be the cable wire lists detailing the electrical circuits between subsystems. Other functions of the cabling subsystem in implementing this interface will be the selection and standardization of electronic package connectors, the definition of connector indexing, the origination, implementation, and maintenance of the connector reference designator system, documentation, and assignments.

System design requirements for electromagnetic control and the electrical reference system provide another definition of the functional interface of the cabling subsystem with other bus subsystems, as well as other interfacing systems. Hardware details for the maintenance of the electrical reference system are provided to the interfacing packages at either end of the cables for bonding, ground, and shield terminations.

12.3.2 Spacecraft Bus-Science Interface

A major factor in the design implementation of the science equipment interfaces concerns the degree of variability of the science subsystem configuration as the design proceeds together with allowances for requirements of future missions. The design needs to attain the flexibility to be able to adapt changes in the science subsystem with minimum changes in the spacecraft hardware. Much of this capability will be provided within the science DAE, and within a junction box mounted on the equipment panel assigned to the science assemblies.

12.3.3 Spacecraft Bus-Flight Capsule Interface

The spacecraft bus-flight capsule interface includes the transmission of the spacecraft bus DC power to the capsule, the transmission of hardline commands to the capsule, and the receipt of hardline data from the flight capsule.

A separable umbilical connector is required to provide for jettison of the capsule adapter and canister. Only the hardmounted connector will remain with the spacecraft if the emergency jettison of the capsule becomes necessary.

12.3.4 Spacecraft Bus-Planetary Vehicle Adapter Interface

The flight spacecraft umbilical connector, which is separated late in the launch countdown sequence, carries the ground power to the spacecraft, control and monitor functions for on-stand tests including hardline telemetry for periods of radio silence, and control and monitoring of the safe/arm device. A connector such as the Lockheed explosively-actuated umbilical connector used on the Agena and the OGO spacecraft is planned. The connectors to the separation devices remain with the portion of the planetary vehicle adapter which stays with the shroud after separation.

12.4 Design Description

To satisfy all of the requirements and constraints of Section 12.2 the cabling subsystem is designed to provide wires and connectors meeting all of the mechanical and electrical specifications imposed and to accomplish several objectives for cabling layout, routing, and harnessing:

- Electrical isolation and physical separation of the following signal types: power, analog signals, digital signals, RF circuitry, and pyrotechnic initiation circuitry
- Routing and separation which will enable the maximum use of structural members for support and for additional shielding
- Simplification of interfaces for the reduction of multiple long runs, the elimination of octopus-like harnesses, and ease of fabrication, installation and logistics.

12.4.1 Equipment Mounting Panels

Most of the electronic subsystems are mounted on the vertical equipment mounting panels Nos. III and VII. Each of these main mounting panels is divided into four subpanels, two each measuring 38 x 38 inches, and two each measuring 38 x 27 inch. The total of eight subpanels permits the assignment of a subpanel for each grouping of electronic subsystems. The tentative subsystem assignments on each panel are as follows:

Panel III

Subsystem	Subpanel size (inch)
Telemetry and Data Storage	38 x 38
Radio	38 x 38
Command and Computing and Sequencing	38 x 27
Guidance and Control UHF Receiver and Recorder	38 x 27

Panel VII

Subsystem	Subpanel size (inch)
Power Equipment and Pyrotechnics	38 x 27
Science and DAE	38 x 38
Batteries and Regulators	38 x 38
Science Support Equipment	38 x 27

The proposed subpanel packaging approach, shown in Figure 147, uses vertical rails, spaced far enough apart to run cable troughs along each row or assemblies. Connectors are placed on the side of the assemblies to minimize cable bending. Another approach under investigation is the use of a fixed intrapanel harness; the assemblies would then plug into connectors provided on the subpanels.

Figure 147. Radio Subsystem Panel and Cabling

The subpanel cable harnesses will be designed to permit the removal and test of each subpanel with its associated subsystems and flight harness,by routing all cabling interconnecting the subsystem with the spacecraft electrical system through the appropriate junction box on the aft equipment module at the base of and on either side of the vertical equipment mounting panels III and VII. Some functions, such as semi-rigid coaxial cables will not be routed through the junction boxes, but these can be disconnected directly at the subpanel to permit removal of the entire subpanel.

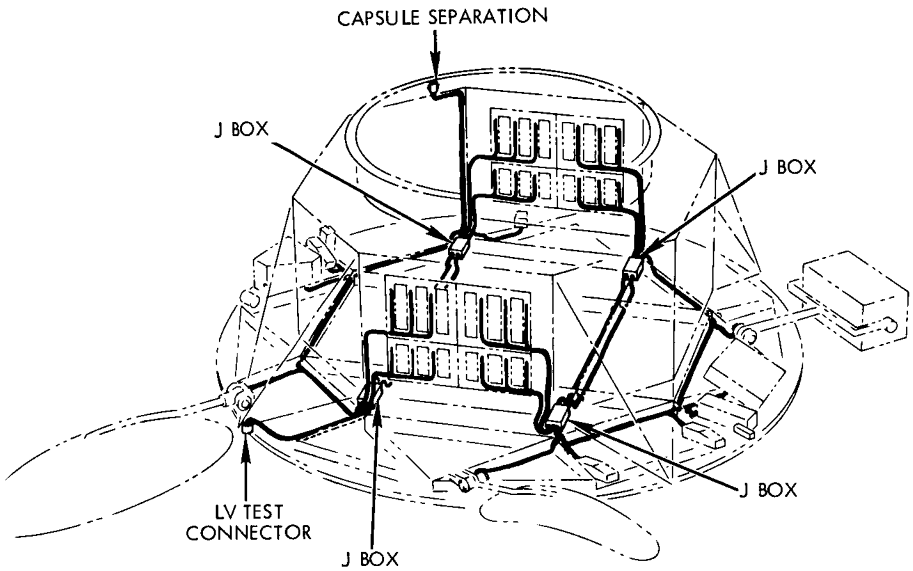
The boxes on the panel are mounted in vertical columns with up to three columns per subpanel. Connectors will be located on the sides of the boxes and the cable harnesses will lay vertically between the columns attached firmly to the mounting rails and panels. Sufficient clearance will be provided at the top and bottom of each panel for routing the main branches of the harness to the sides of the subpanel and down the sides to the main junction boxes. Connector assignments and cabling branch assignments to the top or bottom of the panel will be determined and chosen by signal type, i.e., power, analog, digital, RF, or pyrotechnic.

12.4.2 Aft Equipment Module and Junction Boxes

The aft equipment module is the enclosed area around the structural beams between the eight panels and the solar array as shown in Figure 148. It provides support for the solar array, antennas, science packages, and guidance and control sensors. The four main junction boxes will be located inside the aft equipment module at the base of equipment panels III and VII adjacent to structural tie points C and D, and G and H. The junction boxes will be RF tight and compartmented to provide signal isolation. The primary function of these junction boxes is to permit the removal of a complete subsystem subpanel and interpanel harness. Other functions include the interconnection of subpanels, structurally mounted units, and hardline system level test connections.

PRELIMINARY SPECIFICATION

Cabling Subsystem



Subsystem Function

The distribution of electrical signals and power throughout the spacecraft bus. The integration of the science, capsule, and launch vehicle electrically with the spacecraft. The implementation of system level test points.

Subsystem Elements

Interconnecting cables
Junction boxes
Spacecraft and capsule umbilicals
System level hardline test connectors

Physical Characteristics

Total subsystem weight 228.9 pound

Reliability

Figure 148. Preliminary Specification — Cabling Subsystem

The structural beam latticework provides an excellent basis for spacecraft harness interconnections between these four junction boxes. Attaching the cables to either sides of the beams provides the support necessary and electromatnetic isolation afforded by the material content of the beam. There are two to four such isolated signal paths from any one junction box to any one other junction box in the module.

12.5 Parameters and Performance Summary

The parameters and general arrangement of the cabling subsystem are shown in Figure 14.

III. PROPULSION SUBSYSTEM DESCRIPTION

1. INTRODUCTION

Tradeoff studies were conducted to select the Voyager propulsion subsystem based on the mission requirements and the candidate propulsion systems defined in the JPL statement of work for Phase 1A Task B. These studies, described in detail in Volume 5, resulted in the selection of a modified lunar excursion module descent stage (M-LEMDS). This section of Volume 2 describes the selected propulsion subsystem in terms of design requirements, design implementation, and performance characteristics.

The propulsion subsystem is required to perform three basic maneuvers for the Voyager mission: (1) midcourse corrections during a 7-month interplanetary cruise, (2) insertion into Mars orbit, (3) and trim of the Mars orbit. As many as three midcourse corrections may be used to provide a ΔV of up to 200 m/sec. These maneuvers do not require continuous thrust modulation, therefore the LEM descent engine flow control valve and actuator can be removed. The 7-month interplanetary cruise requires positive sealing of propellant and pressurants. The ball valves and their actuators which are not capable of providing this positive sealing are eliminated and explosive valves substituted. The zero g starting requirement necessitates the addition of positive expulsion start tanks.

The LEMDS propellant tanks have a capacity of 18,000 pounds of propellant while only 11,734 pounds are required for the 1971 Voyager mission. This excess tankage capacity is used to provide a blow down mode of engine operation during the long interplanetary cruise. This method of operation allows the pressurant gas to remain sealed in the high pressure bottle during the entire 7-month interplanetary cruise period.

During the orbit insertion maneuver, the pressurant gas will be regulated in a conventional manner and the engine will operate at a constant 7750-pound thrust. Since the blowdown mode is used during the orbit trim maneuver, the thrust will decrease slightly from its initial value of 1050 pounds.

2. REQUIREMENTS

The total impulse to be provided by the propulsion subsystem is established by the velocity increments for midcourse correction, orbit insertion and orbit trim and the weight allocations specified for the 1971-73 and 1975-77 missions. Other Voyager design requirements include restart capability, space storability, impulse reproducibility and accuracy, and subsystem reliability.

2.1 Mission Requirements

Velocity increment requirements for the various maneuvers for the Voyager spacecraft were specified by Jet Propulsion Laboratory Preliminary Mission Description for the 1971 mission, 15 October 1965. These requirements and the assumed extension of the requirements for the Voyager program through 1977 are presented in Table 71.

Table 71. Velocity Increment Requirements

1971 and 1973 Missions	
Sum of Midcourse Corrections	200 meters/second
Insertion into Martian Orbit	
Required	2.0 kilometers/second
Desired	2.2 kilometers/second
Orbit Trim prior to Capsule Separation	100 meter/second
1975 and 1977 Missions	
Midcourse	200 meter/second
Insertion into Orbit	Maximum possible within Spacecraft weight constraints
Orbit Trim prior to Capsule Separation	100 meters/second

2.1.1 Velocity Increment and Spacecraft Weight

Weight allocations for the Task B Voyager missions were established by the Voyager 1971 Preliminary Mission Description, Jet Propulsion Laboratory, 15 October 1965. These allocations are given in Table 72, where it may be observed that separate quotas were established for the propulsion system and bus. Since many items of

Table 72. Weight Allocations

	1971-73	1975-77
Gross Injected Weight, lb	20,500	28,500
Lander	3,000	10,000
Bus	2,500	3,500
Propulsion	15,000	15,000

structure, cabling and thermal control could be arbitrarily assigned to either category, TRW has assumed for purposes of determining system performance capability that the sum of the bus and propulsion weights must be maintained within the allocation shown. For 1975 and 1977, the 1000-pound increase in payload weight allocation was added to the bus.

2.1.2 Restart

The nominal mission consists of four engine firings: two for mid-course corrections, one for orbit insertion and one for orbit trim. All of these maneuvers occur prior to separation of the landing capsule; subsequent to separation it may be desirable to perform orbit trims during the orbital operations phase. Additional starts may be required during interplanetary cruise.

2.1.3 Space Storage

The Voyager mission from launch to completion of operation of the propulsion system will last from 4 to 8 months. The potential hazards in the space environment include ionizing radiation, hard vacuum, micrometeorites and solar thermal radiation. The expected environments for these factors are defined in the Voyager Environmental Predictions Document, Jet Propulsion Laboratory, 19 October 1965.

2.1.4 Planetary Quarantine—Mars Contamination

The planetary quarantine restriction that the probability that Mars is contaminated prior to the year 2021 shall be less than 10^{-4} will require trajectory biasing and maneuver orientation. Potential failure modes must not lead to explosive failures, the results of which could detach parts of the spacecraft which in turn may exceed the allowable probability of planetary capture. To eliminate the possibilities of structural failures during the required 50-year life in orbit, tanks and pressure vessels will probably have to be vented at the conclusion of all propulsive maneuvers. Analysis of the venting processes will be required to ensure that quarantine is not violated.

2.2 Subsystem Performance Requirements

2.2.1 Engine Performance and Subsystem Weight Requirements

The limiting values of specific impulse for the retro maneuver and propulsion subsystem inert weight which meet the mission requirement of Paragraph 2.1 are shown in Figure 149. Curves are shown for both the JPL "desired" and "required" capabilities. For the nominal retro specific impulse of 305 seconds, the "required" orbital insertion ΔV, 2.0 km/sec, will be obtained if the propulsion subsystem weight is 3900 pounds, while the "desired" ΔV, 2.2 km/sec, is obtained with a weight of 3290 pounds.

2.2.2 Minimum Impulse Bit Requirement

The "Voyager 1971 Mission Guidelines," JPL, 1 May 1965 calls out a "required" value of the minimum velocity correction of 1.0 meter/sec. This value was assumed as a guide line for this study. To achieve this velocity correction with a 19,000-pound spacecraft (estimated weight at end of midcourse corrections), the engine should be capable of firings which produce a total minimum impulse (integrated thrust-time profile) of 1950 lbf-sec.

2.2.3 Impulse Delivery Predictability Requirements

In order that the execution error from one maneuver does not excessively degrade that maneuver's accuracy, the "Voyager 1971 Mission Guidelines", JPL, 1 May 1965 specifies that the velocity error resulting from the non proportional error in the total impulse should not exceed 0.1 m/sec. Thus, the reproducibility for midcourse corrections would be ±195 lbf-sec.

2.2.4 Materials

Moving parts exposed to the space vacuum shall be fabricated from materials which preclude the possibility of failure due to cold welding. Ablative materials shall be selected and the thrust chamber designed,

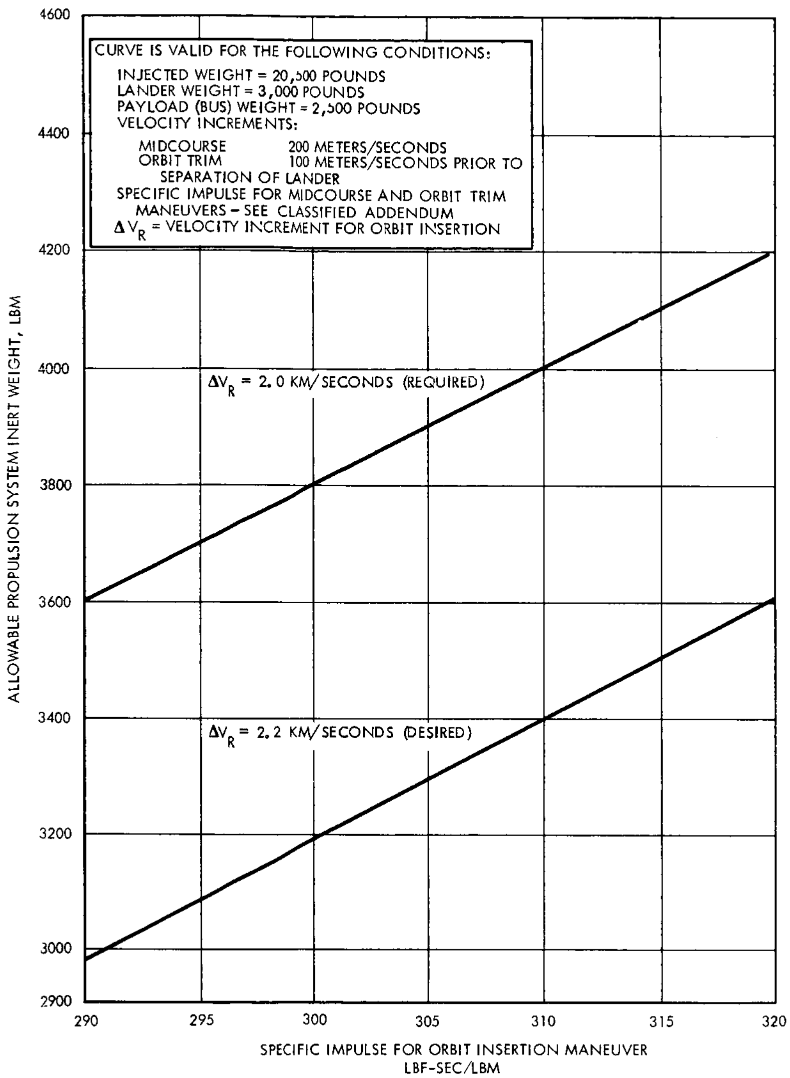


Figure 149. Specific Impulse and Propulsion System Weight Requirements

such that outgassing during vacuum storage shall not degrade thrust chamber life. Materials that are corrosion resistant, in the pre-launch, launch and space environments will be used.

2.2.5 Engine Operational Life

For the 1971-73 missions, the engine must be capable of operation for a total of 467 seconds at 1050-pounds thrust, for the midcourse correction and orbit trim maneuvers, and 380 seconds at 7750-pounds thrust, for the orbit insertion maneuver. For the 1975-77 missions, these times are 690 seconds and 347 seconds, respectively.

2.3 Design Requirements and Philosophy

2.3.1 Design Philosophy

The approach to the Voyager propulsion system design was basically to retain LEMDS structure and component designs and to make modifications only as required to meet minimum Voyager mission requirements or to satisfy Voyager spacecraft interface requirements. The consequences of this philosophy is a minimum cost development effort and an over-capacity, less than optimum performance system. However, as shown in the discussion on system operation, Paragraph 4.3, the "over capacity" not only provides growth potential, but permits an operational technique which is uniquely applicable to the Voyager mission.

In areas where modifications are required, the following guide lines were used:

- Operational functions should not be assigned to components which require invention or development beyond the capacity of existing devices.
- Where possible, components should be used in a derated mode in preference to the use of redundant components.
- Where redundant components are employed, they should be preferentially used in a passive sense such that failure sensing and switching are minimized.

2.3.2 Static Loads

The propulsion subsystem must be designed to withstand a maximum acceleration of 7.0 g's along the booster thrust axis. The acceleration laterally is assumed not to exceed 1.25 g.

2.3.3 Launch Vibratory and Shock Loads

The propulsion subsystem will be designed to withstand the following vibration and shock loads in addition to those that are self-induced. The random vibration environment for a payload attached directly to the shroud will be assumed to be the following omnidirectional input to the spacecraft at the attachment point to the shroud:

1) At liftoff, power spectral density peaks of 1 g^2/cps ranging from 150 to 300 cps with a 4 db/octave roll-off below 150 cps and 6 db/octave roll-off above 300 cps in the envelope defining peaks; the time duration is approximately 30 seconds;

2) At transonic, power spectral density peaks of 0.07 g^2/cps ranging from 300 to 600 cps with a 3 db/octave roll-off below 300 cps and 9 db/octave roll-off above 600 cps in the envelope defining peaks; the time duration is approximately two minutes.

The shock response due to shroud separation and spacecraft separation is approximated by an input consisting of a 200 g terminal peak sawtooth with 0.7 to 1.0 milliseconds rise time.

2.3.4 Zero "g" and Propellant Feed

The propulsion subsystem will be designed to start eight times under zero g conditions. This provides twice the number of starts required by the nominal mission profile.

3. FUNCTIONAL INTERFACES

The functional interfaces of the propulsion subsystem are illustrated in Figure 150. The subsystem interfaces with structure for mechanical attachments of the pressurization assembly, feed system assembly, and engine assembly; and with the sequencer and command subsystem and the power subsystem for actuation of the solenoid valves and engine actuators. The propulsion subsystem interfaces with guidance and control at the gimbal actuator attachment lug. The propulsion subsystem and thermal control subsystem have two interfaces. One is the requirement for thermal control to maintain propellant and hardware temperatures within specified limits in order that satisfactory propulsion subsystem performance is achieved. The other is the requirement for thermal protection and thermal control for other subsystems to satisfactorily tolerate heat rejected from the engine. These interfaces are defined in section II-11 of this volume. The communication subsystem interfaces with propulsion in that it telemeters data for performance and malfunction analysis. Signals to operate pyrotechnic devices are provided by the pyrotechnics subsystem with direct electrical connections to all squibs.

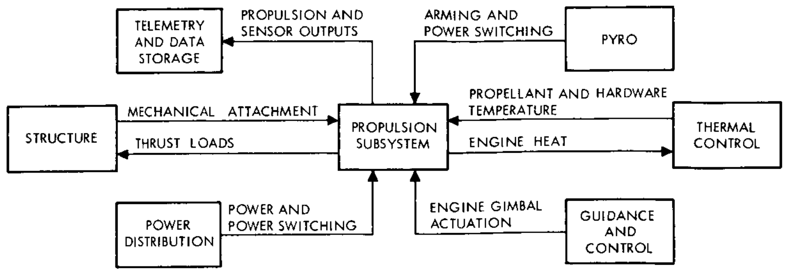


Figure 150. Propulsion Subsystem Functional Interfaces

3.1 Mechanical Interfaces

The propulsion components interface mechanically with the Descent Stage structure for support of the propellant tanks, plumbing, control components, and thrust chamber.

3.1.1 Engine Mount

Support of the engine is accomplished as shown in Figure 154. The engine mount is similar to the mount used on the LEMDS as shown in Figure 152; the difference being that the engine is lowered 36 inches for the Voyager application. Attachments to the gimbal ring and actuators are the same as those used in the LEMDS.

3.1.2 Tank Support

Main propellant tanks will be attached to the structural support members in the same manner as used in the LEMDS assembly.

3.1.3 Propellant Lines and Support

All lines, fittings, and components situated upstream of the propellant feed system-engine interface will be supported by the main structural elements in the same manner used in assembly of the LEMDS. The only modifications consist of removing 36 inches of vertical line and adding an additional 1/2 inch diameter propellant line from the start tanks to the propellant valves.

4. PROPULSION SUBSYSTEM DESIGN DESCRIPTION

This section describes the existing LEMDS and the approach recommended for modifying the basic LEM Descent Stage propulsion subsystem for the Voyager mission. Propulsion functional operations and interface requirements are presented and interrelationships with other subsystems and over-all operations are shown. Table 7 of the Classified Addendum presents a preliminary specification of the propulsion subsystem.

4.1 LEMDS Propulsion Subsystem (Existing Stage for Apollo Mission)

The existing LEMDS is shown in Figure 151 with more detail of the current descent engine installation presented in Figure 152.

A stored helium gas system provides main tank pressurization. Current Apollo effort includes development of both an ambient temperature and a super-critical temperature helium storage system. The supercritical helium storage system provides a significant weight savings for the relatively short Apollo mission; however, it is not suitable for the Voyager application because of the long interplanetary cruise period between engine firings.

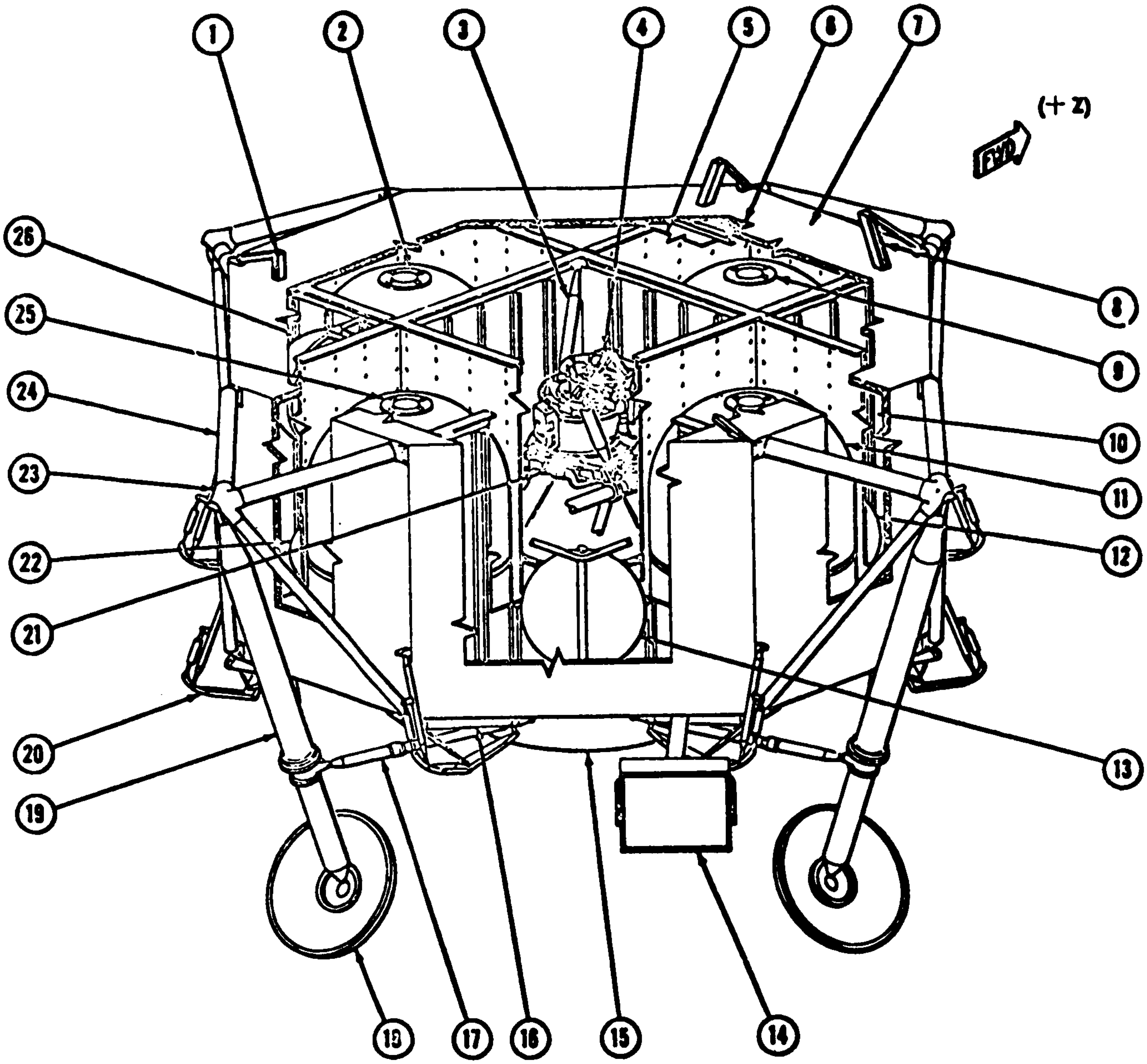
Propellant orientation during zero "g" operation is accomplished on the LEMDS by firing four aft pointing reaction control system 100 lb engines located in the Ascent Stage for a duration of 3 to 5 seconds.

Propellant on-off flow to the rocket engine is controlled by series-parallel bipropellant valves. The valves are mechanically-linked, fuel-actuated ball valves controlled by a solenoid operated pilot valve.

Throttling is achieved with a mechanically linked variable area injector and variable area cavitating venturi flow control valves. The thrust chamber is gimballed at the throat plane to direct the thrust vector through the vehicle center of gravity. Both the injector-flow control valve actuator and gimbal actuator are electromechanical devices.

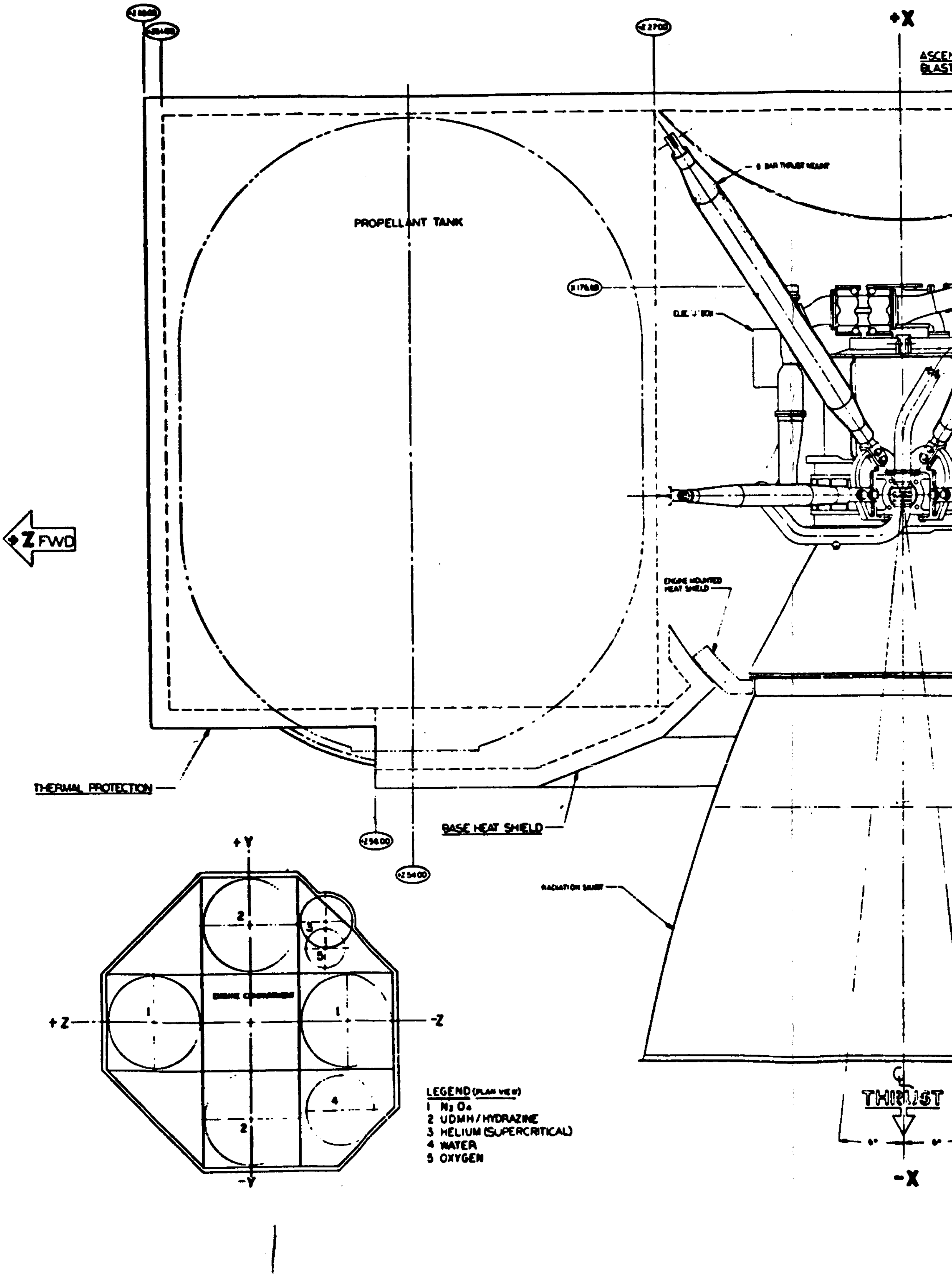
4.2 LEMDS System Considerations for Voyager Application

The basic LEMDS propulsion subsystem has many features readily adaptable for the Voyager mission. Ample propellant capacity is available in the existing propellant tanks. The variable thrust rocket engine assembly provides a range of thrust levels well suited for providing accurate midcourse corrections and efficient retro propulsion. The operational life of the engine is in excess of the anticipated requirement, and the LEMDS is designed to be compatible with a Saturn C-5 launch. Extensive development effort has already been expended to solve the space storability problems.



1. AFT INTERSTAGE FITTING
2. FUEL TANK
3. ENGINE MOUNT
4. DESCENT ENGINE
5. STRUCTURAL SKIN
6. INSULATION
7. THERMAL SHIELD
8. FORWARD INTERSTAGE FITTING
9. OXIDIZER TANK
10. SCIENTIFIC EQUIPMENT BAY
11. FUEL TANK
12. WATER TANK
13. HELIUM TANK
14. LANDING RADAR ANTENNA
15. DESCENT ENGINE SKIRT
16. TRUSS ASSEMBLY
17. SECONDARY STRUT
18. PAD
19. PRIMARY STRUT
20. LOCK ASSEMBLY
21. GIMBAL RING
22. OXYGEN TANK
23. ADAPTER ATTACHMENT POINT
24. OUTRIGGER
25. OXIDIZER TANK
26. HYDROGEN TANK

Figure 151. LEM Descent Stage


+X
PROPELLANT TANK
+Z FWD
THERMAL PROTECTION
BASE HEAT SHIELD
+Y
+Z
-Z
-Y
LEGEND (PLAN VIEW)
1 N_2O_4
2 UDMH/HYDRAZINE
3 HELIUM (SUPERCRITICAL)
4 WATER
5 OXYGEN
-X

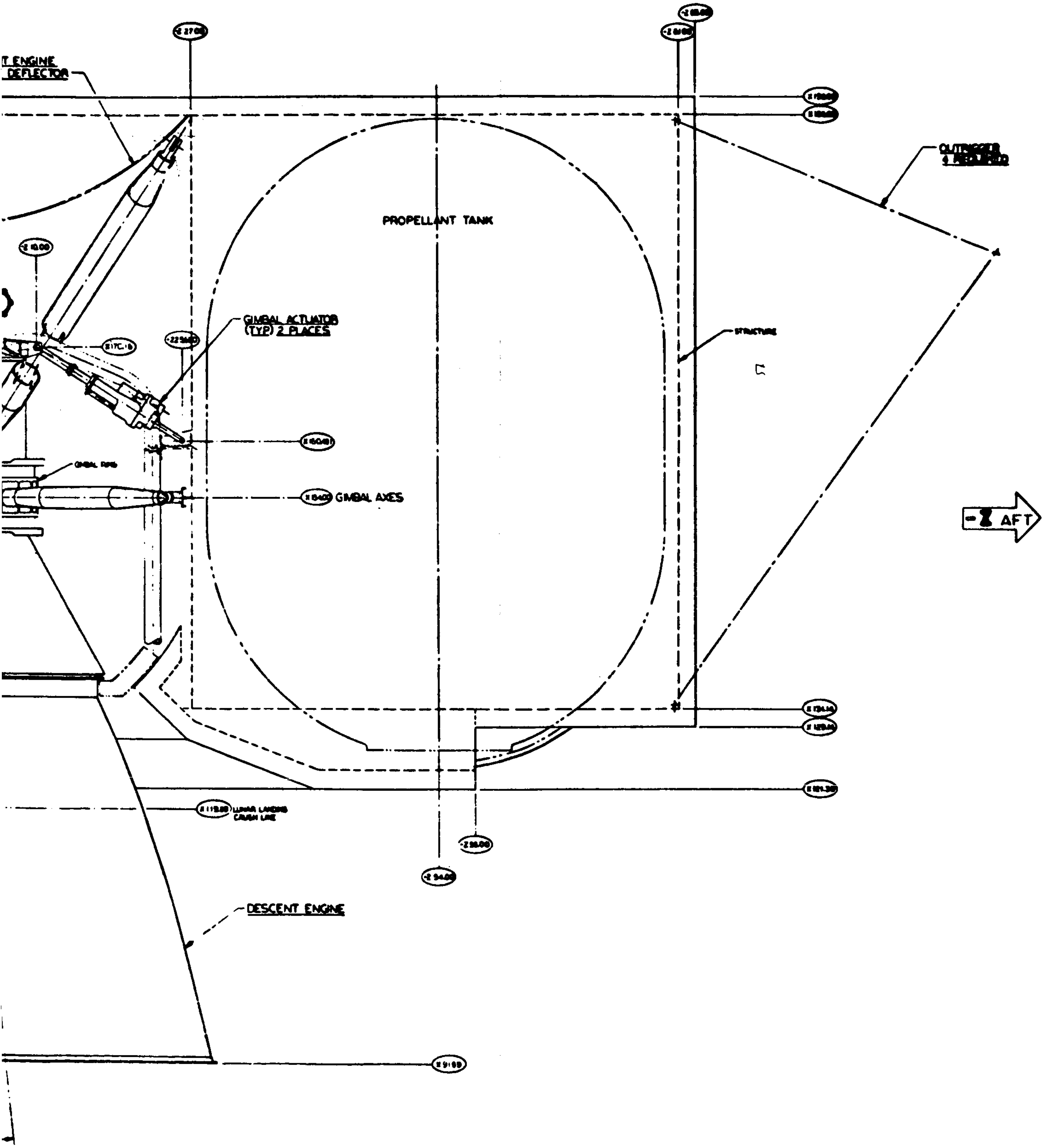


Figure 152. Descent Propulsion Engine Installation

However, since the descent stage structural arrangement and components were specifically optimized for the LEM vehicle and the Apollo mission, some modifications are required for application to the Voyager spacecraft.

4.3 Recommended Modifications

A major portion of the crew environmental control and life support equipment, water, hydrogen, and oxygen, are stored in the LEMDS. The landing gear and portions of the landing radar and electronics are also contained in the descent stage, as are other specialized electronics and scientific equipment designed especially for the Apollo mission. Since these subsystems are not required for Voyager mission, they may be removed and additional space is available for installing Voyager equipment.

In addition to the removal of LEMDS equipment not required for the Voyager mission, modifications must be incorporated to provide for zero "g" start capability, for long term space storability, for relocation of the engine gimbal plane, and for a means of reducing the radiant heat flux from the nozzle extension to the spacecraft solar array. One additional change recommended for the engine is to replace the continuous throttling mechanism with a simplified two-level thrust control system.

4.3.1 Zero "g" Start Capability

Propellant orientation prior to main engine ignition on the LEM vehicle is provided by firing the Ascent Stage aft pointing reaction control engines. This method is used since the reaction control engines are required for ascent stage operation and therefore available for propellant orientation. To utilize the descent stage without an ascent stage would require auxiliary motors. Auxiliary motors for propellant orientation, however, implies a complete feed system including positive displacement tanks. A simpler approach is to incorporate the positive displacement tanks in the LEM propellant system to feed propellant directly to the main engine. Several design options were considered, and the use of metal bellows start tanks immersed in the main tanks, as shown in Figure 153. was selected.

No modification of the existing LEM propellant tanks are required since the start tank is assembled on the reworked existing propellant

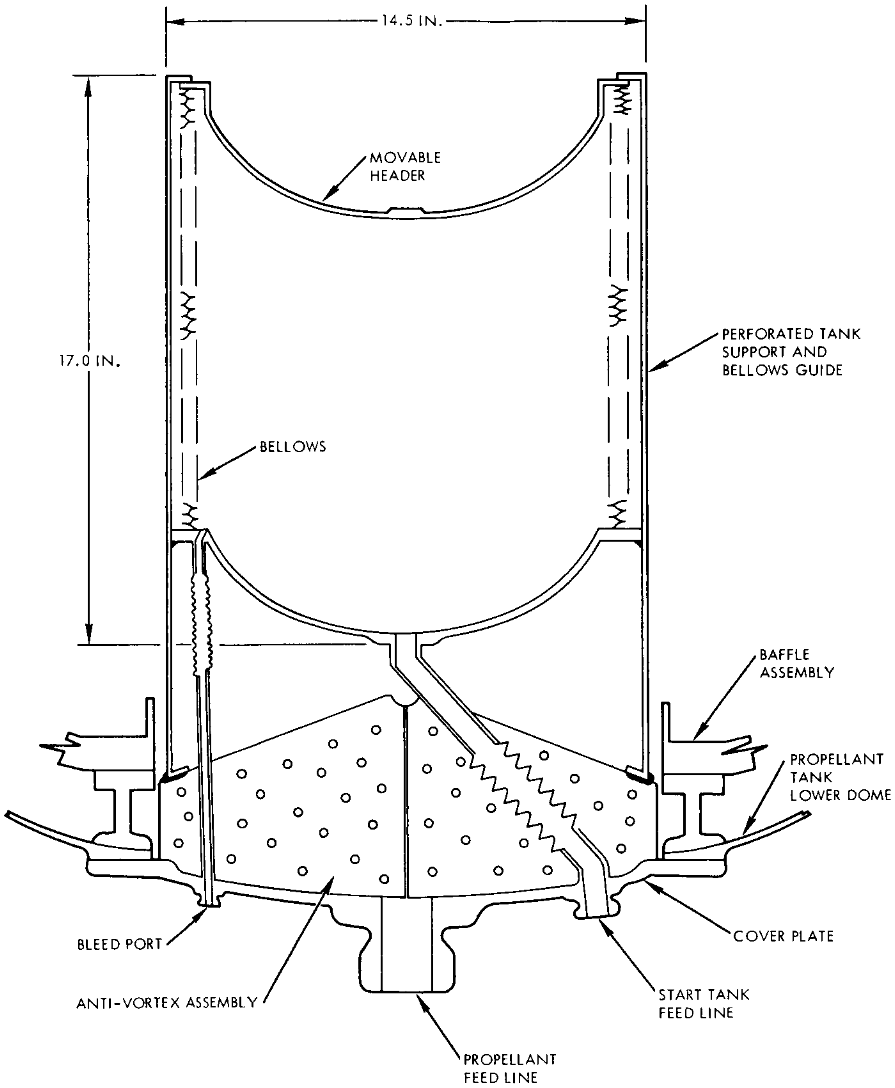


Figure 153. Start Tank Assembly

tank bottom cover plate. When assembled on the bottom cover, the tank will fit through the existing bottom access port.

4.3.2 Long Term Storability

The major problems associated with long term storability for the LEMDS are gas and liquid leakage, corrosion and contamination of feed system components exposed to the oxidizer vapors, and stress corrosion of the pressure vessels. The leakage problem was solved by two straightforward techniques. First, the multicycle LEM shutoff valves were replaced with positive sealing explosive actuated valves. Secondly, the gas pressurization system is maintained in the sealed condition during the entire interplanetary cruise phase by employing brazed or welded joints and connections and by using explosive valves at the helium tank outlet and below the quad check valves (marked "D" and "K" Figure 155)

The LEMDS propellant tanks have a capacity of 18,000 pounds while only 11,374 pounds of propellant are required for the 1971 Voyager mission. This excess tank capacity and resultant additional gas ullage is used to provide a blow down mode of engine operation during the midcourse maneuvers required during the interplanetary cruise.

The tanks will be prepressurized to 125 psia prior to launch. Assuming 1400 lb. of propellant are required for the midcourse maneuvers and the tanks are at $50^{\circ}F$, the final tank pressures will be 95 psia. The propellant supply pressure decay of 25 percent will result in a thrust decay of 15 to 20 percent. There is therefore no need for using the sealed high pressure gas source during this phase of the mission.

The isolation of the gas pressurization system also eliminates the probability of component failures resulting from corrosion since explosive valves located downstream of the quad check valves isolate the pressurization components from the propellant vapors.

The problem of stress corrosion of the main propellant tanks may require development of chemical inhibitors in the propellant or coating of the tank walls. However, operating the feed system at reduced pressures for the midcourse maneuver will substantially reduce the problem and should, as a minimum, provide additional margin in the tank design.

To eliminate the possibilities of structural failures of the high pressure tankage and the subsequent contamination of Mars because of meteorite puncture during the 50 year life in orbit, all tanks will be depressurized after the last required orbit trim maneuver.

Cold welding problems will be avoided by using the following techniques:

- The pintle actuator and its electric motor will be housed in a hermetically sealed housing. This housing will provide an atmosphere to prevent evaporation of lubricants and surface films and provide a heat transfer medium to conduct heat away from the actuator. The housing will actually provide a second margin of safety because the actuator will be capable of operating in a vacuum for the life of the mission. The dc motor used in the design will be of the brushless design utilizing bearings capable of space operation and all sliding surfaces will be coated with solid film lubricants.

- The external moving parts of the injector which normally has no sliding surfaces will however have surfaces where small clearances occur coated with a solid film lubricant to prevent cold welding in the event contact does occur.

- The quad solenoid packages which operate for the orbit insertion maneuver and the orbit phase of the mission and that are exposed to a hard vacuum once the normally closed isolation squib valves are fired, will incorporate non-metallic seats, probably Teflon. The evaporation rate of teflon at temperatures less than 200°F is approximately 10^{-5} cm/year.

- The gimbal bearings presently being used in the gimbal assembly are impregnated teflon sleeve bearings which are capable of meeting the Voyager mission requirements because of the low evaporation rates of teflon.

The effects of space environment on the phenolic resin impregnated silica fibers which are used in the construction at the LEM descent engine ablative thrust chamber have been investigated by numerous researchers.* Further material on the subject can be found in the list of references in each of these reports. The discussion in the following paragraphs is in essence an abstract of those reports.

The significant natural and induced environments to which ablative materials are subjected in propulsion system applications are as follows:

- Penetrating radiation
- Solar ultraviolet radiation
- High vacuum
- Post-run cooling in a vacuum

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*Gayle, J. B., et al., "Vacuum Compatibility of Engineering Materials," NASA TMK-51166, Marshall Space Flight Center, September 16, 1963.

Rollbuhler, "Experimental Investigation of Rocket-Engine Ablative Material Performance After Postrun Cooling at Altitude Pressures," TN D-1726, NASA, June 1963.

"Space Materials Handbook," ML-TDR-64-40, AF Materials Laboratory and Technology Division, Air Force Systems Command, Wright-Patterson Air Force Base, Ohio, 2nd Edition, January 1965.

Schwartz, H. S., "Conference on Behavior of Plastics in Advanced Flight Vehicle Environments," WADD Technical Report 60-101, September 1960.

The effect of penetrating radiation up to dosages of 10^4 rads has been found to have no effect on the mechanical or electrical properties of phenolic impregnated silica. Data are scant on the behavior during irradiation in a vacuum; however, it is surmised on the basis of experimental work on other polymeric systems that the effects in a vacuum are not as severe as in air, since oxygen attack has been shown to play an important role in polymer degradation mechanisms.

Penetrating radiation dosages in space produced by atomic particles have been shown to be negligible insofar as cosmic radiation and inner and outer belt radiation dosages are concerned. This applies to both ionization and displacement of atoms. Solar flares will also not produce any significant damage due to atomic displacement in ablatives, and the ionization dosage for a year in space is 10^5 to 10^7 rads for the surface (10 microns in depth) and 10^2 to 10^3 rads for depths of 0.05 inch. It therefore would appear that penetrating radiation will not be a problem for ablative thrust chambers on the Voyager propulsion system.

Solar ultraviolet radiation has been shown to produce negligible changes in the properties of ablative materials. Tests have shown that combined effects of vacuum and ultraviolet radiation for periods of 500 hours result in a 1 percent weight loss due to evaporation and a 10 percent loss of flexural strength. Due to the limited penetration of this radiation, less than 0.010 inch, the inner surface of the thrust chamber which is pointed in the direction of the sun which will acquire a protective layer of char after the first 0.05 second of firing; and since the outer fibers are protected by the titanium outer shell or the glass wrap, it can be safely assumed that no damage will be done to the ablative material due to the sun's ultraviolet radiation.

The effect of high vacuum on ablative materials has been examined by many investigators and the essence of their findings has been twofold: first, there is a slight weight loss of approximately 1 percent; and second, there is a 36.5 percent improvement in compressive strength even at elevated temperatures up to 400°F. In general, it can be said that the effects of high vacuum on ablative materials do not appear to be a serious design consideration.

The last item in question is: what happens to ablative materials which have been subjected to combustion chamber gases when they are allowed to cool in a vacuum. This problem has been investigated in a controlled experiment by NASA.* Two identical ablative nozzles of 150 pound thrust were test fired six times each (40 second tests) at 100 psia chamber pressure using hydrogen-oxygen as the propellant combination. One nozzle was cooled after each test while it was at ambient pressure while the other nozzle was cooled after each test while it was subjected to low-pressure conditions. The results of the tests were presented in terms of nozzle weight losses, char-layer thickness, and internal dimension changes. The results of these test series revealed that there was no difference in erosion rates, char rates, or weight loss between the two thrust chambers. The conclusions of this study were that cooling ablatives at low pressures as compared with cooling at atmospheric pressure had little noticeable effect on the ablative material.

The temperature extremes the propulsion subsystem will encounter during storage and operation on Voyager are well within the allowable values. Propellant temperature will be maintained in the range of 50 to 90°F with a maximum differential between tanks of 10°F. Since the freezing point of the propellants is 11 and 21°F (oxidizer and fuel respectively), a margin for design uncertainty exists. The allowable temperature range of engine components is -65 to 160°F, which can be provided.

The engine ablative material, which will be almost continuously exposed to the sun's ultraviolet rays, is capable of withstanding temperatures of 800°F which is well beyond the predicted equilibrium temperatures.

4.3.3 Engine Burn Time Capability

The present design burn time for the LEMDS engine is 1200 seconds. The LEMDS duty cycle requires 887 seconds of firing and this has been

*Rollbuhler, "Experimental Investigation of Rocket Engine Ablative Material Performance After Postrun Cooling at Altitude Pressures," TND-1726, NASA; June 1963.

demonstrated at Arnold Engineering Development Center at 100,000 foot altitude operation. The Voyager duty cycle for 1971-73 requires 847 seconds of firing. The LEMDS engine injector and combustion chamber can therefore be applied to the Voyager program without any modifications.

4.3.4 Relocation of the Engine Gimbal Plane

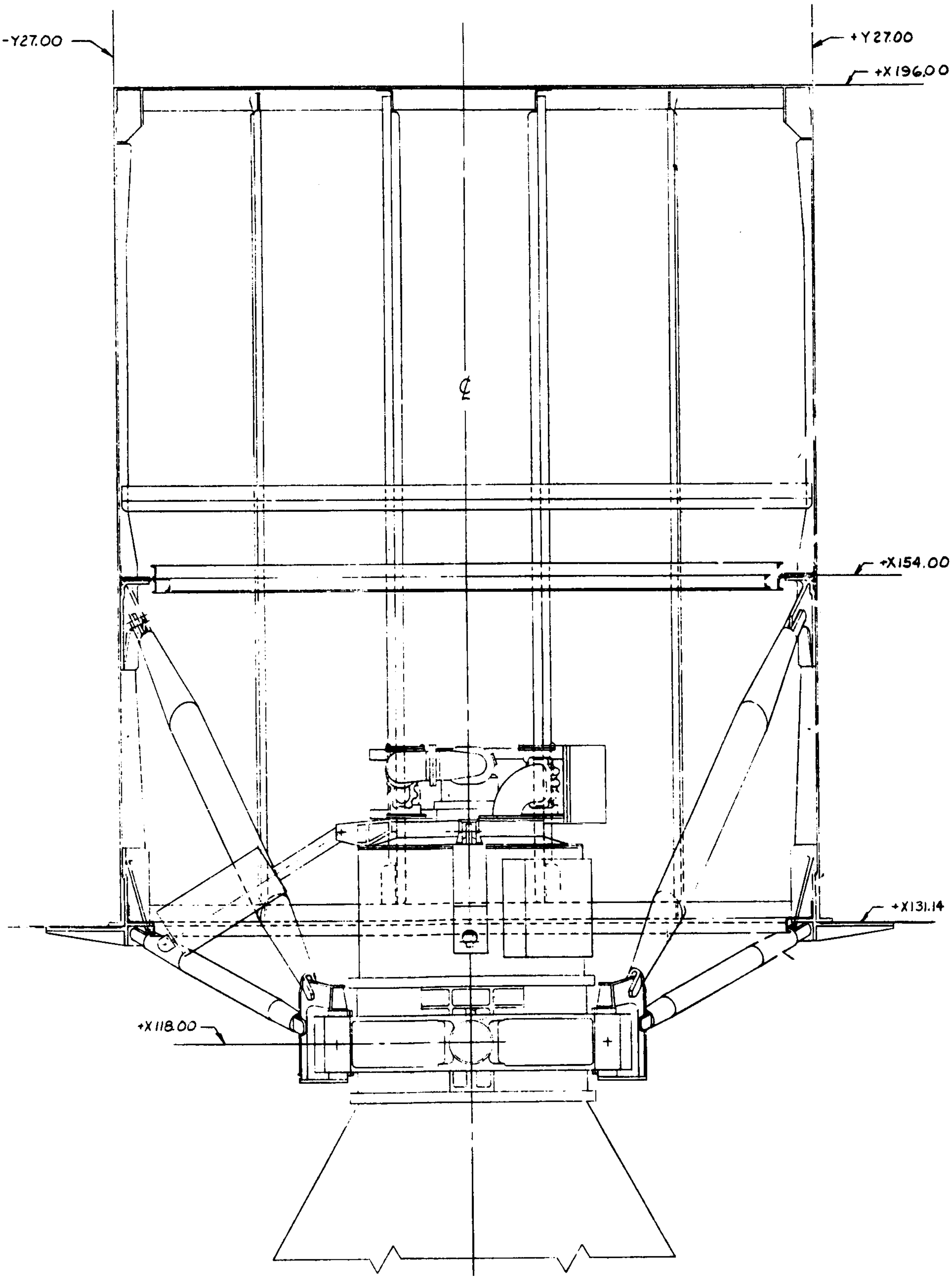
For the normal spacecraft modes of operation, where the engine is fired with the landing capsule in place, the LEM thrust mount and engine gimbal plane location would have been satisfactory. However, in the event that a maneuver would be required with the landing capsule removed, the CG of the vehicle would shift close to the engine gimbal plane, effectively reducing the control capability of the system. This potential problem was remedied by lowering the engine 36 inches as shown in Figure 154 as opposed to the original location shown in Figure 152.

4.3.5 Reduction of Radiant Heat Flux to the Solar Array

The LEM engine nozzle extension currently reaches a maximum temperature in the order of 2200°F. Lowering of the engine in the structure has the effect of reducing this temperature by eliminating the reradiation from the spacecraft heat shield. However, even at the anticipated lower temperatures, heat transferred to the solar array in the vicinity of the nozzle extension could permanently degrade the performance of the array. For the proposed design and for purposes of performance comparison, a conservative assumption was made to replace the columbium extension by an ablative extension as shown in Figure 156.

4.3.6 Modification of Throttle and Flow Controls

Since the Voyager mission can be readily accomplished with a "step" thrust engine, considerable savings in weight and reduction in complexity of the thrust control system can be effected by removing the continuous throttling capability of the engine. In the proposed modified engine shown in Figure 156 and schematically in Figure 157 the variable area cavitating venturi valves, the high speed electromechanical servo actuator, the quad redundant fuel actuated ball valves, and the mechanical linkage drive system were replaced with a two-position electromechanical actuator mounted directly on the injector, fixed area flow control orifices, and explosive actuated control valves.



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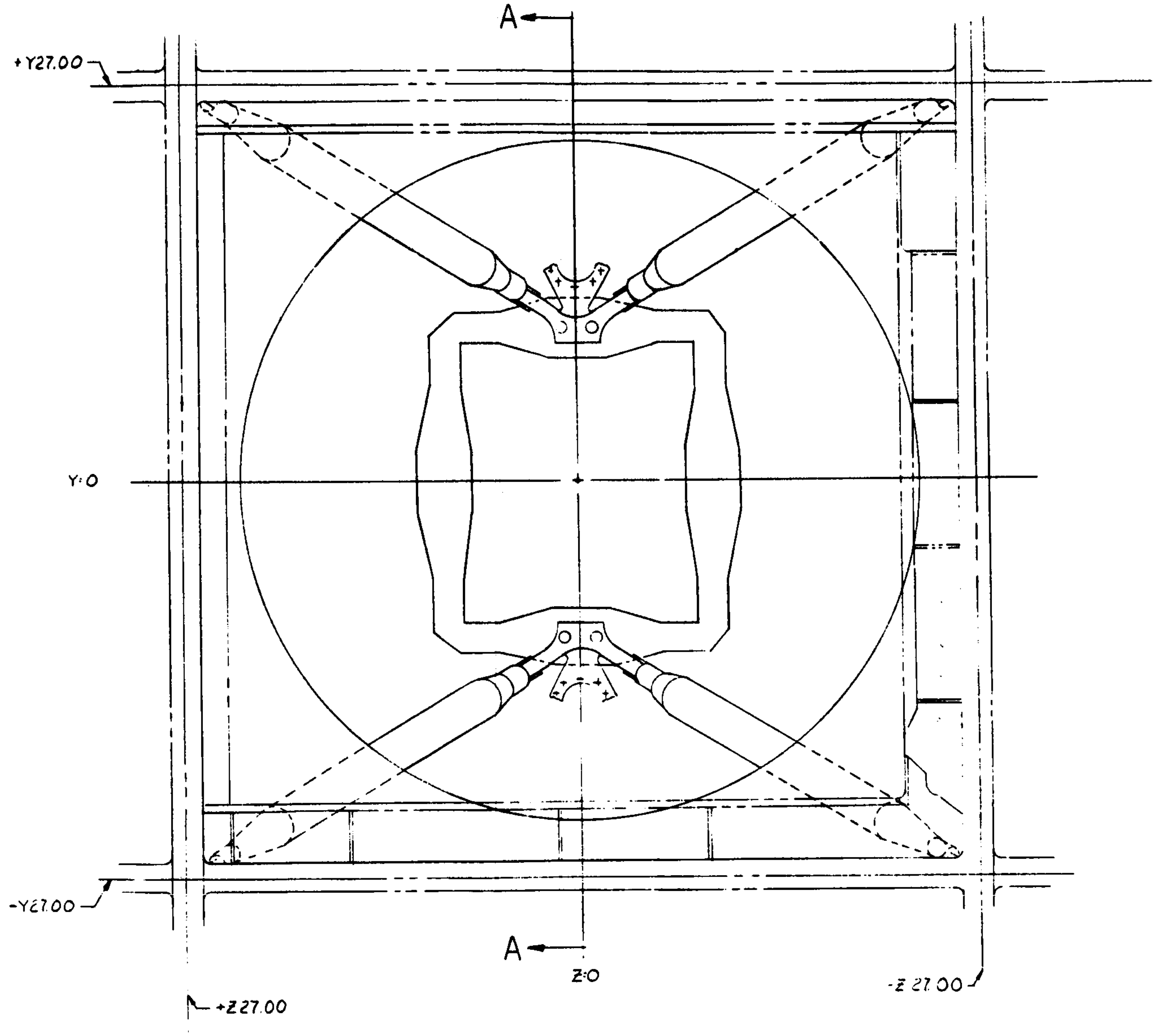


Figure 154. Engine Location for Voyager Vehicle

4.4 Voyager Propulsion Subsystem

The above modifications were made to the LEMDS propulsion subsystem. The original LEMDS feed system schematic is shown in Figure 155 and the modifications made are shown on the overlay. As can be seen from this figure, additional pressure transducers were incorporated into the system to assist in isolating possible malfunctions. Also, the two helium tanks were replaced with one larger tank in order to improve the spacecraft e.g. location. The tank with characteristics as shown in Table 75 is already developed and is fabricated from the same materials as the original tanks and has a diameter of 40.9 inches. This provides a volume of 38,368 cubic inches with a weight of 392 pounds. The new tank will be filled with 40 pounds of helium as opposed to 50 pounds for LEM.

In addition, the zero "g" start tanks and their lines, filters, and pressure transducers were added as previously discussed.

As can be seen, very little change was made to the existing LEMDS feed system so that all of the LEM major development items and qualified components are retained for the Voyager Mission.

The same philosophy of retaining the major development items in the present LEM propellant feed system was also followed in the engine area. The injector, combustion chamber and gimballing assembly remain unchanged. The only changes occurred in the control components and the nozzle extension as shown in Figure 156. These changes, discussed in Paragraphs 4.3.5 and 4.3.6, are minor since the major development dollars are associated with the development of the injector, combustion chamber, and gimbal assembly.

A schematic of the proposed engine is shown in Figure 157. The midcourse maneuver explosive valves and line sizes are one-half inch diameter while the orbit insertion explosive valve sizes are one inch. The orbit trim, start quad solenoid valves and quad check valve sizes are three-eighths inch size. Filters are provided downstream of all explosive valves to prevent contamination of downstream components. Filters are also provided upstream of the quad solenoid valves to prevent valve seat contamination. Orifices are used for low pressure drop

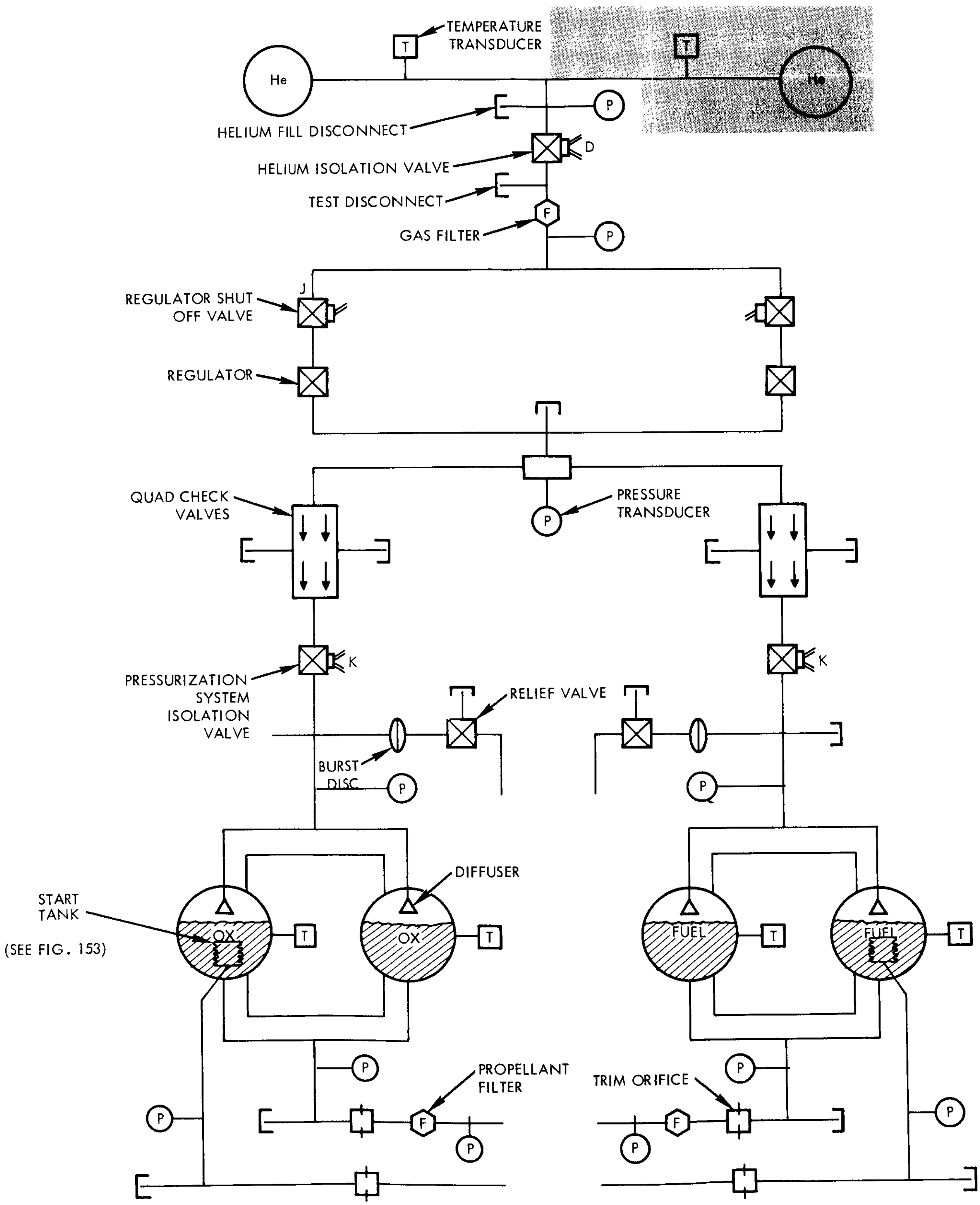
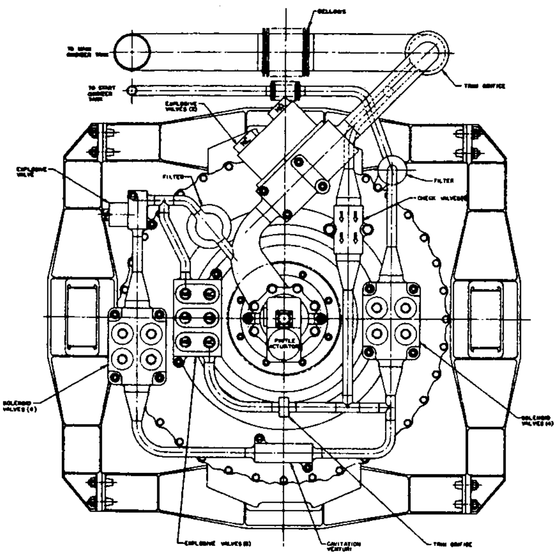
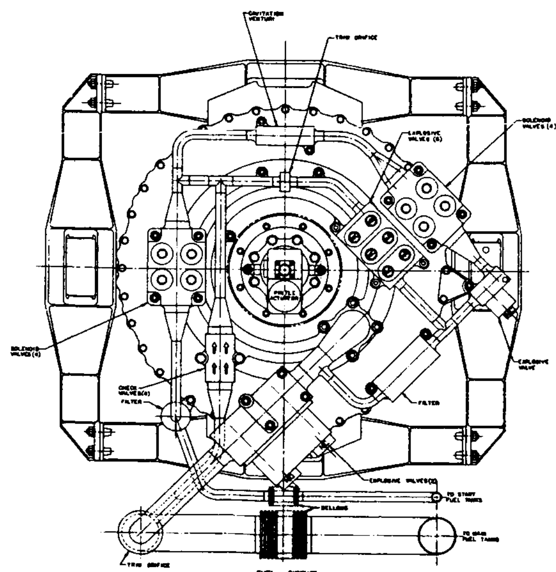


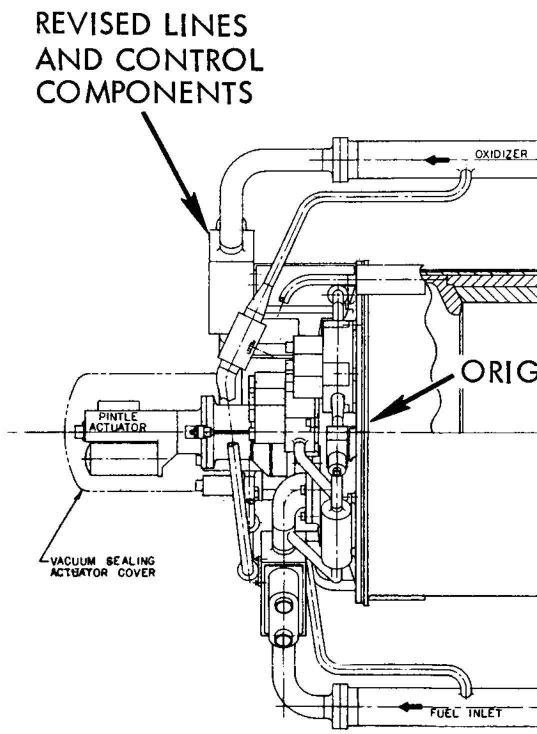
Figure 155. Present LEM Feed System



OXIDIZER CIRCUIT



FUEL CIRCUIT



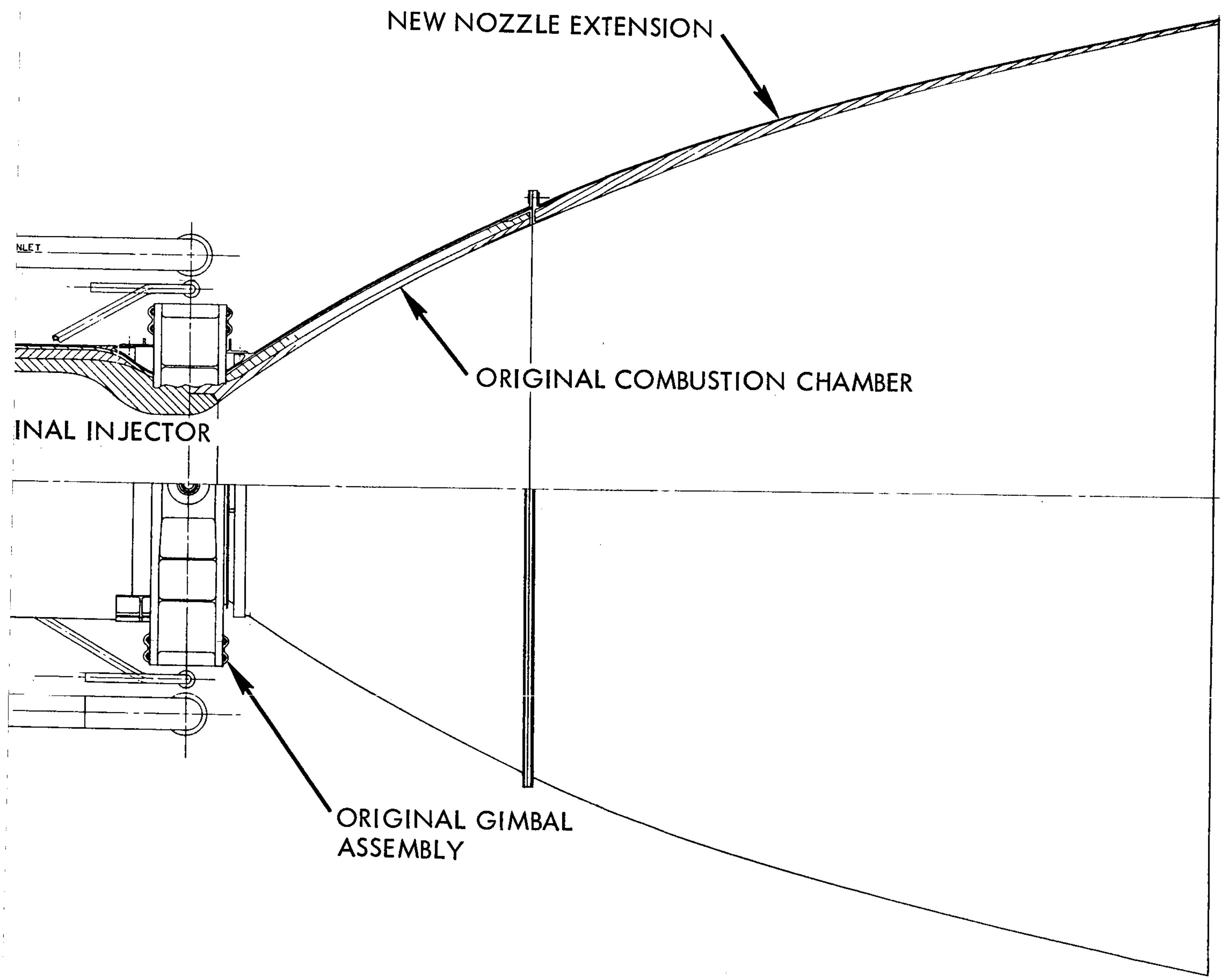


Figure 156. Voyager Rocket Engine

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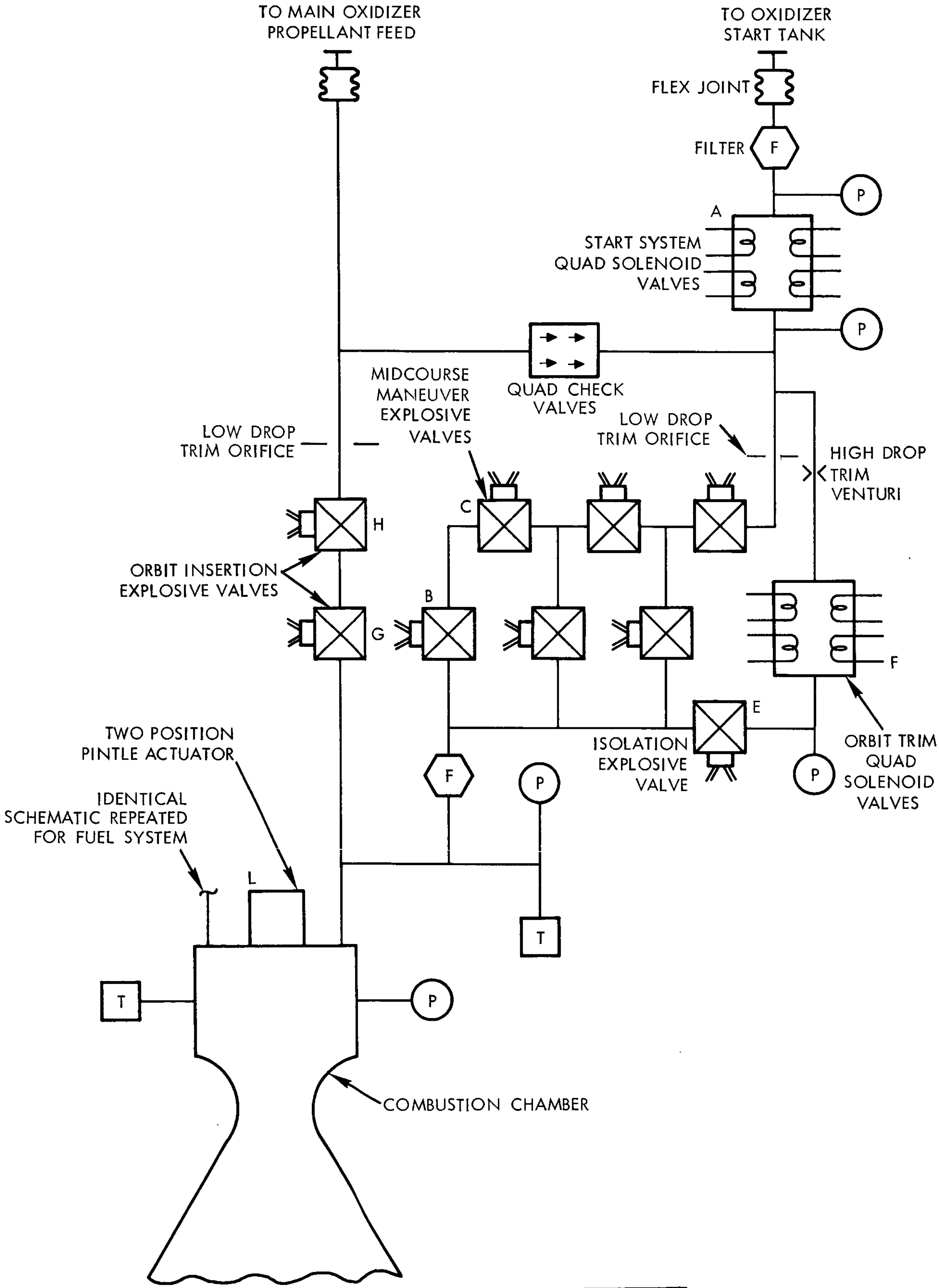


Figure 157. Voyager Engine Schematic

trimming when maximum pressure is required for engine operation and venturies are used for high pressure drop trimming when excessive tank pressure is available for satisfactory engine operation. Quad packaging of check and solenoid valves are employed in a series parallel arrangement so that no single malfunction can effect engine operation.

The propulsion subsystem vehicle control interrelationship is shown in Figure 158.

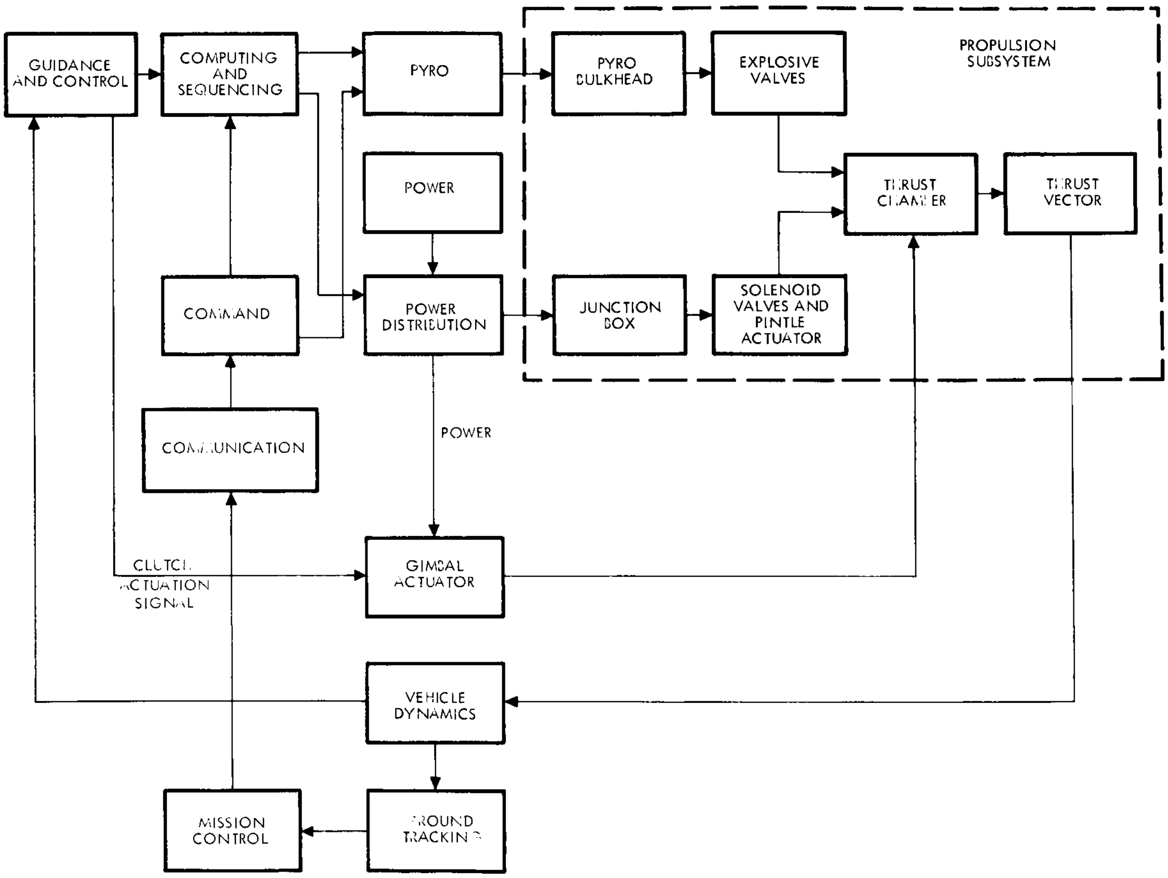


Figure 158. Propulsion Subsystem Vehicle Control Interrelationship

a. Sequence of Operation

The Voyager propulsion subsystem sequence of operation is as follows:

At Launch check to see that:

Main propellant tanks are pressurized to 125 psi.

One regulator shutoff valve is latched open (J) Figure 155.

Injector pintle is set at low thrust position.

First midcourse firing

1) Open starting quad solenoid valves (A), Figure 157. Main tank ullage pressure used to expel start tank. Check valves stay closed.

2) Open normally closed explosive valve (B), Figure 157. Fire six seconds to settle main tank propellants.

3) Close starting tank quad solenoid valves (A). Check valves now open and propellant flows from main tanks. The engine operates in a blow down mode since the helium isolation valve (D), Figure 155, has not been opened as yet.

4) Continue firing until accelerometer or timer shuts normally open explosive valve (C) after required impulse expended.

7-10 Month Cruise

All systems sealed.

Propellant tank pressure level low.

30 Days Prior to Encounter (Second midcourse firing)

Same firing sequence as the first midcourse firing.

Orbit Insertion Firing

1) Open helium and pressurization system isolation valves (D and K) Figure 155.

   Main tank pressurizes to 225 psia.

2) Open starting quad solenoid valves (A) Figure 157. Check valves stay closed.

3) Open normally closed explosive valve (E) downstream of quad solenoid valves.

4) Open quad solenoid valves (F).

5) Fire six seconds to settle propellants then shut starting quad solenoid valves (A).

6) Check valves open. Main tank feeds engine thru quad solenoid valves (F).

7) Open normally closed explosive valve (G) and move injector pintle to full thrust position.

8) Close quad solenoid valves (F).

9) Continue firing until accelerometer or timer shuts normally open explosive valve (H) after required impulse expended.

10) Move pintle to low thrust position using actuator (L).

Orbit Trim Firings

1) Open starting quad solenoid valves (A). Check valves stay closed.

2) Open quad solenoid valves (F).

3) Fire six seconds to settle propellants then shut starting quad solenoid valves (A).

4) Continue firing until accelerometer or timer shuts quad solenoid valves (F).

Tank Depressurization

1) Open oxidizer quad solenoid valves (A and F) until tank pressure reaches desired level.

2) Close above valves.

3) Open and close the fuel valves in a similar manner.

b. Propulsion Electrical System

The propulsion electrical system encompasses the circuits and electrical equipment associated with primary power and signal distribution. It also includes power dissipating components such as the pressurization system regulator isolation solenoids, start and shutdown propellant quad solenoid valves, explosive valves and the pintle actuator. All of the above engine items are tied together by an electrical harness through a propulsion junction box, which contains the electrical control interface connectors, instrumentation interface connectors and the system checkout connectors. This approach will be reviewed in regard to the desirability of bringing the pyrotechnics firing signals through connectors. All wiring will be twisted shielded pairs. An electrical system block diagram is shown in Figure 159.

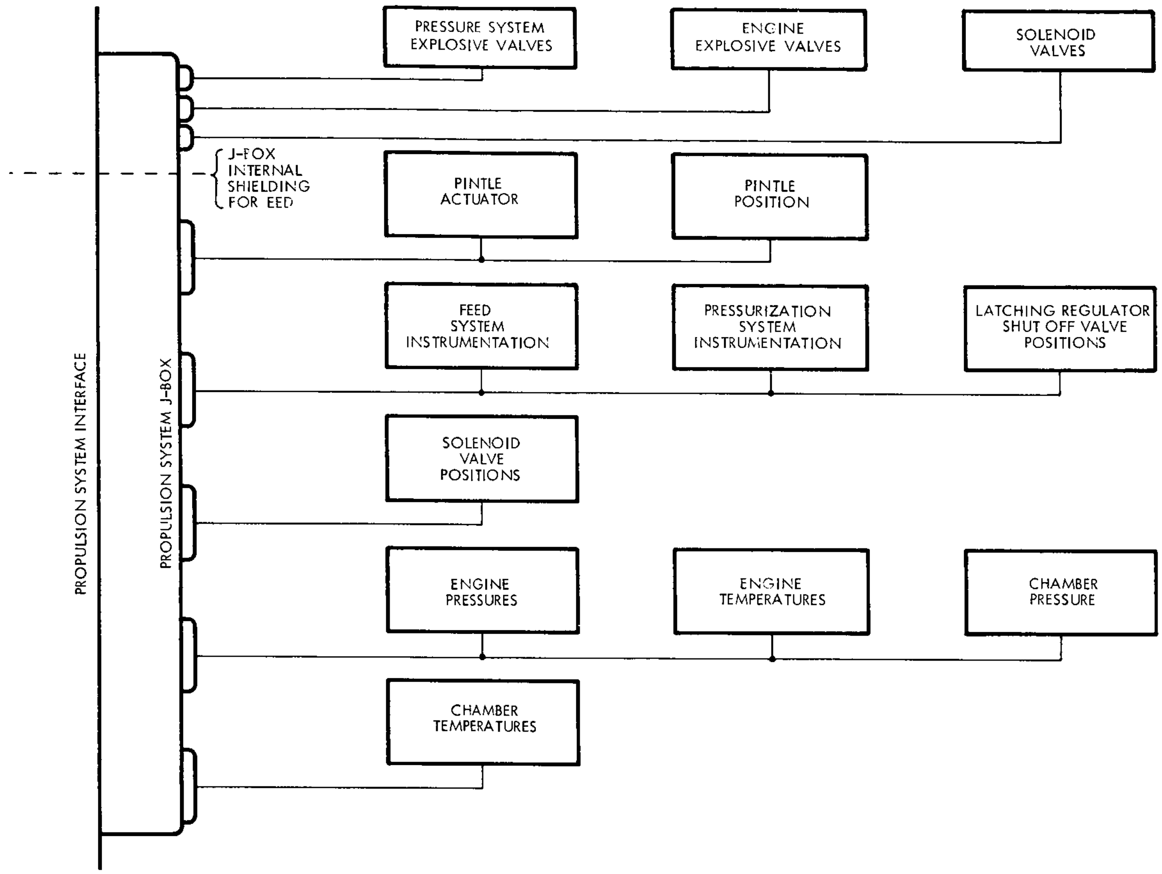


Figure 159. Propulsion Subsystem Electrical Block Diagram

The electrical system instrumentation sensors are listed in Table 73 and shown in Figures 155 and 157. The transducers include pressure sensors, thermocouples, valve position indicators and current monitors. An electrical harness is provided for the feed system control wiring and needed sensors. The Electrical Operational Support Equipment needed to check out the above system and its use is presented in Volume 4, Section 4.9 of the EOSE Section II.

The propulsion electrical power requirements are summarized in Table 74.

4.5 Component Descriptions

Mechanical design features and operational characteristics of the major components used in the Voyager propulsion subsystem are presented in the preliminary specifications which follow (Tables 75 through 86). Component selections were made on the basis of utilizing existing, proven hardware.

Table 73. Propulsion Subsystem Measurements

Item	Accuracy	Quantity	Range (psia)	Desired Sampling Rate (spm)
PRESSURE				
1. Pressurization System				
Helium tank	2%	1	0-4000	1
Regulator inlet	2%	1	0-4000	1
Regulator discharge	2%	1	0-300	1
2. Feed System				
Oxidizer tank inlet	2%	1	0-300	1
Fuel tank inlet	2%	1	0-300	1
Oxidizer tank discharge	2%	1	0-300	1
Fuel tank discharge	2%	1	0-300	1
Oxidizer start tank discharge	2%	1	0-300	1
Fuel start tank discharge	2%	1	0-300	1
3. Engine				
Engine feed system interface oxidizer	2%	1	0-300	100
Engine feed system interface fuel	2%	1	0-300	100
Engine start inlet oxidizer	2%	1	0-300	100
Engine start inlet fuel	2%	1	0-300	100
Downstream of start tank quad solenoid valve package, oxidizer	2%	1	0-300	1
Downstream of start tank quad solenoid valve package, fuel	2%	1	0-300	1
Downstream of orbit trim quad solenoid valves package, oxidizer	2%	1	0-300	1
Downstream of orbit trim quad solenoid valves package, fuel	2%	1	0-300	1
Injector inlet oxidizer	2%	1	0-300	60
Injector inlet fuel	2%	1	0-300	60
Chamber pressure	2%	1	0-200	100

Table 73. Propulsion Subsystem Measurements (Continued)

Item	Accuracy	Quantity	Range (°F)	Ground Redundant Signal	Desired Sampling Rate (spm)
TEMPERATURE					
1. Pressurization System					
Helium tank	5%	1	0-150		1
2. Feed System					
Propellant tank, main oxidizer	5%	2	0-150		1
Propellant tank, main fuel	5%	2	0-150		1
Propellant inlet to engine, oxidizer	5%	1	0-150		1
Propellant inlet to engine, fuel	5%	1	0-150		1
3. Engine					
Thrust chamber case	5%	4	-65 to 1000		1
EVENTS					
1. Pressurization System					
Explosive valve actuation	5%	3		x	1
Operation of regulator shut off valves	5%	2		x	1
2. Engine					
Pintle actuator current	5%	1		x	1
Pintle actuator position	5%	1		x	1
Quad solenoid valve current	5%	4		x	1
Start and shutdown explosive valve current	5%	18		x	1
Quand solenoid valve position	5%	16		x	1

Table 74. Electrical System Energy Requirements

Loads	Form	Energy Watt-Seconds						Notes
		Pre Launch	Trajectory Corrections (3 Starts-400 sec Total Burn)	Orbit Insertion (1 Start-307 sec Burn)	Planetary Vehicle Orbit Trim (1 Start-31 sec Burn)	Flight Spacecraft Orbit Trim (3 Starts-62 sec Total Burn)	Propellant Dump	
Explosive Valves (21)	24-32 VDC		18	11		3		150 watts @ 0.01 sec per valve
Latching Regulator Shut-off Solenoid Valves (2)	18-28 VDC	18		18	36	108	18	90 watts @ 0.1 sec for each valve, each actuation
Start Tank Solenoid Valves (8)	24-32 VDC		1728	864	864	2456	~1000	18 watts each valve for 6 seconds each start
Propellant Flow Solenoid Valves (8)	24-32 VDC			864	4470	8940	~1000	18 watts each valve during engine on time
Pintle Actuator (1)	24-32 VDC			200				50 watts for 2 seconds during injector pintle positioning

Table 75. Preliminary Specification, Tanks

Propellant Supply Tanks	
Total volume (minimum):	108,672 $in.^3$
Liquid volume capacity (minimum):	106,273 $in.^3$
Maximum propellant capacity:	
Oxidizer:	13,000 pounds
Fuel:	5,000 pounds
Maximum operating pressure:	270 psia
Operating pressure (nominal):	
Orbit insertion (regulated):	225 psia
Midcourse and orbit trim (blowdown):	125 - 95 psia
Proof pressure:	375 psia
Burst pressure:	420 psia
Service life (design):	400 cycles
Weight:	118 pounds
Material:	6Al-4V titanium
Baffle volume:	250 $in.^3$
Number required per subsystem	4-(2 oxidizer, 2 fuel)
Current use:	LEM descent stage
Helium Supply Tank	
Configuration:	Spherical
Outside diameter:	40.9 inches
Internal volume (minimum):	38,368 $in.^3$
Operating pressure:	3000 psia
Rated proof pressure:	5867 psia
Rated burst pressure:	6600 psia
Maximum gas capacity at operating pressure:	45 pounds helium
Tank weight	392 pounds
Tank material:	6Al-4V titanium
Fatigue life:	50 cycles at proof pressure & 90°F
Design operating temperature range:	-35 to 160°F
Number required per assembly:	1
Current use:	NAA S&ID
Manufactured by:	Airite Div 6438-1 Electrada Corp.

Table 76. Pyrotechnic Valve—Pressurant and Feed System Isolation

Number of Valves per System:	3 Normally-closed
Operating Pressure:	0 to 4000 psia
Proof Pressure:	6000 psia
Burst Pressure:	8000 psia
Connector:	Bendix PT-06-12-8S
Squib Data:	1 amp, 1 watt - no fire
	4 amp - all fire
	24 to 32 Volt dc
	Dual Bridgewire,
	Single Cartridge
	Resistance: 1.1 ± 0.2 ohms
Material:	
Body:	304 Stainless Steel
Ram:	303 Stainless Steel
Non-metallic Parts:	Teflon
Flow Passage:	0.387 inch Diameter (minimum)
Weight:	.785 pounds
Maximum Pressure Drop at Rated Flow:	
Gaseous Helium:	8 psi
Similar to:	Pyronetics, Inc. No. 1187

Table 77. Pyrotechnic Valves—Low Thrust System

Number of Valves per System:	6 Normally-open 8 Normally-closed
Operating Pressure:	0 to 1000 psia
Proof Pressure:	2000 psia
Burst Pressure	3000 psia
Connector:	Bendix PT 06-12-8S
Squib Data:	1 amp, 1 watt - no fire 4 amp - all fire 24 to 32 Volt DC Dual Bridgewire, Single Cartridge Resistance: 1.1 ± 0.2 ohms
Material:	
Body:	304 Stainless Steel
Inlet Manifold:	304 Stainless Steel
Ram:	303 Stainless Steel
Non-Metallic Parts:	Teflon
Flow Passage:	0.387 inch Diameter (minimum)
Weight:	.785 pounds
Maximum Pressure Drop at Rated Flow:	3 psi
Similar To:	Pyronetics, Inc. No. 1187

Table 78. Pyrotechnic Valve—High Thrust System

Number of Valves per System:	2 Normally-open 2 Normally-closed
Operating Pressure:	0 to 1000 psia
Proof Pressure:	2000 psia
Burst Pressure:	3000 psia
Connector:	Bendix PC 06-16-8S
Squib Data:	1 amp, 1 watt - no fire 4 amp - all fire 24 to 32 Volt dc Dual Bridgwire, Single Cartridge Resistance: 1.1 ± 0.2 ohms
Material:	
Body:	304 Stainless Steel
Ram:	303 Stainless Steel
Non-Metallic Parts:	Teflon
Flow Passage:	1.00 inch Diameter (minimum)
Weight:	3.0 pounds
Maximum Pressure Drop at Rated Flow:	2 psi
Similar to:	Pyronetics, Inc. No. 1181

Table 79. Quad-Redundant Solenoid Valves

Number of valves per System	16 Normally-closed
Operating Pressure:	225 psia
Proof Pressure:	510 psia
Burst Pressure:	825 psia
Pressure Drop at Rated Flow (maximum):	2 psia
Flow Passage	0.387 inch diameter (minimum)
Duty Cycle:	Intermittent
Power Requirements:	
Voltage (nominal):	28 Volts DC
Power (maximum):	18 Watts/valve
Response Time, 0 to 100% open (maximum):	18 milliseconds
Synchronization:	±3 millisecond Tolerance Band
Service Life (minimum):	15,000 Cycles
Weight:	1.75 pounds
Material:	
Body:	304 Stainless Steel
Poppets:	FEP Teflon 100X
Seals:	Teflon
Springs:	302 Stainless Steel
Armature:	Armco Iron (alphatized)
Connector:	Bendix PT06CE-16-26S
Similar to:	James, Pond and Clark, Inc. V4133

Table 80. Quad-Redundant Check Valve Package

Number Required per subsystem:	16
Operating Pressure:	225 psia
Proof Pressure:	340 psia
Burst Pressure:	500 psia
Cracking Pressure, each valve:	2.0 ±0.5 psi
Reseat Pressure:	1.0 psi
Pressure Drop, at rated flow (maximum):	
Propellant:	1.0 psi
Pressurant:	3.0 psi
Material:	
Body:	304 Stainless Steel
Spring:	302 Stainless Steel
Seal:	Teflon
Flow Passage:	0.387 inch diameter (minimum)
Weight:	.125 pounds
Service Life (minimum):	8000 cycles
Similar to:	James, Pond and Clark No. P 1-754
Used on:	LEMDS

Table 81. Latching Regulator Shut-off Valve

Operating Pressure:	0 to 4000 psia
Proof Pressure:	5335 psia
Burst Pressure:	8000 psia
Flow Rate:	0 to 4.25 lt/minute
Pressure drop at Rated Flow (maximum):	8 psi
Power Requirements:	
Voltage:	18 to 28 Volts DC
Amperage (maximum):	3.0 Amps
Duty Cycle:	Closed - Open - Closed
Service Life (minimum):	4000 cycles
Materials:	
Body:	304 Stainless Steel
Armature:	Armco Iron (alphatized)
Poppet:	Teflon 100X
Weight:	2.3 pounds
Number Required per subsystem:	2
Used On:	LEMDS

Table 82. Pressure Regulator

Operating Pressure:	
Inlet:	4500 to 350 psia
Outlet:	230 to 233 psia
Proof Pressure:	
Inlet:	5320 psia
Outlet:	360 psia
Burst Pressure:	
Inlet:	8000 psia
Outlet:	540 psia
Rated Flow (He Gas):	0.43 to 4.25 lb/min
Duty Cycle:	0.45 to 4.25 lb/min Helium flow at 4000 psi inlet pressure
Service Life (minimum):	3000 cycles
Weight per Unit:	1.5 pounds
Number Required per Subsystem:	2
Used on:	LEM Descent Stage

Table 83. Fill and Vent Coupling

Type:	Manual Engagement, Disengagement, Lockup
Operating Pressure:	0 to 4000 psia
Proof Pressure:	6000 psia
Burst Pressure:	8000 psia
Filter:	75 micron
Flow Rate:	2.75 lb/min N_2 at 30 psig
Pressure Differential at Rated Flow:	8 psi
Service Life:	400 cycles
Weight per Component:	0.45 pounds
Number Required per Subsystem:	7
Used on:	LEM Descent Stage

Table 84. Relief Valve

Operating Pressure:	
Burst Disc Rupture:	272 ±12 psia
Relief Check Valve Crack:	279 ±4 psia
Reseat:	269 psia
Proof Pressure:	555 psia
Burst Pressure:	825 psia
Flow Rate (maximum):	10 lb/min He Gas
Duty Cycle:	Closed - Open - Closed
Service Life:	3000 Cycles
Line Size:	0.5 inch
Weight per Unit:	0.8 pound
Number Required per Subsystem:	2
Used on:	LEM Descent Stage

Table 85. Main Propellant Line Filter

Nominal Particle Size Rating:	40 micron
Operating Pressure:	225 psia
Proof Pressure:	555 psia
Burst Pressure:	825 psia
Flow Rate (maximum):	
Oxidizer:	21.2 lb/sec
Fuel:	13.3 lb/sec
Pressure Differential at Rated Flow:	
Oxidizer (maximum):	2 psi
Fuel (maximum):	3 psi
Service Life:	100 cycles/10 hours
Weight:	1.2 pounds
Line Size:	1.0 inch
Number Required per Subsystem:	One Oxidizer; One Fuel

Table 86. Filter—He Gas System

Nominal Particle Size Rating:	5 micron
Operating Pressure:	0 to 4000 psia
Proof Pressure:	8000 psia
Burst Pressure:	12,000 psia
Flow Rate (maximum):	4.25 lb/min He Gas
Pressure Differential at Rated Gas Flow:	10 psi
Duty Cycle:	0 to 4000 psia
Service Life:	3000 cycles/1000 hours
Weight:	0.3 pounds
Number Required per Subsystem:	1
Used on:	LEM Descent Stage

5. VOYAGER PROPULSION SUBSYSTEM PERFORMANCE SUMMARY

The orbit insertion capabilities for the 1971 and 1973 missions are compared with the JPL requirements in Figure 160. The independent variable for this figure is the midcourse velocity increment which appears as a variable since the weight at the orbit insertion maneuver initiation is less than the injected weight by the amount of midcourse propellant expended. This figure shows that the performance capability exceeds the JPL required capability level for the 1971 and 1973 missions under all conditions and the desired capability when midcourse ΔV expenditure is less than 90 meters/second.

For the 1975 and 1977 periods, orbit insertion velocity was not specified. Therefore, ΔV requirements have been computed for a range of orbits as shown in Figures 161 and 162 for the 1975 and 1977 opportunities, respectively. The independent variable is the period during

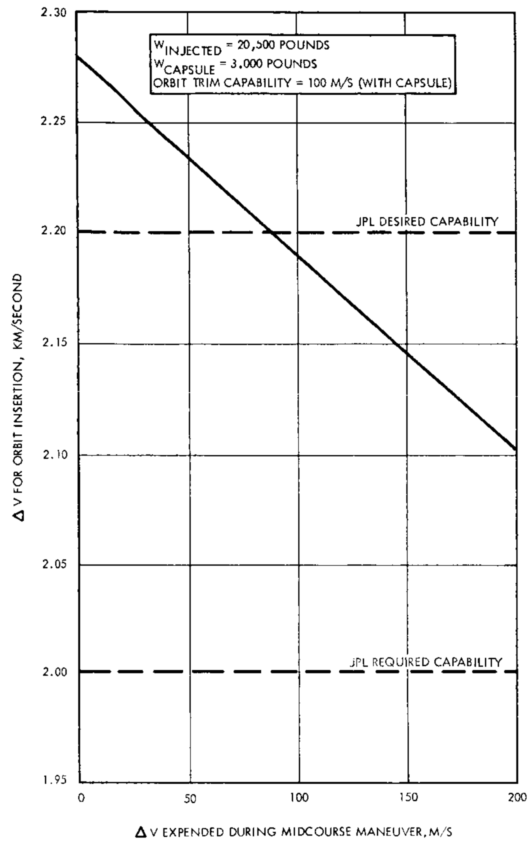


Figure 160. Velocity Increment Capabilities of Voyager Propulsion System for 1971-73

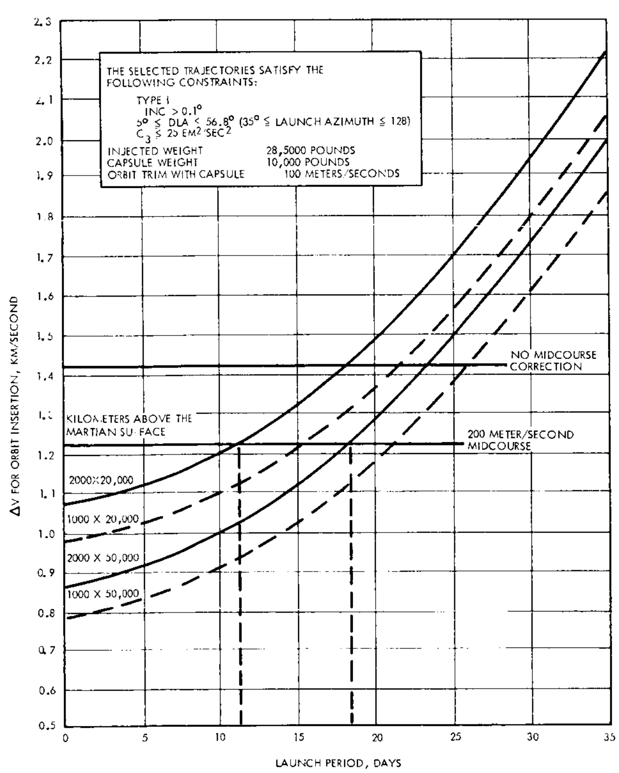


Figure 161. Velocity Increment Requirements for Martian Orbit Insertion During the 1975 Launch Interval

which launches may be attempted under the listed constraints. The figures show that the nominal 2000 x 20,000 km orbit may be obtained over a launch period of 11 days in 1975, and 17 days in 1977 assuming a midcourse correction of 200 meters/second. By allowing the apoapsis to increase to 50,000 km, an additional period of 7 days is available in 1975 and 5 days in 1977. Curves are also shown for a periapsis of 1000 km which is probably a lower bound on the range of possible values for this parameter. Due to the increased eccentricity of the lowered periapsis, a 2 to 4 day increase would be obtained over the respective orbit with the 2000 km periapsis.

If these launch periods are felt to be too short, use may be made of the longer transit time Type II trajectories, which result in launch periods which are approximately double the above values. Thus, an adequate velocity increment potential is available for all launches in the 1971 to 1977 time span.

One of the most desirable features of the single propulsion system utilizing liquid propellants is the ability of the system to apply propellants which are not used in one mission phase to a subsequent phase. This capability is illustrated in Figures 160 to 162, which show the additional ΔV_{retro} which is possible if the propellant allotted for midcourse is not completely expended. This type of flexibility is also indicated in Figure 163, where variable amounts of the orbit trim allotment are expended prior to capsule separation. The capability also exists to use part of the nominal retro ΔV capability for correction of an unexpectedly large injection error (i.e., one for which the required midcourse velocity increments exceeds 200 meters/sec). For example, if the nominal apoapsis was 20,000 km, an additional ΔV of 200 meters/sec could be made available for midcourse corrections by letting the apoapsis increase to 50,000 km. Operation of the system in the blowdown mode for an additional 200 meters/sec correction would, however, require an increase in the nominal 125 psi initial tank pressure.

Another inherent flexibility feature of the liquid propulsion system is its ability to readily accommodate variations in propellant loading or mission total impulse requirements. For example, no change would be necessary to fly a mission without the landing capsule. Since the tanks have considerable excess capacity, considerable flexibility to absorb system changes by loading additional propellants is available. Changes in mission planning can also be readily accommodated. For example, if the 1000 pound payload increase scheduled for the 1975 and 1977 launches were to be added in the form of additional propellant rather than additional payload, the retro ΔV would increase to 1.443 km/sec for 200 meters/sec midcourse and 1.656 km/sec for 0 meters/sec midcourse. The additional flexibility thus gained in mission selection may be determined from Figures 161 and 162. For example, with the additional propellant and a 200 meter/sec midcourse maneuver, launch period for the nominal 2000 x 20,000 km orbit would be 18 days in 1975 and 22 days in 1977 and increase of 7 and 5 days respectively.

The capability of the system to absorb propulsion system changes may be determined from Figures 164 and 165. Figure 164 shows the variation in retro velocity increment capability for a ±5 percent change in the retro specific impulse. Figure 165 shows the change in ΔV capability for a ±5 percent change in the inert system weight.

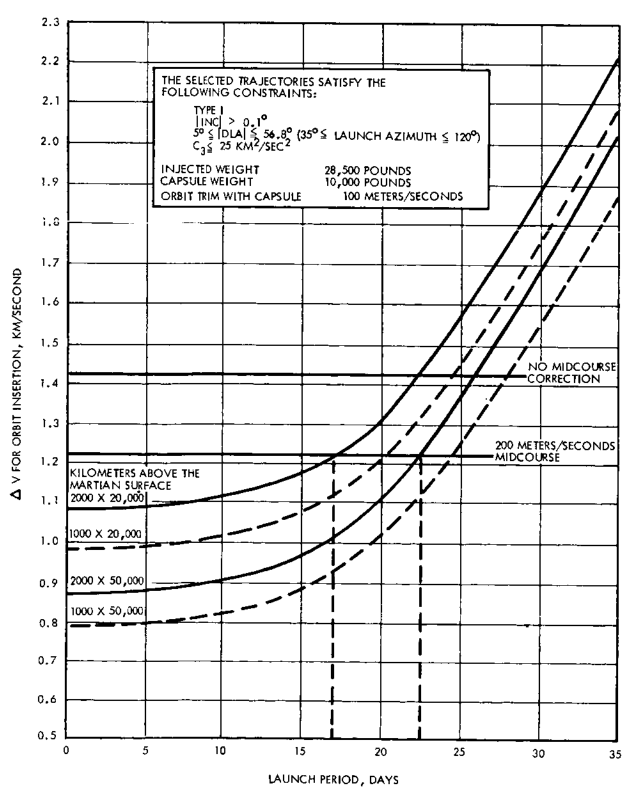


Figure 162. Velocity Increment Requirements for Martian Orbit Insertion During the 1977 Launch Interval

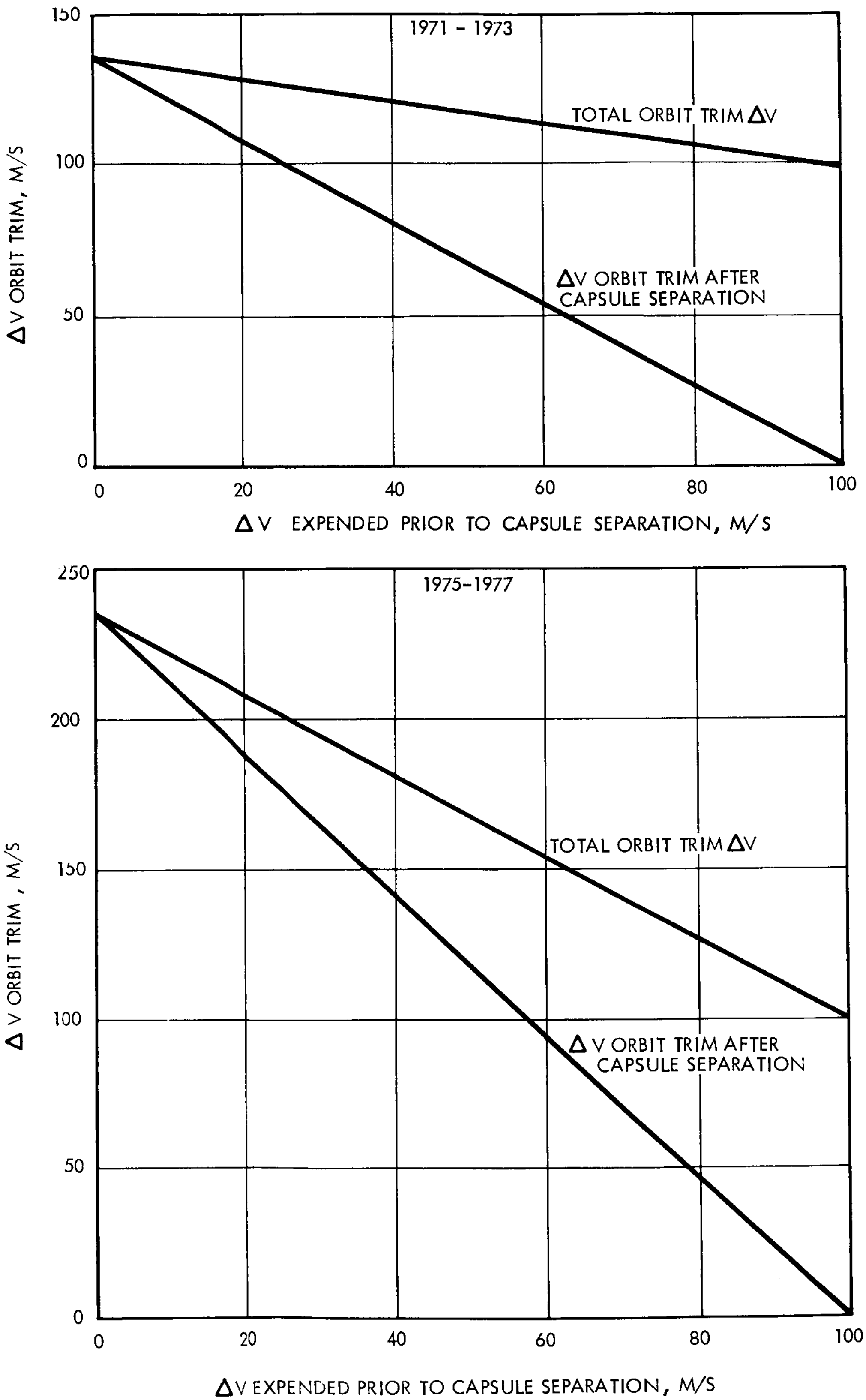


Figure 163. Orbit Trim Capabilities Prior to and Subsequent to Capsule Separation

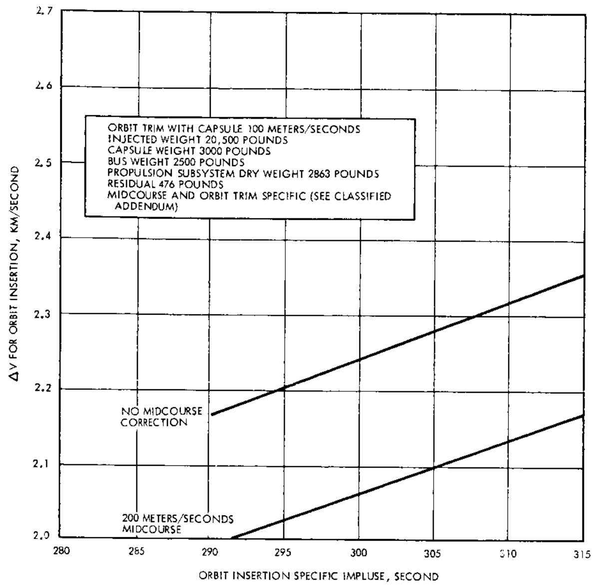


Figure 164. Effect of Orbit Insertion Specific Impulse on Orbit Insertion ΔV Requirements

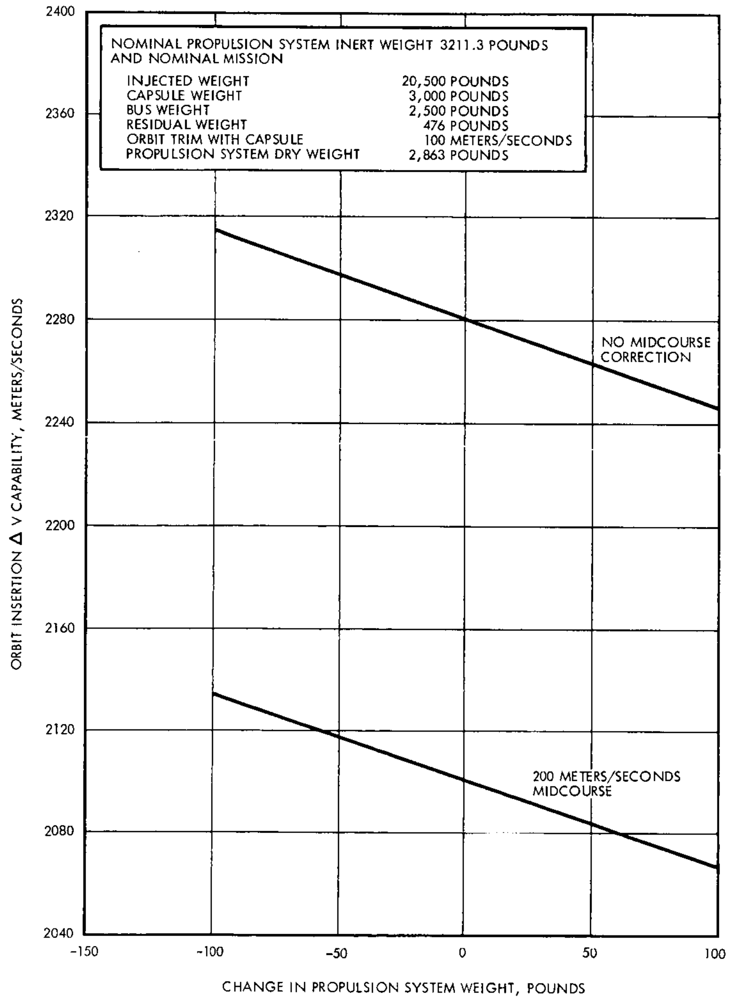


Figure 165. Orbit Insertion ΔV vs. Propulsion System Weight

5.1 Subsystem Reliability

The reliability predicted for the LEM Descent Propulsion Stage, as adapted for the Voyager Spacecraft mission, is 0.9917. This figure takes into account the redundancies which have been provided. It is based upon a 10-minth mission.

The mission profile used in the reliability analysis is presented in Table 87.

Table 87. LEM Descent Propulsion Stage Mission Profile

Maneuver	Time (Hours)
Boost	0.3
Cruise	4320 (6 months)
Midcourse Corrections (during cruise engine firing time)	0.111 (400 seconds)
Mars Orbit Injection	0.089 (320 seconds)
Mars Orbiting Period to allow for Trim	50.0
Trim Operating Time	0.055 (100 seconds)

The midcourse corrections and orbit trim operations are performed at the minimum thrust of approximately 1140 lbs and the orbit injection operation is performed at the maximum thrust level of approximately 7800 lbs.

5.1.1 Failure Rate Data

The primary failure rates data used in the above propulsion reliability analysis was obtained from JPL Memo dated 12 November 1965, entitled "Design Data for Candidate Voyager Spacecraft Propulsion Systems," and are presented in Table

In order to use the reliability figures shown in Table , it was necessary to reduce the values to generic failure rates such that the mission profile of the Voyager Spacecraft could be substituted. The mission profile and environmental (K) factors of TRW report 843-6145-SC000 were used for this purpose. The environmental (K) factors used in the determination of the failure rates are shown in Table

Table 88. LEM Descent Propulsion Subsystem Reliability Estimates for Apollo Mission (Supplied by Grumman)

Equipment	Estimate
DESCENT PROPULSION SUBSYSTEM	0.998830
Descent Engine, Variable Injector	0.999688
Propellant Press & Feed Subsystem	0.999142
Coupling, Fuel, Manual Disconnect, Fill & Drain	0.999995
Coupling, Oxidizer, Manual Disconnect, Fill & Drain	0.999995
Tank, Helium, Storage	0.999994
Filter, Helium, In-Line Non-By-Pass	0.999999
Valve, Helium, Latching, Solenoid Operated	0.999972
Valve, Helium, Explosive Operated	0.999895
Valve, Helium, Pressure Reducing	0.999946
Valve, Helium, Press. Relief & Burst Disc	0.999989
Coupling, Fuel, Manual Disconnect, Fill & Vent	0.999995
Coupling, Oxidizer, Manual Disconnect, Fill & Vent	0.999995
Coupling, Helium, Manual, Disconnect, Fill & Test Point	0.999995
Valve, Helium, Quad Check	0.999976
Filter, Fuel, In-Line, Non-By-Pass	0.999999
Filter, Oxidizer, In-Line, Non-By-Pass	0.999999

Table 89. Environmental (K) Factors

Mission Phase	Environmental (K) Factor
Component Operating	200
Component Non-Operating But Under Pressure	20
Component Non-Operating But Not Under Pressure	0.1

Table 90. Modified LEM Descent Propulsion Stage Reliability Prediction for Performing the Voyager Mission

Component	No. Used	Redundant	Mission Reliability Prediction
1. Helium Tankage (Tank, Relief Valve and Lines and Fittings)	2	No	0.9^3059
2. Filter	7	No	0.9^3741
3. Latching Solenoid Valve	2	Yes	0.9^3884
4. Squib Valve	21	No	0.9^2885
5. Regulator	2	Yes	0.9^4773
6. Quad Check Valve	4	Redundancy Within The Quad	0.99796
7. Fill and Drain Valve	7	Assumed Capped	0.99850
8. Propellant Tank	4	No	0.9^3002
9. Relief Valve	2	Burst Disc Included	0.9^3571
10. Start Tank (Bellows)	2	No	0.9^3644
11. Quad Solenoid Valve	4	Redundancy Within The Quad	0.9^5626
12. Orifice	4	No	0.9^6623
13. Cavitating venturi	2	No	0.9^6867
14. Thrust Chamber Assy (Includes injector, chamber and nozzle)	1	No	0.9^4289
15. Pintle Actuator	1	No	0.9^5415
16. Lines and Fittings	20	No	0.9^3656
TOTAL			0.99166

5.1.4 Discussion

The reliability of the major components that make up the Modified LEM Descent Propulsion State for the Voyager Mission are shown in Table 90. The total reliability of the system is calculated to be 0.99166. This is the probability of the subsystem performing the mission as required. The reader is referred to Volume V Appendix A for an alternate calculation of reliability based on the use of comparable data between the LEM, transtage, and solid propulsion options. The reliability calculated there for the selected LEM based propulsion system is 0.973. The difference between the number and the current calculation of 0.99166 may be attributed to lower failure rate assumptions by Grumman. Since lower rates are appropriate to the additional development and testing associated with a manned mission, some benefit of this development and testing should accrue to a LEM based Voyager.

In either case, the high reliability assessment of the subsystem is attributed to system simplicity and the selective use of redundancy, i.e., quad redundant check valves and solenoid valves, plus redundant regulators. In addition, explosive valves are used during the midcourse phase (4320 hrs) to "lock-up" the system after midcourse corrections. This protects against the leakage failure mode. Further, it was assumed that the midcourse corrections could be accomplished through the quad solenoid valves in case of failure of an explosive valve.

A major portion of the figures in the table are associated with the leakage mode of failure which would use gas and/or propellant at a rate that would not allow completion of the mission. It is recognized, however, that all leaks would not result in a catastrophic failure but would be dependent on leakage rate and the time in the mission where leakage began. Although a detailed failure mode and effect analysis was not performed, it was assumed that the effect was accounted for in the environmental (K) factors listed in Table 89.

Arrangement of the components in the propulsion subsystem was based on providing alternate capability in the event of component failure. Components used throughout the assembly were selected on the basis of proven operational capability and reliability, which reduces the probability that alternate modes of operation would be required during completion of the mission. In the event of failure, however, the subsystem will not be rendered inoperative. Those components with the least history of operation after long exposure to the space environment have been duplicated in the system through redundant paths. A discussion of alternate modes as a result of typical component malfunctions is included below.

In the event of failure to open one of the solenoid valves, in the start or low thrust system, an alternate path containing redundant solenoid valves has been included in the subsystem. In the event of failure of a normally-closed explosive valve to open, an extra set of explosive valves has been included in the propellant feed system for oxidizer and fuel. The solenoid package also provides an additional back-up alternate routing for flow to the thrust chamber.

Reliability of the explosive valves and solenoids used in the system is in the order of 0.9999 and with the alternate paths, the reliability of the system is increased to 0.999999.

Check valve malfunctions have been compensated for by use of a series-parallel arrangement. No single malfunction can cause engine failure.

Latching regulator shut off valves are included in the present LEMDS pressurant lines to allow for shutoff of the gas flow through a failed regulator. Parallel regulators and parallel latching regulator shutoff valves are included in the system to increase overall reliability. The regulators are isolated prior to initiation of the orbit insertion maneuver and are only required to function for a short period of time. An additional safety feature is included in the form of relief valves in the LEMDS situated in the pressurant lines downstream of the regulators. In the event of regulator failure and regulator shutoff valve failure, the pressure will be decreased by bursting a diaphragm and allowing gas to flow through a relief valve to prevent catastrophic tank failures.

5.2 Subsystem Weight Summary

Table 91 provides a weight breakdown of the propulsion subsystem.

Table 91. Spacecraft Propulsion Weight

Item	Weights, lb	
Structure and Mechanical		923.6
Primary structure	671.5	
Reaction control supports	11.7	
Meteoroid protection	226.1	
Attachments and miscellaneous	14.3	
Temperature Control		73.9
Insulation	68.8	
Heaters and thermostats	2.0	
Attachments and miscellaneous	3.1	
Engine		586.1
Injector	29.3	
Combustion chamber assembly	421.6	
Injector pintle actuator	4.0	
Propellant lines and ducts	13.9	
Cabling set	11.0	
Instrumentation	7.4	
Gimbal assembly	27.2	
Hardware-engine integration	9.4	
Trim orifice-fuel	1.8	
Trim orifice-oxidizer	1.8	
Cavitating venturi (2)	2.0	
1-inch explosive valve (4)	12.0	
3/8-inch explosive valve (14)	11.2	
Solenoid valve (16)	28.0	
Quad check valve (2)	1.0	
Trim orifice (2)	0.5	
Filter (4)	4.0	
Propellant Feed System		528.6
Pressurization System		438.4
Valves, regulators, etc.	33.2	
Plumbing and fill and vent	13.2	
Tank (1)	392.0	
Tank and Engine Supports		161.8
Start Tank Assembly		22.6
Contingency		128.0
Spacecraft Propulsion Dry Weight		(2863)
Spacecraft Propulsion Inert Fluids		(476)
Residuals, propellant feed	424.1	
Residuals, start tank	12.2	
Helium	40.0	
Spacecraft Propulsion Inert Weight		3339

IV. SPACECRAFT SCIENCE SUBSYSTEM

The functional interface requirements for the science subsystem have been discussed in detail in Volume 1, Section III, 9., as requirements on the spacecraft bus. In order to provide supporting data for the more detailed interface considerations contained in this volume, a summary description of a hypothetical science subsystem is given here from the standpoint of the science subsystem designer. In most cases the results of the Task A study have been assumed to apply, since the typical payload is similar to the hypothetical science payload of Task A. The subsystem and its functional interfaces are shown in Figure 166.

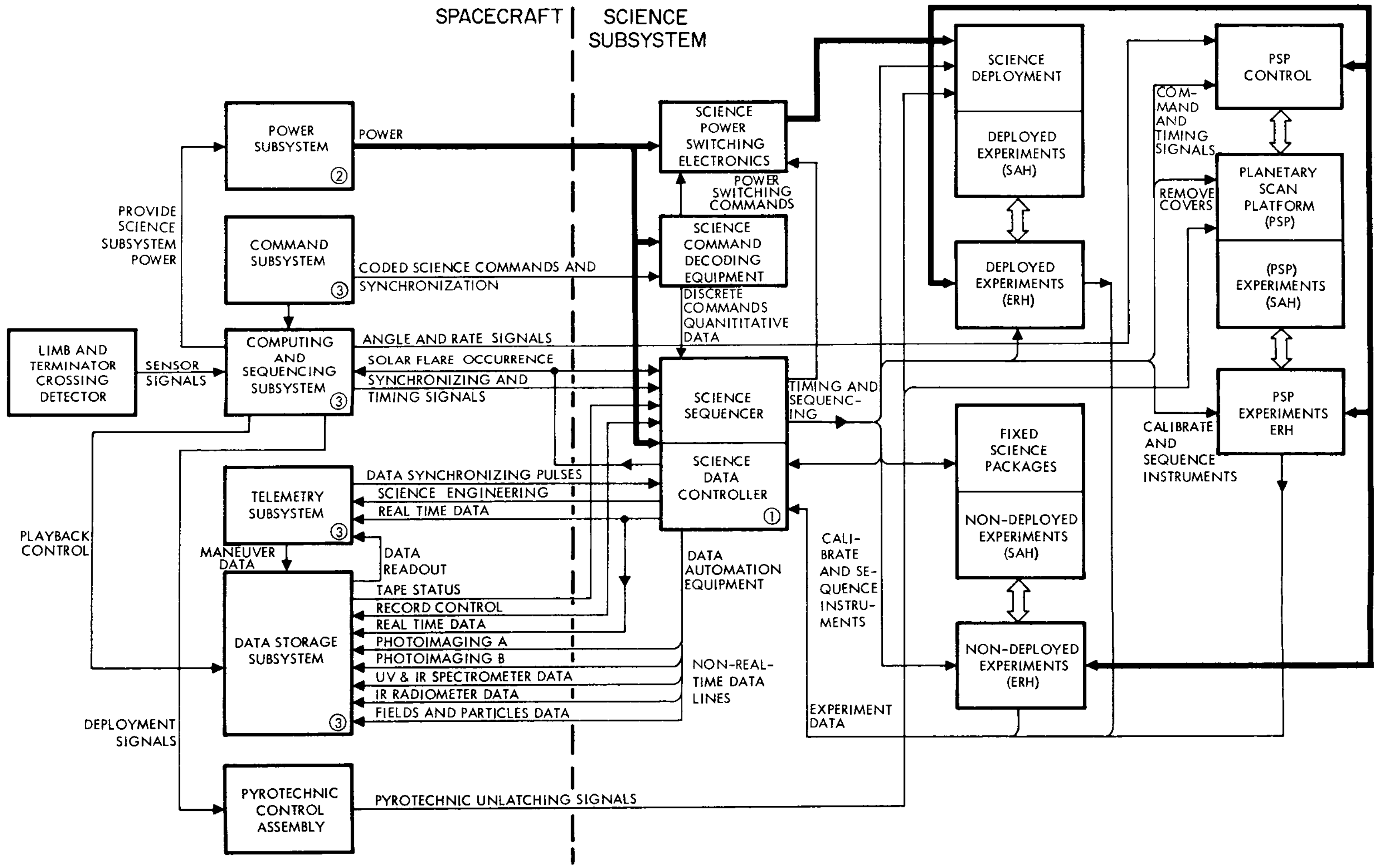
1. SUBSYSTEM ELEMENTS

The science subsystem is made up of equipment of the classes listed below. Detailed identification of the science sensors and their size, weight, power consumption, etc., are listed in the table of component design parameters in Volume I.

- Science experiment equipment is that furnished by the experimenters for integration into the science subsystem. The equipment consists of sensors, associated hardware, and experiment remote hardware. Sensors and associated hardware are those elements which sense the physical phenomena involved in the experiments, and which consequently require proper view angles, pointing, etc. Experiment remote hardware are those parts of the equipment which can be mounted apart from the sensors as a convenience in packaging. These are mounted on the spacecraft equipment mounting panels. (See section II-12)

- Planetary experiments are those experiments which require pointing at known directions with respect to the center of Mars. Deployed experiments are those which require that their sensors be positioned or erected so as to achieve adequate antenna patterns, adequate view angles, or adequate isolation from spacecraft effects. Non-deployed experiments are those which can be mounted on, or in a fixed relation to, the spacecraft structure.

Data automation equipment (DAE) will be capable of accepting binary-coded commands and data from the science command decoder and timing signals and initiation signals from the computing and sequencing subsystem (including initiation signals derived from the terminator or limb crossing



① SCIENCE ENGINEERING DATA LINES OMITTED FOR CLARITY

② POWER TO OTHER SUBSYSTEMS OMITTED FOR CLARITY

③ SEQUENCING OF SPACECRAFT SUBSYSTEMS OMITTED FOR CLARITY

SAH SENSORS AND ASSOCIATED HARDWARE

ERH EXPERIMENT REMOTE HARDWARE

⇔ DATA, POWER, AND CONTROL LINES

Figure 166. Spacecraft Science Subsystem Functional Interfaces

sensors). Based on stored science sequences and commands, the DAE will carry out the detailed steps necessary to implement the sequences. The science sequencer in the DAE accepts timing and signals from the C and S subsystem to control the experiment and the transfer of data not in real time to the data storage subsystem. The science sequencer will accept revisions to its stored sequences from the science command decoder.

The science data controller in the DAE converts or controls the conversion of science data and subsystem engineering data to digital binary form in a format compatible with the telemetry subsystem format. Output of the non real-time data will be provided as follows:

- One line for non real-time photoimaging data A
- One line for non real-time photoimaging data B
- One line for non real-time UV and IR spectrophotometric data
- One line for non real-time IR radiometric data

The format for these data will include 6 per cent gaps distributed throughout the data to provide time to insert other telemetry during playback from the data storage subsystem as described in Section II-6. Output for real-time data will be provided as follows:

- One line for real time fields and particles data. This line may be switched to the data storage subsystem either during solar flare mode, or for certain maneuvers, for subsequent playback not in real time.

- One line for science subsystem engineering data.

The science command decoder accepts binary digital command words and synchronizing signals from the command subsystem. It decodes and distributes the science commands within the science subsystem.

Science power switching electronics accepts the discrete commands from the data automation equipment, or from the C and S subsystem, and power from the power subsystem and energizes the appropriate elements of the science subsystem.

Planetary scan platform and control consists of a deployable gimballed platform which provides attachment, alignment, view angles, pointing, thermal control, and isolation for the planetary sensors and associated hardware together with their control electronics

Fixed science packages provide attachment, alignment, view angles, thermal control, and isolation for the non-deployed experiments.

Science deployment devices provide for the requirements of deployed experiments together with protective covers for actuators where required.

Science interconnecting wiring is all wiring and connectors running between elements of the science subsystem.

2. PLANETARY SCAN PLATFORM AND CONTROL

The PSP, illustrated in Figure 167 is made up of four principal elements: the science equipment platform, the gimbal assembly, the deployment mechanism, and the Mars tracker.

2.1 Science Equipment Platform

The science equipment platform consists of an enclosure which provides for the proper mounting of science instruments, the Mars tracker, and other sensors that require articulation with respect to the spacecraft. Means are provided for accommodating fields of view, protection from micrometeroids and radiation, thermal control, vibration and shock isolation, and electrical signals and power for all science instruments.

2.2 Gimbal Assembly

The gimbal assembly provides the equipment platform with two axis of rotation as shown in Figure 167. The drives for each axis, servo motors and gear trains, are equipped with position increment pickoffs and tachometers. Cablewraps are used to provide electrical paths through rotating gimbal members, and gimbal locks are provided to secure the platform when stowed.

2.3 Deployment Mechanism

The PSP is deployed by a spring-operated hinge at the base of the gimbal boom. Deployment occurs immediately upon release of squib-operated locks which secure the gimbal fork to the spacecraft. Deployment shock loads are absorbed by a damper on the deployment hinge.

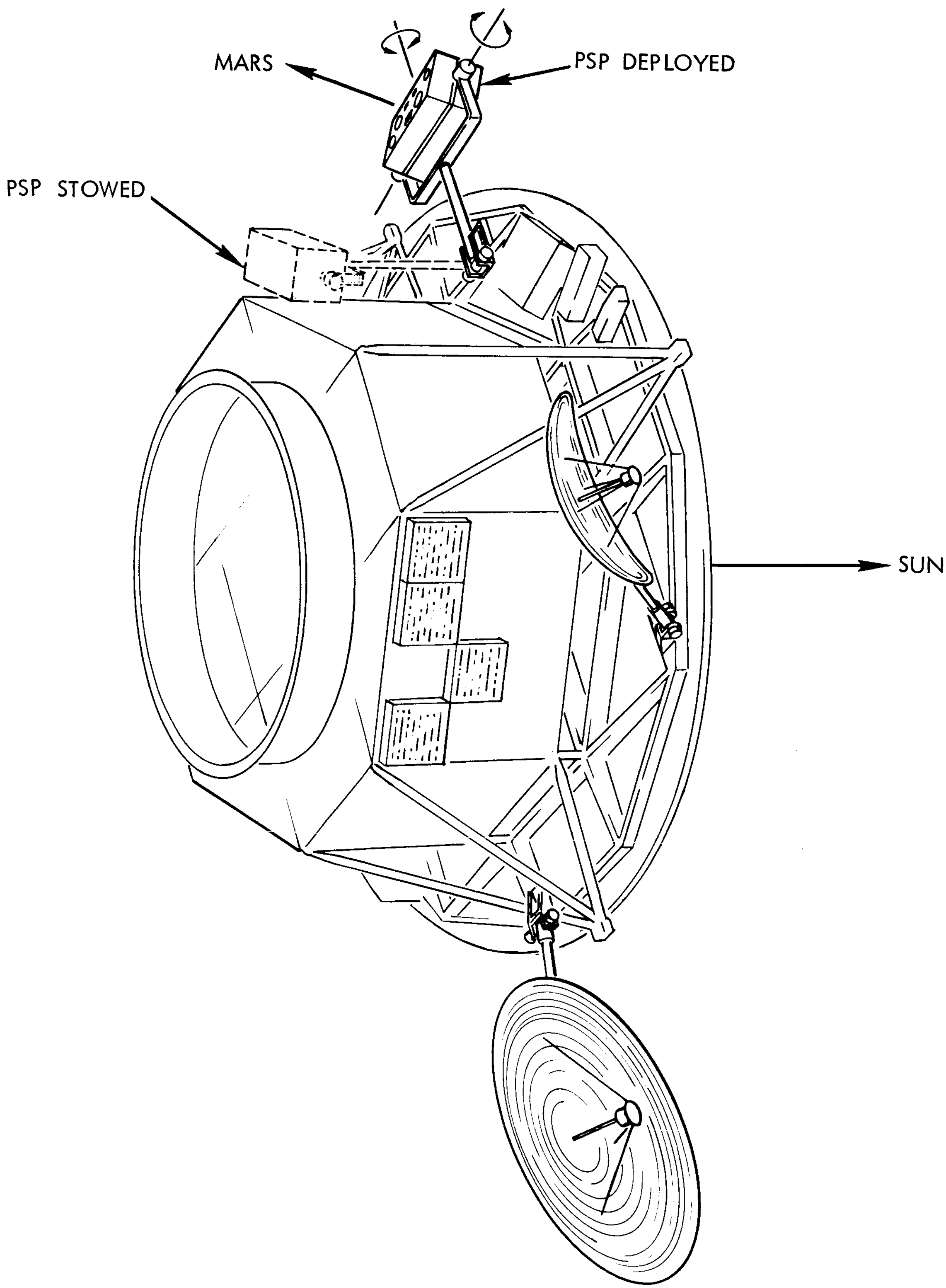


Figure 167. PSP Gimbal Assembly

Once deployed the PSP gimbal boom is secured in place by a spring operated latch.

2.4 Mars Tracker

As mentioned above, the Mars tracker is used for Mars tracking mode of the PSP. The tracker provides error signals in two coordinates for indicating a departure from the planatary local vertical. Obtaining null signals representing positions other than the local vertical is accomplished by means of biasing the tracker's output.

2.5 PSP Control

A single-channel block diagram of the PSP electronics is shown in Figure 168. The PSP will have three pointing modes:

- Mode 1: Gimbal Angle Command. The PSP is pointed in selected directions based on gimbal angle commands from the DAE or C and S.

- Mode 2. Mars Tracking. This mode is normally entered from Mode 1; it utilizes error signals from a Mars tracker to maintain pointing along the planet local vertical. For pointing in directions away from the local vertical the output of the tracker is biased accordingly.

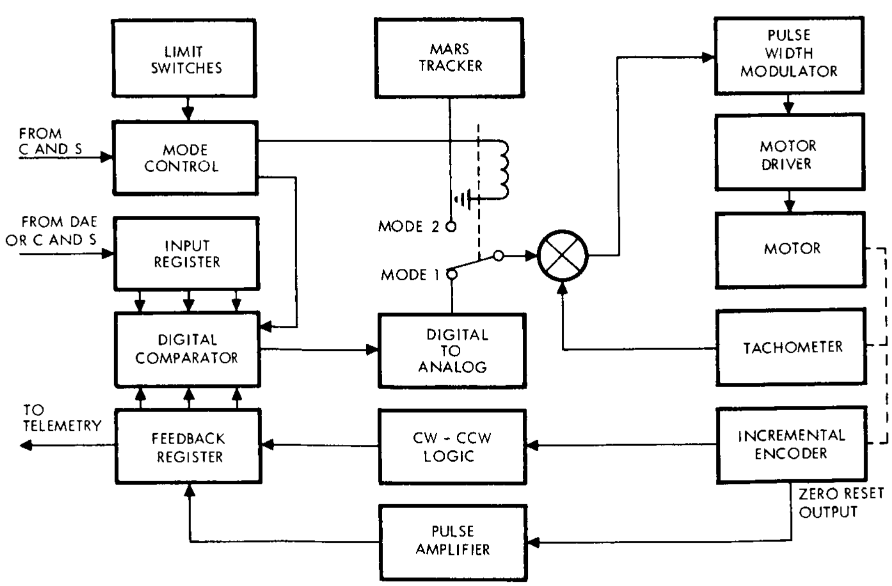


Figure 168. PSP Drive Electronics Block Diagram

- Mode 3: Image Motion Compensation. During photographic sequences, this mode provides image motion compensation by causing the boresight axis of the PSP to point at a fixed point on the planet surface. Angles and/or angle rate commands are issued by the C and S subsystem. Normally, Mode 3 would be entered from Mode 2 just before photographic exposure to provide image motion compensation during the exposure period.

2.5.1 Gimbal Angle Command

The DAE or C and S commands gimbal angles by loading an input storage register. A digital comparator then compares the output of the input register to that of a position feedback register. The output of the digital comparator is converted to analog form to control a pulse width modulator which provides the motor driver with on-off control. The motor driver then gates power to the gimbal drive motor.

Gimbal position feedback is obtained from an incremental encoder mounted on the gimbal axis trunnion. The output pulses from the encoder are entered into the feedback register, which constantly accumulates the absolute gimbal angle. This information is then used for comparison in the digital comparator and for telemetry. Loop stability is provided by rate feedback signals received from a tachometer.

Since the use of an incremental encoder requires that the feedback register be loaded serially false gimbal angle indication may occur if a pulse is lost. To minimize this problem, a feedback reset pulse is provided from the encoder to reset the register each time the encoder passes through null.

2.5.2 Mars Tracking

During Mars tracking, the output of the Mars tracker is fed into the pulse width modulator to cause the PSP to track the center of Mars. The pulse width modulator and motor driver function as in Mode 1. For pointing in directions away from the planet local vertical (if required) a bias signal can be summed with the Mars tracker output by removing the feedback and inserting a bias signal in the command register.

2.5.3 Image Motion Compensation

During image motion compensation the inputs are initial angle settings followed by a rate signal from the DAE or C and S. This rate signal is obtained by opening the encoder feedback and inserting a rate command in the input register. This signal is summed with the tachometer feedback to produce the desired angular motion.

For all modes of operation, the total gimbal angle is obtained from the feedback register for telemetry back to earth.

3. PSP REQUIREMENTS AND DESIGN CONSTRAINTS

3.1 Mission Constraints

Since the exact definition of science instruments may occur late in the design phase, and since science equipment may vary from one mission to another, the PSP should be flexible enough to accommodate a variety of instruments without requiring major redesign. This felxi-bility may be provided by the following:

- Instrument mounts: standardize instruments mounts or removable equipment shelves which may be tailored to a specific complement of instruments.
- Cabling and connectors: excess capacity in cable harnesses, cable wraps, and connector banks to provide for growth.
- Aperture panels: removable aperture panels mounted on the walls of the equipment platform enclosure which may be individually tailored to fit a select group of instruments.
- Platform balancing: flexibility in the location of instruments to avoid excessive gimbal moments and to minimize the use of counterweights.

The recommended Task B spacecraft configuration requires that the PSP be deployable to prevent obscuring its field of view by the solar array and to minimize the blocking caused by the bottom portion of the canister. In Task A the capsule canister could be jettisoned, and was not present during Mars orbit so the PSP required no deployment. Figure 169 illustrates the differences between the Task A and Task B PSP configurations.

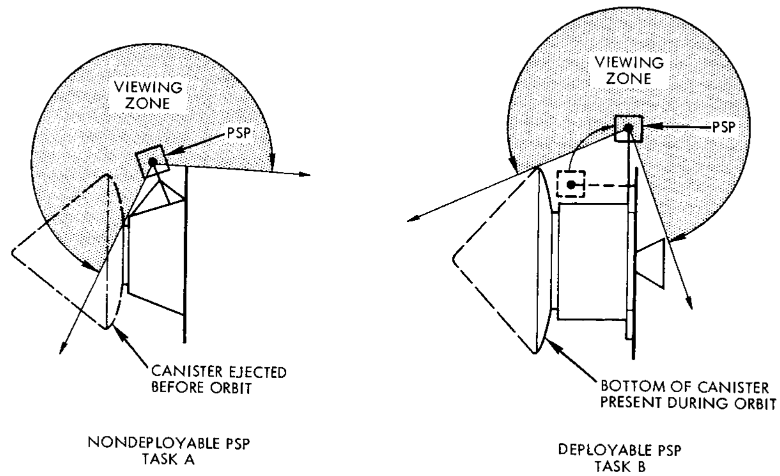


Figure 169. Differences Between Task A and Task B PSP's

Due to the restrictions regarding the contamination of Mars, the PSP design must lend itself to being sterilized according to the process required for the spacecraft.

3.2 Performance Requirements

The PSP must meet the requirements given in Volume 1, Section III, 9.

Gimbal travel shall be $\pm$ 190 degrees about the boom axis and $\pm$ 150 degrees about the platform axis. The minimum angular acceleration about each axis must be 0.6 mr/sec^2. The gimbal axis must be aligned relative to the spacecraft coordinate system within 0.25 degree, and the gimbal axes incremental position encoders must be capable of detecting gimbal position change of 0.1 degree.

The tachometer rate accuracy must be within 1 per cent of the absolute indicated rate, or 0.1 mr/sec, whichever is greater. The Mars tracker must operate over altitudes from 500 to 30,000 km and be capable of detecting the center of Mars to within 0.9 degree.

Total errors of the PSP drive electronics can result in uncertainties no greater than 0.1 degree in gimbal angle and 0.05 mr/sec in gimbal rate.

3.3 Design Requirements

The platform will be capable of accommodating 100 pounds of science instruments and directly associated hardware, in addition to the Mars sensor. It will provide a minimum of 12 cubic feet of volume for housing this equipment and the minimum usable mounting area for planetary sensors and directly associated hardware will be 6 square feet.

The science equipment compartment temperatures must be maintained within the limits of zero to $80^{\circ}F$ during science equipment operation. The maximum temperature gradient between any two points on the platform instrument mounting surface will not exceed $1^{\circ}F/ft$.

The total weight of the PSP, not including experiments, will not exceed 104 pounds, divided as follows:

Science equipment platform	42 pounds
Gimbal assembly and deployment mechanism	50 pounds
Mars tracker	12 pounds
Total	104 pounds

All high speed moving elements of the gimbal drives will be sealed in a pressurized inert atmosphere, and the gimbal drives must be able to withstand stalled conditions without damage.

A maximum of 14 watts of secondary spacecraft power will be required. Electrical paths through the gimbal will use cable wraps. The cable wraps will accommodate a minimum gimbal rotation of ±190 degrees.

The PSP is mounted to the spacecraft solar array structure as shown in Figure 167. The structure provides a mounting pad to which the PSP deployment hinge is attached. In the stowed position, the PSP gimbal fork is secured to the spacecraft structure by means of a locking mechanism released by a pyrotechnic pin puller just after deployment (see Section 9.4.)

The following engineering measurements are provided for telemetry:

- Science platform temperatures (10)
- Gimbal drive temperatures (4)
- Gimbal angle positions (2)
- PSP deployment indication (1)

- Gimbal unlock indications (2)
- Aperture cover removed indications (5)
- Sealed drive internal pressures (2)

4. FIXED SCIENCE PACKAGES

The fixed science packages contain the nondeployed science sensors and associated hardware. The packages provide the science instruments with a protective, controlled environment, apertures for accommodating fields of view, and the necessary cabling to other elements of the science subsystem. Two packages are provided with locations as shown in Figure 14, Volume 1.

4.1 Components

The sensors and associated hardware within the packages are as follows with quantities as indicated:

Package A	Number of Sensors
Cosmic Ray Telescope	2
Trapped Radiation Sensor	3
Gamma Ray Sensor	3
Plasma Probe	2
Cosmic Dust Detector	3
Package B	
Cosmic Ray Telescope	1
Cosmic Dust Detector	1

The primary elements in each package are as follows:

- Instrument Mounts. These mounts provide rigid mechanical support and alignment provisions for the science instruments.

- Protective Enclosures. Each package is completely enclosed, except for field of view apertures, to provide science instruments with a uniform thermal environment and protection from micrometeoroids and radiation.

- Thermal Control. Means are provided for maintaining the science instruments within proper temperature limits through the use of thermal insulation blankets, heaters, and radiating surfaces as required.

- Cabling and Connectors. Cable networks and connector links provide the electrical interface between the science instruments and other protions of the science subsystem.

- Apertures and protective Covers. Each package is provided with apertures for accommodating fields of view; deployable aperture covers are provided for the protection of science equipment.

4.2 Mission Constraints

The exact definition of science instruments is likely to occur during the late design phases of the fixed science packages. In addition, the instruments for one mission will probably be different from another. The fixed science packages will have to be flexible enough to accept a variety of different instruments with minimum redesign by providing the same design approach indicated for the PSP.

4.3 Design Requirements

The fixed science packages must be capable of accommodating a minimum of 19 pounds of science instruments and directly associated haredware. Means will be provided for accommodating fields of view, protection from micrometeoroids and radiation, thermal control, alignment, vibration and shock isolation, and electrical signals and power for all science instruments.

The internal temperature of fixed science packages will be maintained within the limits of zero to $80^{\circ}F$ during science equipment operation. The weight of the fixed science packages, excluding science instruments, will not exceed the following:

- Package No. 1: 15 Pounds
- Package No. 2: 5 Pounds

The internal volume of the fixed science packages will be at least:

- Package No. 1: 13 ft^3
- Package No. 2: 0.95 ft^3

Means will be provided for venting all excessive differential pressure buildups occurring up to time of shroud separation which could cause damage to the fixed science packages or other equipment.

4.4 Functional Interfaces

The fixed science packages will provide the science instruments with the necessary cabling for signals and power for accomplishing the required electrical interface to other portions of the science subsystem.

Engineering measurements for telemetry consist of monitoring. Instrument compartment temperatures, and the requirements for the sensors and associated hardware are given in 9., Section III, Volume 1.

At present the fixed science packages are shown located on the backside of the solar array panel at clock angles of 90 and 270 degrees (see Figure 14, Vol. 1. The instruments in Package No. 1 that require viewing towards the sun look through cutouts in the solar array panel. This concept is based on a solar cell arrangement which permits cutouts of sufficient size to be placed along the straight portion of the solar array panel segments. If it becomes necessary to place solar cells in the area where cutouts are planned, those instruments which require viewing towards the sun will be relocated. If required, alternative locations and methods for accommodating the fields of view will be determined during Phase IB. Possible alternate solutions are as follows:

- Locating the instruments separately over the cutouts existing on the perimeter of the solar array
- Locating the instruments separately further inboard along the edge of the solar array segments
- Housing the instruments in a deployable package.

V. PLANETARY VEHICLE ADAPTER

1. GENERAL DESCRIPTION

The planetary vehicle adapter includes all structure, cabling, and hardware between the planetary vehicle inflight separation joint and the associated points of attachment to the nose fairing. The adapter, as shown in Figure 170, consists of the following elements:

- Main frame
- Intermediate frame No. 1
- Intermediate frame No. 2
- Adapter fittings (4)
- Shroud support fittings (4)

In the selected adapter design, spacecraft loads are introduced at four points. The necessity to distribute loads into the shroud dictates the need for continuous rings at the shroud attach points. The necessity to open the shroud dictates the need for parting joints in the rings. The shroud support fittings are designed to distribute the planetary vehicle loads into the shroud uniformly through selection of the length of the fittings, and the use of doublers and a shear-resistant panel design. The adapter attach points are located radially as close to the nose fairing as possible to minimize the moments introduced in the nose fairing and to permit a maximum diameter solar array.

2. REQUIREMENTS AND CONSTRAINTS

2.1 Use of Saturn V

The natural and induced environments associated with Saturn V are indicated in Table 92. These loads must be transferred to the launch vehicle through the adapter.

The adapter will attach to the Saturn V shroud and support a single planetary vehicle during all load conditions from preflight through launch vehicle separation. Since two planetary vehicles are positioned in tandem within the nose fairing, two adapters are required.

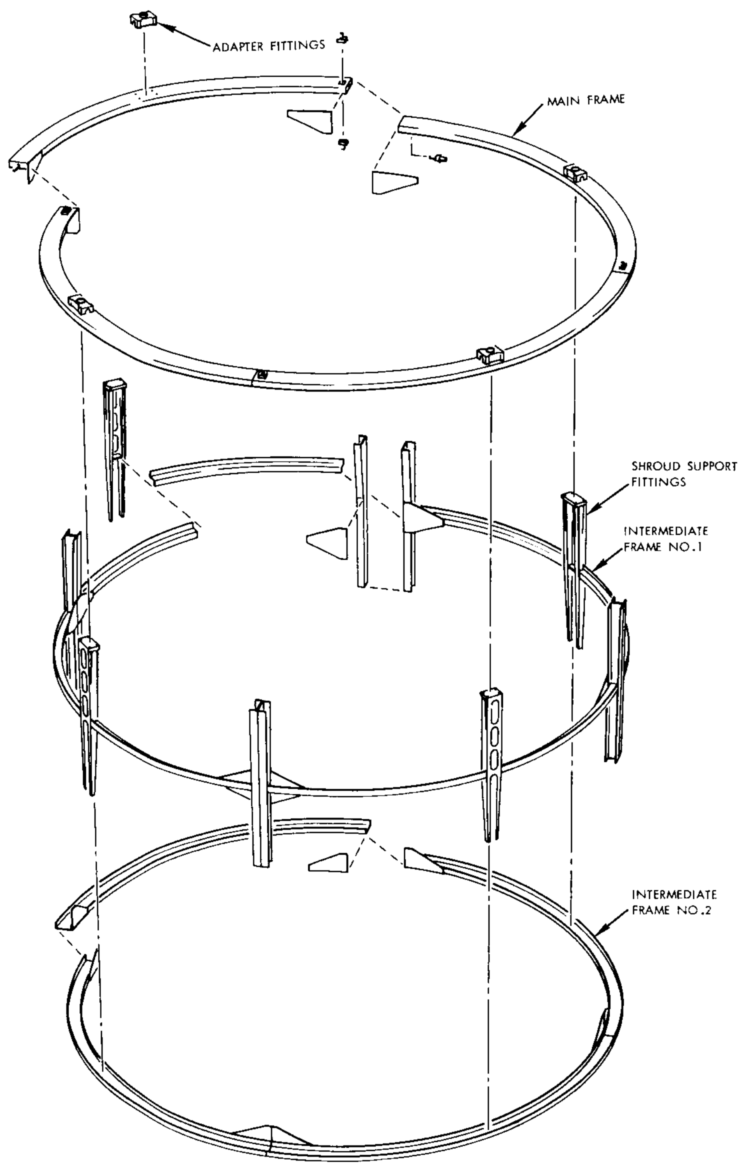


Figure 170. Planetary Vehicle Adapter, Exploded Detail View

Table 92. Static and Dynamic Load Criteria

Flight Event	Static - g		Dynamic - g(o-p)	
	Axial	Lateral	Axial	Lateral
Launch release	1.5	≤ 0.5	±3.6	±0.45
End of boost (S-IC)	5.6	≤ 0.5	-	—
Cutoff (S-IC)	—	—	±2.00	≤ ±0.5

2.2 Separation Sequence

The Voyager separation sequence requires the portion of the shroud enclosing the aft planetary vehicle to fold back, providing clearance for separation. The forward planetary vehicle adapter must split and fold back with the shroud, thereby requiring a separation capability within the adapter. For the selected adapter design, these operations require the main frame (ring) to be segmented to match the nose fairing segments. Mechanical devices join the frame segments to form a continuous member. The separation of these devices must take place prior to or during shroud foldback.

Maximum weight of the adapter will be 1500 pounds. Limit loads are multiplied by a hazard factor to obtain design loads. Design loads when multiplied by an ultimate factor of safety become ultimate design loads. The following factors of safety are the minimum values to be applied for general structure:

Hazard factor - 1.0

Ultimate factor of safety - 1.25

Margins of safety are to be computed at both limit and ultimate load levels. All structures will have a positive margin of safety, i.e., margin of safety ≥ 0 which will be computed in accordance with MIL-HDBK-5 procedures. The load at the attachment to the nose fairing will be uniformly distributed into the nose fairing through the support fittings. Relative separation of the adapter joints must be prevented during periods of negative acceleration. The rigidity of the adapter must be such that the deflections of the planetary vehicle do not violate the fairing dynamic envelope.

All materials and processes used in the adapter will be selected on the basis of suitability for application and reliable performance, and will conform to specifications and standards selected from released lists of NASA, military, industry, and company specifications and standards, in that order of preference if there is conflict. If the requirements of specifications and standards selected are less stringent than those imposed by the application and reliability goal specified for the part, material, or process covered, the more stringent requirement will apply. Materials and processes selected will be subject to approval by JPL.

Design simplicity, conservatism, and testing will characterize the structural design to increase reliability. State-of-the-art concepts, materials, and analytical techniques will be used throughout the development of the adapter design.

The adapter design will provide a field joint that is below the inflight separation joint. No inflight electrical disconnects will be required. A preflight disconnect and actuating mechanism will be incorporated into the adapter design. Only simple mechanical alignments and no critical electrical alignments will be required at matchmate. Environmental transducers will be incorporated without requiring an inflight disconnection.

3. FUNCTIONAL INTERFACES

3.1 Cabling and Umbilicals

All interface cabling between the planetary vehicle and launch vehicle will be mounted to and supported by the adapter. A single umbilical disconnect fitting will be provided at the separation plane.

3.2 Electrical Bonding

The structural interface between the planetary vehicle and the nose fairing will have a conductive finish on the faying surfaces to provide a low impedance path for planetary vehicle grounding.

3.3 Structural

All planetary vehicle loads are transmitted to the launch vehicle through the adapter. Attachment is at the four outrigger fittings at the field joint interface. The planetary vehicle loads are transmitted through four fittings, located 90 degrees apart around the circumference of the shroud, transferring the loads uniformly into the shroud over the full length of the fittings.

The adapter will provide an interchangeable attach pattern at the field joint. The inflight separation joint between the planetary vehicle and the adapter and the field joint between the four adapter fittings and the main adapter structure will provide alignment within the required tolerance.

3.4 Engineering Measurements

Both strain gage and accelerometer measurements will be made on each planetary vehicle adapter. The load distributions on each adapter fitting will be measured during critical flight periods. Load data derived from the strain gage data will be correlated with low-frequency and high-frequency accelerations, also measured on the adapter. Low-frequency acceleration and strain gage data will be used to establish the mechanical impedance characteristics of the adapter structure for proper test simulation on future spacecraft tests and to verify the design and test criteria requirements specified for the 1971 mission. High-frequency type accelerometers will be used to verify random vibration inputs to flight spacecraft during liftoff and transonic flight. High-frequency transient accelerations, resulting from pyrotechnic firings, will also be measured for verification of design and test criteria for future missions. These data will be transmitted via the launch vehicle telemetry subsystem.

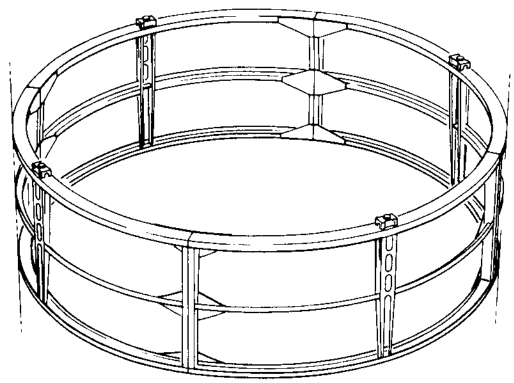
4. DESIGN DESCRIPTION

The proposed planetary vehicle adapter is attached to the launch vehicle shroud. This selection is based on the fact that the modified LEM outriggers provide attachment close to the shroud diameter and do not introduce high local bending moments. By attaching directly to the shroud, a significant weight advantage is obtained over designs using a completely independent adapter design. This design also provides a larger clearance during planetary vehicle separation. Loads are transmitted to the nose fairing portion of the adapter through four intermediate fittings.

The general arrangement, as shown in Figure 171, consists of the four adapter fittings, four support fittings, a main frame, and two intermediate frames. The loads are introduced into the shroud uniformly over the 81.6-inch length of the four main support fittings. Side and torsional loads are transmitted to the shroud through the main frame.

PRELIMINARY SPECIFICATION

Planetary Vehicle Adapter



Subsystem Function

Provide the structural support and attachment of the planetary vehicle to the launch vehicle

Subsystem Characteristics

The subsystem is composed of four adapter fittings, four shorud attach fittings, a main frame and two intermediate frames

Performance Characteristics

LOAD FACTORS	LONGITUDINAL		LATERAL	
	Static	Dynamic	Static	Dynamic
Liftoff	1.5	±3.6	≤0.5	±0.45
1st Stage Burnout	5.6	—	0.5	—
1st Stage Cut Off	—	±2.5	—	≤±0.5

FACTORS OF SAFETY

	Yield	Ultimate
Gen. Structure	1.00	1.25

Intercahgeable and aligned support points

Physical Characteristics

Configuration

COMPONENT	OVERALL DIMENSIONS	WEIGHT	MAT'L AND CONSTRUCTION
Adapter Fittings	3 x 5-1/2 x 10		7075-T73 Mach. Fitting
Support Fittings	6 x 9-1/2 x 60		7075-T73 Mach. Fitting
Main Frame	6 x 1-1/4 x 260 dia		7075-T73 Mach. Channel Frame
Intermediate Frames:			
No. 1	4 x 1 x 260 dia		7075-T6 Sheet "Z" Frame
No. 2	4 x 1 x 260 dia		7075-T6 Sheet "Z" Frame
Total Adapter	260" dia x 7" deep x 60 " long		————

Interfaces

Planetary Vehicle - 4 Attach Points 250 inches in Diameter bolt circle
Shroud - Frame and fitting distributed load attachment at shroud diameter

Figure 171. Preliminary Specification - Planetary Vehicle Adapter

The adapter fitting is in the form of a box end fitting (a 7075-T73 aluminum machined fitting), 4.5 inches high, 9.5 inches long, and 5.5 inches wide. Attachment to the planetary vehicle consists of a single tension bolt at each fitting which incorporates a separation device as described in Section 9.4 of this volume. Four bolts on the corners of the fitting provide the field joint attachment to the main support fittings.

The main support fittings uniformly distribute the vertical component of the planetary vehicle loads into the shroud through a double row of attachments over the 81.6-inch length of the fittings. The fittings are machined from 7075-T73 forgings. The upper end of the fitting attaches to the main frame and adapter fitting with an interchangeable bolt pattern.

The main frame at the separation plane consists of four identical 90 degree segments terminating at the four shroud vertical separation planes. If the shroud has only two separation joints, the frame would consist of two 180 degree segments. Each segment is machined from a 7075-T73 forging. The frame cross-section is a channel, 7 inches high with 2 by 0.25 inch caps. A bathtub type receptacle is machined into each end of each segment. The segments are attached to each other with single 5/8 inch diameter bolts with dual separation nuts which, when preloaded, transmit all tension, compression, shear, and moments across the joints. The frame transmits all transverse and torsional loads from the planetary vehicle into the nose fairing. The ends of the frame segments are 45 degrees from the spacecraft attach points. The frame segment separation bolts are fired at the time of shroud separation. It is not necessary to separate the aft planetary vehicle adapter; therefore, this frame can be designed as a continuous ring frame if there are significant design advantages.

Two intermediate frames are located 40.8 and 81.6 inches below the main frame for shroud stability and local moment reaction. Both frames are interrupted at the shroud joints, and the middle frame is also interrupted at the support fittings. Both frames are 4 inches deep, 7075-T6 aluminum sheet metal "Z" frames.

For this study, the shroud is assumed to be of aluminum honeycomb construction; however, the design is equally adaptable to semi-monocoque construction. Doublers are integrated into the shroud in the area of the support fittings to help distribute the vertical loads.

See Section II, 9.4.2 for planetary vehicle separation system component description and Section 10, 3.4.1 for description of the umbilical disconnect.

5. PARAMETERS AND PERFORMANCE SUMMARY

Margins of safety for the major structural members were computed in accordance with MIL-HDBK-5 procedures. The results are summarized in Table 93.

The planetary vehicle adapter weight presented in Table 94 is composed of the structural components, the umbilical components, the umbilical separation mechanism, and telemetry sensors. Since the weight specification allowed 1500 pounds for the adapter and the design weight is only 422 pounds, the weight margin is 1078 pounds. All weights are based on stress analysis and design information.

Table 93. Adapter Margins of Safety

Item	Description	Critical Load and Condition	Margins of Safety
Upper ring	2x7x0.125 Channel 7075-T6 Aluminum	192,000 in.-lb Moment launch release	+0.08 (crippling)
Lower ring	1x4x0.1 Channel 7075-T6 Aluminum	46,500 in.-lb Moment launch release	+0.20 (crippling)
Longitudinal fittings	Machined fitting 7075-T6 Aluminum	60,500# Compression launch release	+0.06 (crippling)
Separation bolt	5/8 diameter Bolt 160,000 H.T. Steel	36,140# Tension launch release	+0.20 (tension)

Table 94. Planetary Vehicle Adapter Weight Breakdown

Item	Weight (lbs)
Structural and Mechanical Subsystem	(377)
Adapter fittings	26.0
Main support fittings	60.5
Upper frame	126.0
Mid frame	58.1
Lower frame	58.1
Doublers (nose fairing)	28.0
Attachments and miscellaneous (6 per cent)	19.8
Umbilical system	8.0
Telemetry sensors and attachments	5.0
Separation mechanism	32.0
Margin	(1078.0)
Planetary Vehicle Adapter	(1500.0)